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R107-014

RESTRICTED

SUBJECT

BASIC FLIGHT CRITERIA

PACK OFF

MODEL: **XC-120**

FAIRCHILD AIRCRAFT

Division of

FAIRCHILD ENGINE & AIRPLANE CORPORATION

HAGERSTOWN, MARYLAND

RESTRICTED

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BASIC FLIGHT CRITERIA - PACK OFF

PART I-A

A. INTRODUCTION

It is the purpose of this report to determine the aerodynamic characteristics of the complete airplane (pack off) 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 consists of the determination of the aerodynamic characteristics of the complete airplane (pack-off) and the critical loads on the component parts for symmetrical flight, considering pitch only; Part III covers the aerodynamic characteristics of the complete airplane and the critical loads on the vertical tail surfaces for asymmetrical 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 asymmetrical flight, considering roll only. The four parts combined form one report, "Basic Flight Criteria", Volume II, Pack Off. It should be appreciated that this report, Volume II, covers only pack-off configuration. Volume I of this same report has been issued on 15 March 1949 and covers pack-on 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.

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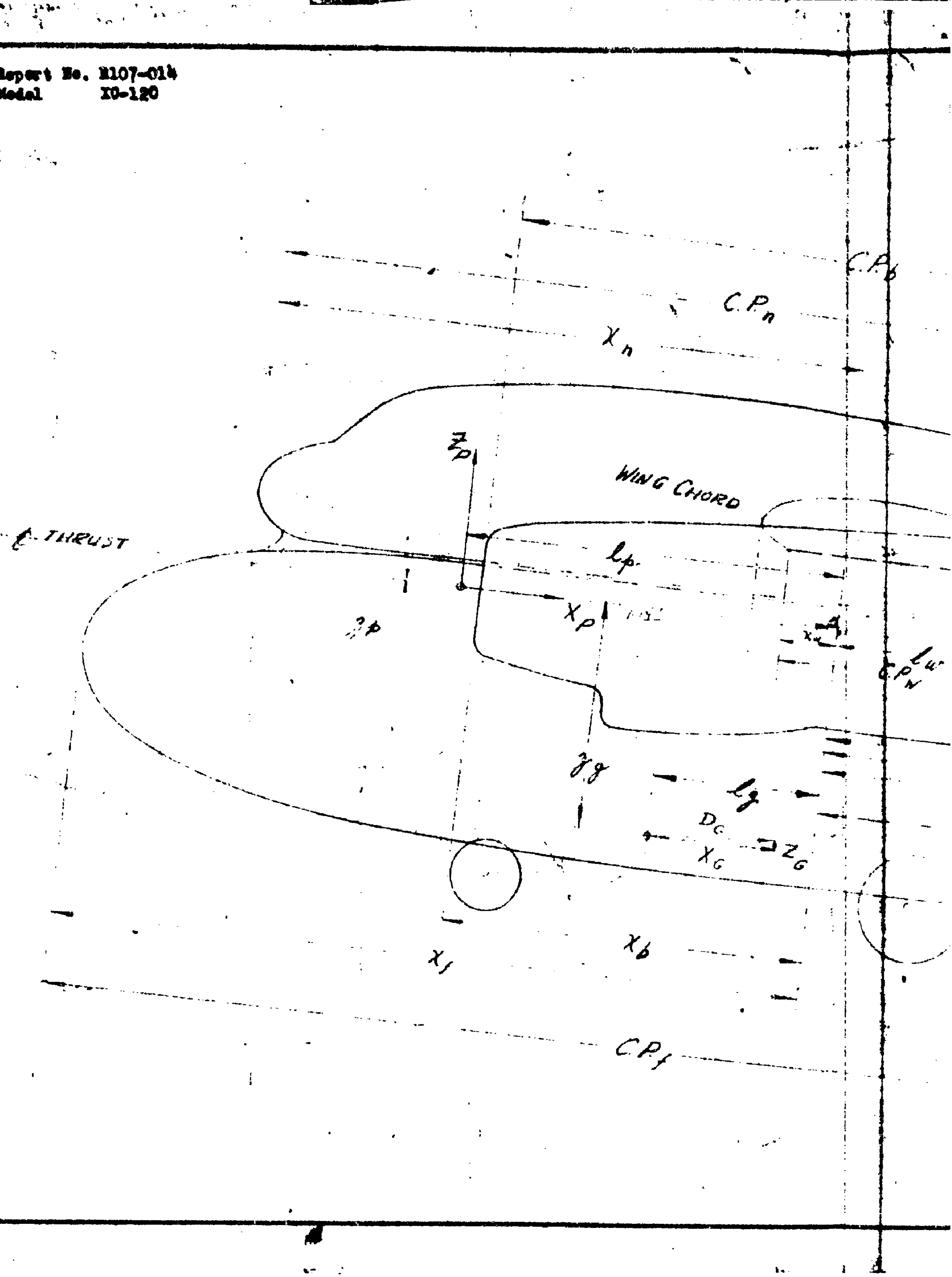
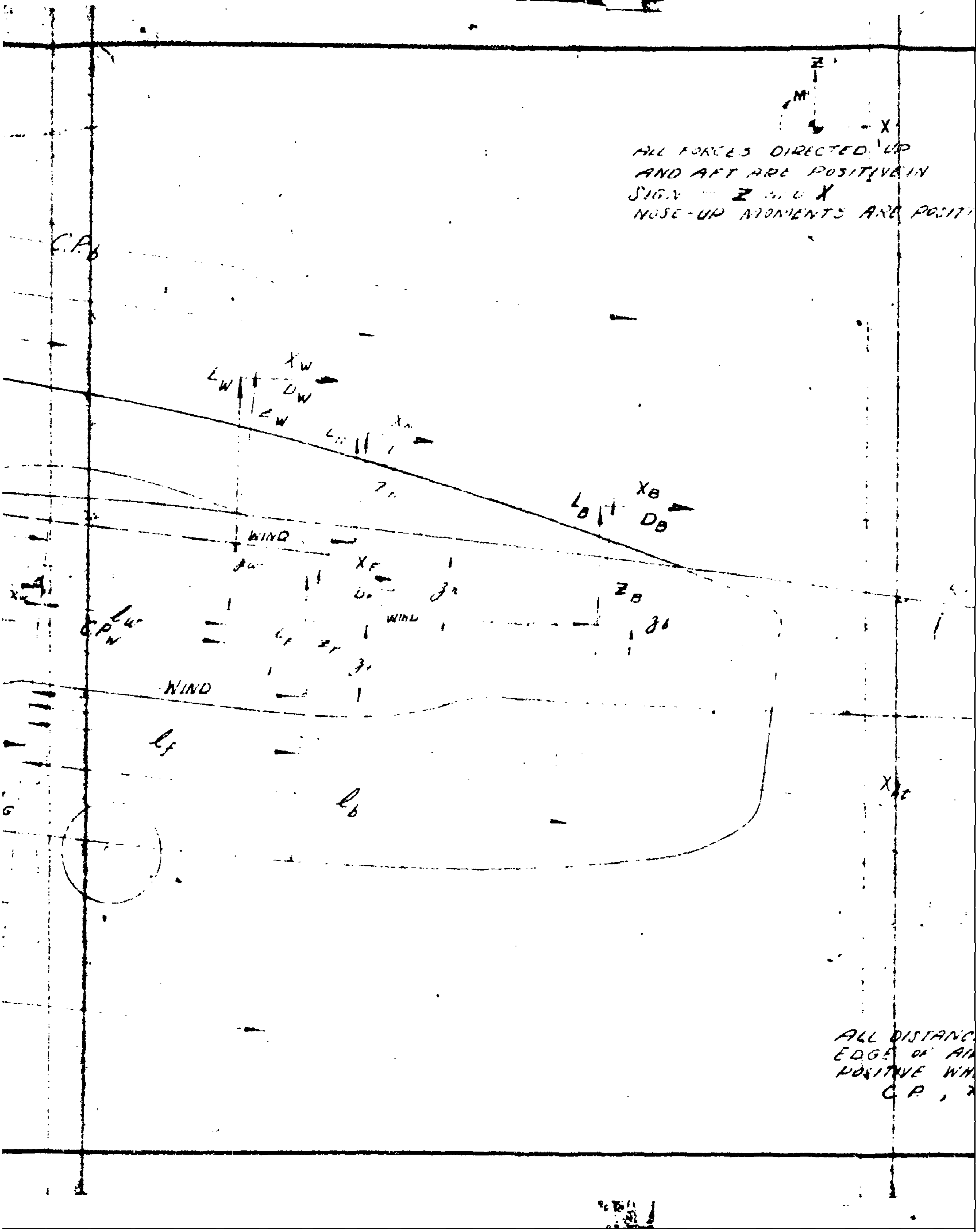
Model XO-120




Z
M
X
ALL FORCES DIRECTED UP
AND AFT ARE POSITIVE IN
SIGN — Z and X
NOSE-UP MOMENTS ARE POSIT
C.P.b
XW
LW
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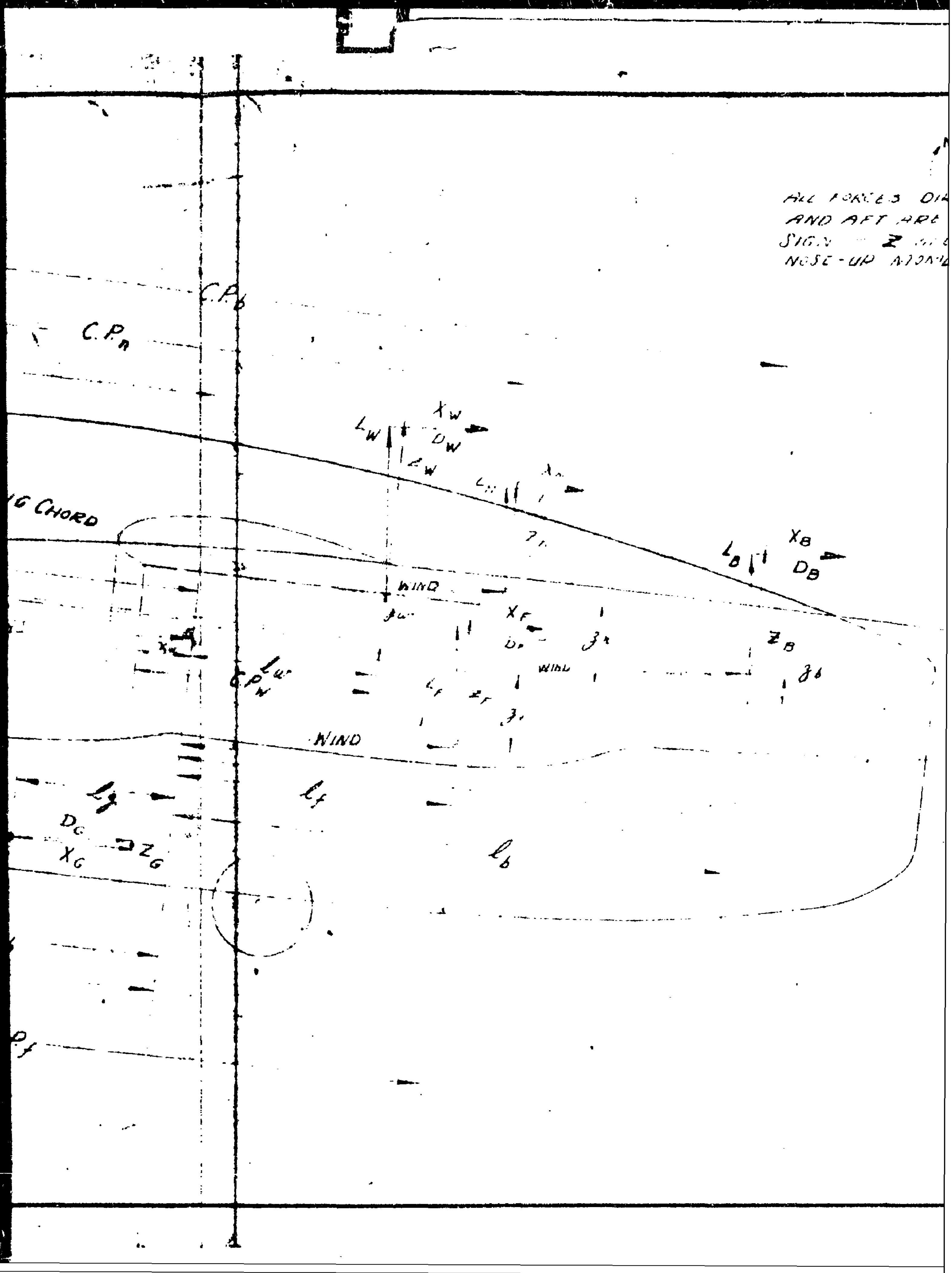

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Z_W
G CHORD
X_B
L_B
D_B
WIND
X_F
Z_B
$C.P._W$
l_f
l_b
D_G
X_G
Z_G

FIG. 1

Z+

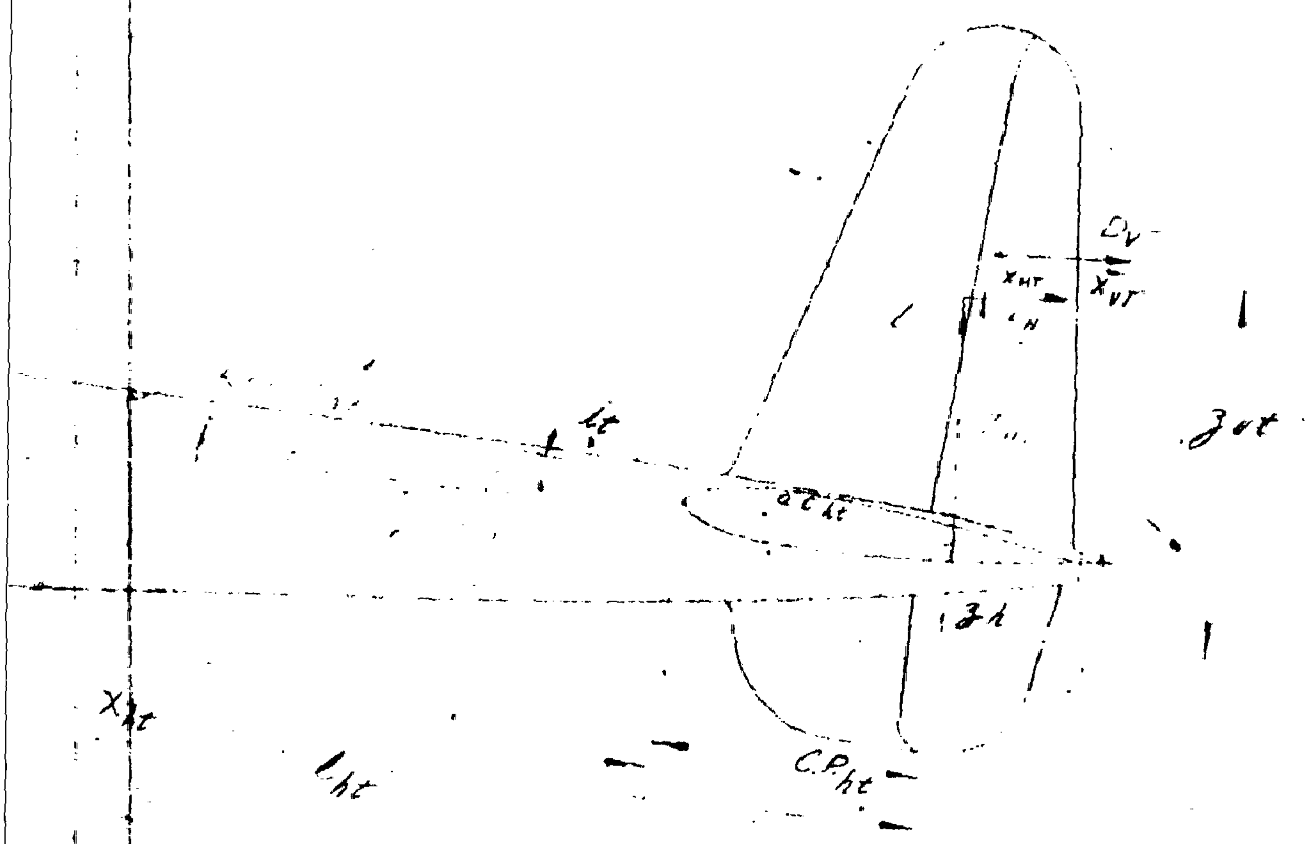
-X

...TED UP
...SITIVELY

...S ARE POSITIVE

-l+

ALL DISTANCES MEASURED UP FROM AND TO THE REAR OF THE C.G. ARE POSITIVE IN SIGN ~ Z AND l



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

REPORT NO. R107-014 | FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGES | PAGE 5

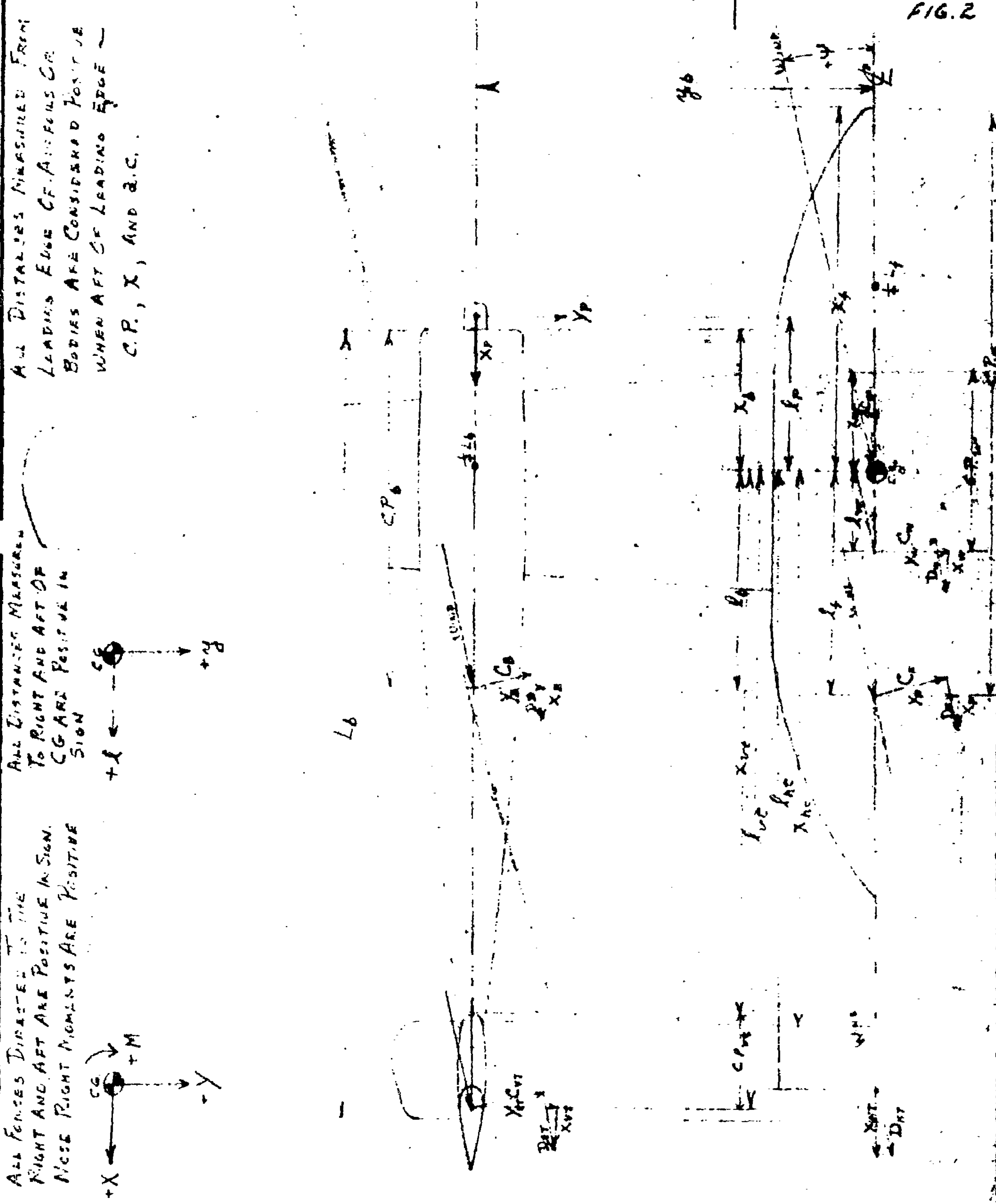
MODEL XC-120 | PREPARED BY | CHECKED BY | APPROVED BY

DATE February 18, 1949

Subject:— BASIC FLIGHT CRITERIA

REVISED

FIG. 2



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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).

PART I-C-1

1. Symmetrical Flight - Pitch

a. Wing

The wing loads as determined in Part II-1 are tabulated below with the possibly critical conditions.

Condition	Power	C.G. Loc.	Gross Weight Lbs.	Airplane Load Factor	C_{L_W}	$V\sqrt{\sigma}$ mph	Z wing load Lbs.	X wing load Lbs.	Moment about AC_W Ft.Lbs.
1. PHAA	off	3	54000	3.0	1.4010	176.9	162,361	-11,829	-102,353
(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. $PLAA_{VH}$	off	1	54000	3.0	.5761	265	175,736	7,174	-229,631
(1) Upper surface in compression. (2) Outer panel to center section splice, front and rear spar. (3) Lower surface in tension, center section. (4) Front spar, lower cap, center section.									
3. $PLAA_{VD}$	off	1	54000	2.425	.3924	331.3	158,800	15,072	-358,895
(1) Rear spar, outer panel.									
4. NHAA	off	1	54000	-1.5	-.5670	176.3	-62,413	-14,358	-101,637
(1) Lower surface in compression. (2) Upper surface outer panel to center section splice.									
5. $NLAA_{VD}$	off	7	54000	-1.079	-.0387	331.3	-15,965	356	-358,895
Outer panel skin splices.									
6. $PLAA_{VH}$ Gust	on	5	37015	3.732	.57845	265	160,965	8,777	-229,631
Local attachments of weight items.									
7. PHAA	off	1 & 3	54000	3.0	1.761	156.4	158,997	-16,205	- 79,987
Special leading edge conditions.									
8. NHAA	off	1 & 3	54000	-1.5	-.709	159.5	-64,827	-16,545	- 83,183
Special leading edge conditions.									

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 occurring in accelerated flight on the forward 20% of the wing.

b. Ailerons

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

c. Aileron Tabs

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

d. Flaps

The average load on the flaps as determined 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 107.39 lbs/sq.ft. on the flap area outside of the slipstream.

e. Horizontal Tail

The maximum horizontal tail balancing loads as determined in Part II-H, are -12,526 lbs. with power off and -13,574 lbs. with power on under the following conditions:

Condition	Power	C.G. Loc.	Gross Weight lbs.	Airplane Load Factor	α_{ht} deg.	δ_e deg.	$V\sqrt{\sigma}$ mph	$\frac{q_{ht}}{q}$	Horizontal Tail Loads - lbs.
$NLAA_{VD}$	off	7	54000	-1.079	-7.44	8.36	331.3	1.00	-12,526
$NLAA_{VD}$	on	7	54000	-1.079	-7.36	7.78	331.3	1.00	-13,574

PART I-C-1

e. Horizontal Tail (Cont.)

The maximum horizontal tail loads occurring with a 50 fps gust as determined in Part II-M-2 is -14,455 lbs. with a negative gust and the following conditions.

Condition	Power	C.G. Loc.	Gross Weight Loads	Airplane Load Factor	α_{ht} deg.	δ_e deg.	$V\sqrt{\sigma}$ mph	$\frac{q_{ht}}{q}$	Total Horiz. Tail Loads - lbs. Gust & Balance
Gust	On	1	54000	1.0	-7.593	5.25	265	1.00	-13,216

The maximum maneuvering tail loads as determined in reference (13) are 10,272 lbs. and -7687 lbs.

Condition	Power	C.G. Loc.	Gross Weight Loads	Airplane Load Factor	α_{ht} deg.	δ_e deg.	$V\sqrt{\sigma}$ mph	$\frac{q_{ht}}{q}$	Total Horiz. Tail Load - lbs.
Maneuvering	On	35% MAC_W Max. Upper	54000	3.0	.73	13.796	167	1.117	10,272
Maneuvering	On	20% MAC_W Max. Lower	54000	3.0	1.40	-13.90	167	1.117	-7,687

f. Elevator Tabs

The maximum load on the elevator trim tab as determined in Part II-M-1 is 94.3 lbs/sq.ft. of tab area.

The maximum load on the elevator spring tab as shown in Part II-M-2 is 83.2 lbs/sq.ft. of tab. area.

PART I-C-2

2. ASYMMETRICAL 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 5594 lbs. per vertical tail.

Condition	Power	α_{vt} deg.	δ_r deg.	$V\sqrt{\sigma}$ mph	$\frac{q_{vt}}{q}$	Vertical Tail Load - lbs.
50 fps Side Gust	On	7.3	0	265	1.020	5594

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 -1176 lbs. The maximum load on the other vertical tail is -828 lbs.

Condition	Power	α_{vt} deg.	δ_r deg.	$V\sqrt{\sigma}$ mph	$\frac{q_{vt}}{q}$	Vertical Tail Load - lbs.
Zero - Yaw	On Operating Engine Side	0	-16.1	100	1.42	-1176
Zero - Yaw	Off Dead Engine Side	0	-16.1	100	1.00	- 828

PART I-C-2

2a. Vertical Tail (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 -3335 lbs., with the gust load additive. The maximum load on the other vertical tail is -3200 lbs., with the gust load additive.

Condition	Power	α_{vt} deg.	δ_r deg.	$V\sqrt{\sigma}$ mph	$\frac{q_{vt}}{q}$	Vertical Tail Load - lbs.
Zero Yaw 30 fps Gust	On - Operating Engine Side	-5.833	-3.32	200	1.042	-3335
Zero Yaw 30 fps Gust	Off - Dead Engine Side	-5.833	-3.32	200	1.00	-3200

For 5° yaw two engine operation with the resulting rudder deflection at V_D the maximum load on each vertical tail is 1290 lbs.

Condition	Power	α_{vt} deg.	δ_r deg.	$V\sqrt{\sigma}$ mph	$\frac{q_{vt}}{q}$	Vertical Tail Load - lbs.
5° Yaw	On	4.327	-6.76	331.3	1.016	1290

For 5° yaw two engine operation with zero rudder deflection at V_D, the maximum load on each vertical tail is 5150 lbs.

Condition	Power	α_{vt} deg.	δ_r deg.	$V\sqrt{\sigma}$ mph	$\frac{q_{vt}}{q}$	Vertical Tail Load - lbs.
5° Yaw	On	4.327	0	331.3	1.016	5150

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.

PART I-C-3

3. ASSYMETRICAL FLIGHT - ROLL

a. Aileron Loads

The maximum aileron load, determined in Part IV-D, obtained in meeting the required rates of roll is -1515 lbs. and occurs at $V\sqrt{\sigma}$ = 212 mph on the right aileron with δ_{a_R} = -24° (Left aileron δ_{a_L} = +12°).

Condition	Aileron Deflection	Airplane Load Factor	Aileron Load lbs.	$V\sqrt{\sigma}$ mph
$\frac{p\,B_w}{2\,V} = .08$	-24°	-1.0	-1515	212

b. Aileron Tabs

The maximum loads on the aileron trim and balance tab as determined in Part IV - did not exceed the minimum design requirements of 50 lbs./sq. ft. Therefore, the aileron tab loads for design 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.

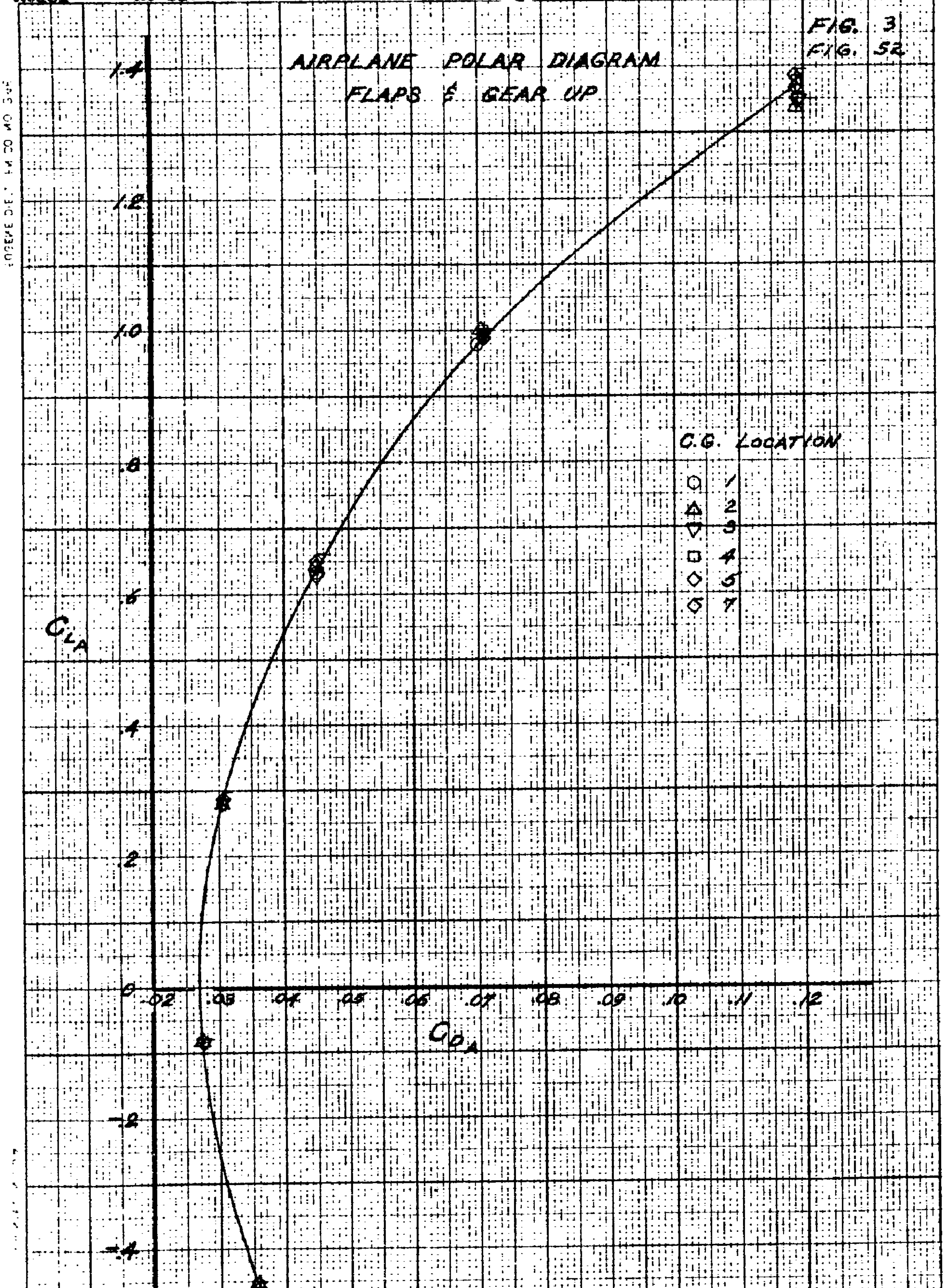


FIG. 3
FIG. 52
AIRPLANE POLAR DIAGRAM
FLAPS & GEAR UP
C.G. LOCATION
1
2
3
4
5
7
C_{L_A}
C_{D_A}
1.4
1.2
1.0
.8
.6
.4
.2
0
-.2
-.4
.02
.03
.04
.05
.06
.07
.08
.09
.10
.11
.12

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.

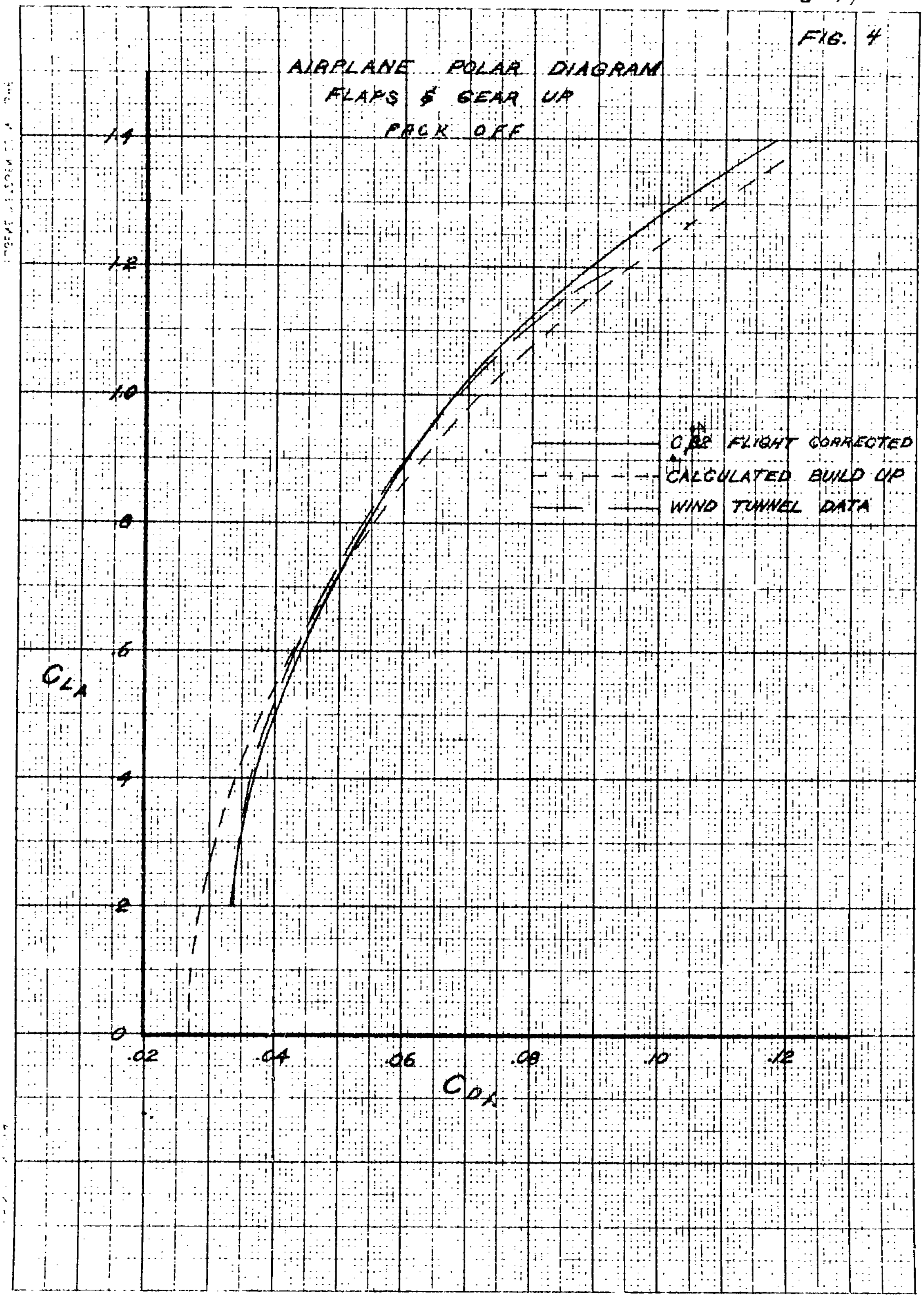


FIG. 4
AIRPLANE POLAR DIAGRAM
FLAPS & GEAR UP
PACK OFF
C_{L_A}
C_{D_A}
1.4
1.2
1.0
.8
.6
.4
.2
0
.02
.04
.06
.08
.10
.12
FLIGHT CORRECTED
CALCULATED BUILD UP
WIND TUNNEL DATA

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 25% 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. Later data indicated an upper aft C. G. location at 30% of the mean aerodynamic chord of the wing. 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	Aft and highest
3	Most forward and lowest
4	Most aft and lowest
5	Minimum flying weight, flaps up
6	Minimum flying weight, flaps down
7	Most aft and highest

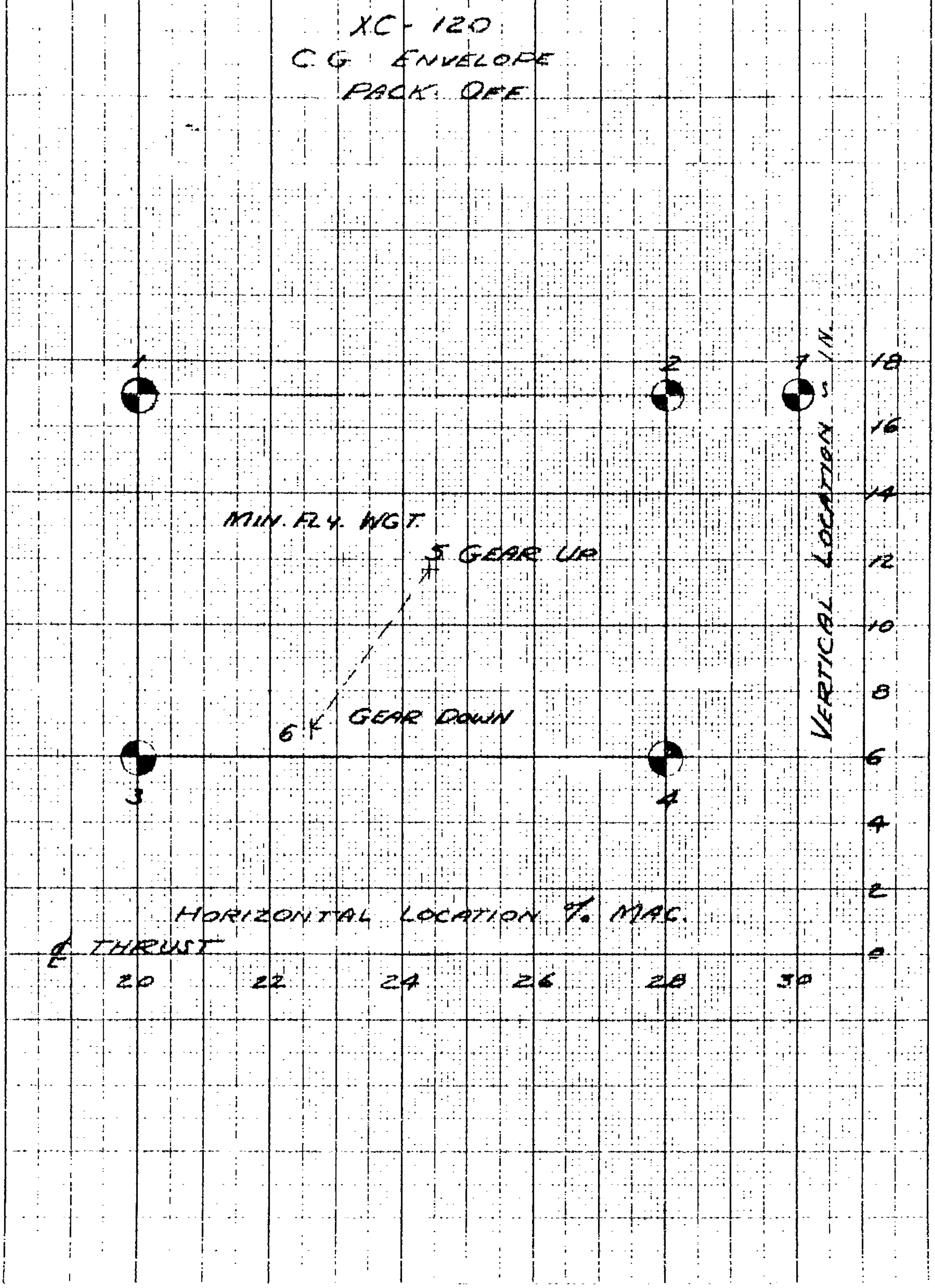
C. G. locations 1 2 3 4 5 and 7 as shown on the diagram were all considered as flaps up conditions while 1 2 3 4 6 and 7 were considered as flaps and gear down conditions.

PART I-E

ANALYSIS OF LOADING CONDITIONS CONSIDERED

	Weight, lbs.
Design weight (normal take-off weight) Specified by contractor in reference (6)	54,000
Maximum landing weight Specified by contractor in reference (6) as being 60,000 lbs. for pack-on but limited by take-off weight for pack-off.	54,000
Light weight condition This condition selected for zero fuel in the wing Design gross weight 54,000 Fuel weight 16,020 37,980 lbs.	37,980
Minimum flying weight As defined by reference (2) Weight empty 35,344 Pilot & Co-pilot 400 5% of fuel plus trapped fuel 901 25% of oil plus trapped oil 370 37,015	37,015
Take off Weight Considered as design weight for normal take off	54,000

FIG. 5


XC-120
C.G. ENVELOPE
PACK OFF
MIN. FLY. WGT.
5 GEAR UP
6 GEAR DOWN
1
2
7
3
4
VERTICAL LOCATION IN.
18
16
14
12
10
8
6
4
2
0
HORIZONTAL LOCATION % MAC.
℄ THRUST
20
22
24
26
28
30

PART I-F

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. Loc.	Gust Loads	Comments
Design Take-Off Max. Landing Light Alt.	3.0,-1.5 3.0,-1.5 3.0,-1.5 3.0,- 1.5	up up up up	Off & On Off & ON Off & On Off & On	envelope envelope envelope	All conditions to be checked for gust load from minimum flying speed to high speed level flight	No Pitching Acceleration Required
Minimum Flying Wt.	3.0,-1.5	up	Off & On	fixed	$V\sigma^{\frac{1}{2}}$ = 50 K fps	
Design Take-Off Max. Landing	None Specified Except Gust	down down down	Off & On Off & On Off & On	envelope envelope envelope	All conditions to be checked for gust load from minimum flying speed to $V\sqrt{\sigma}$ = 160 mph(3)	No Pitching Acceleration Required
Light Alt.		down	Off & On	envelope		
Minimum Flying Wt.		down	Off & On	fixed	$V\sigma^{\frac{1}{2}}$ = 50 K fps	

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.

(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.

PART I-F-1 (Cont.)

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.

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.

PART I-F-1 (Cont.)

d. Elevator Tabs (Cont.)

3. For balancing condition the 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 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. Asymmetrical 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 dege 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 $(1 - \eta/7.33)$ where η 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 $\sqrt{\sigma}\, V_S$ for maneuvering distribution.

(η = 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 aerodynamics characteristics of the component parts which are summed up to obtain the complete airplane aerodynamic characteristics.

The actual physical characteristics are the same as presented in Part I-C of reference (1).

The table 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 OFF

PART I-G

Dimensions from Various C. G. Locations

C. G. Location	1	2	3	4	5	6	7
% MAC_w	20.0	28.0	20.0	28.0	24.4 Min. Flying Wt.	22.6	30.0
Dimensions, Ins.							
x_w	33.66	47.10	33.66	47.10	41.02	38.00	50.48
x_n	268.1	281.6	268.1	281.6	275.6	272.8	284.9
x_b	166.1	179.6	166.1	179.6	173.6	180.8	182.9
x_{wl}	40.75	54.20	40.75	54.20	48.15	45.09	57.57
z_w	18.89	18.89	29.89	29.89	24.12	29.14	18.89
z_n	31.83	31.83	42.83	42.83	37.06	44.08	31.83
z_b	-17.00	-17.00	-6.00	-6.00	-11.70	-6.80	-17.00
z_p	-17.0	-17.0	-6.0	-6.0	-11.7	-6.8	-17.0
z_{wl}	5.87	5.87	16.87	16.87	11.10	16.12	5.87
z_g	-95.0	-95.0	-84.0	-84.0	-89.7	-84.8	-95.0
z_{vt}	40.65	40.65	51.65	51.65	45.95	50.85	40.65
l_g	31.5	18.04	31.5	18.04	24.1	27.12	14.7
l_p	-174.5	-187.96	-174.5	-187.96	-181.9	-178.88	-191.3
l_{ht}	621.1	607.64	621.1	607.64	613.7	616.7	604.2
l_{vt}	621.1	607.64	621.1	607.64	613.7	616.7	604.2

Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART I-G

Dimensionless Ratios

C. G. Location	1	2	3	4	5	6	7
% MAC_w	20.0	28.0	20.0	28.0	24.4 Min. Flying Wt.	22.6	30.0
$\frac{x_w}{MAC_w}$	.20	.28	.20	.28	.244	.226	.30
$\frac{AC_w}{MAC_w}$	.256	.256	.256	.256	.256	.256	.256
$\frac{x_w}{MAC_w} - \frac{AC_w}{MAC_w}$	-.056	.024	-.056	.024	-.012	-.030	.044
$\frac{z_w}{MAC_w}$	.1123	.1123	.1780	.1780	.1436	.1732	.1123
$\frac{P_n}{S_w}$	.3109	.3109	.3109	.3109	.3109	.3109	.3109
$\frac{L_n}{MAC_w}$	3.58	3.58	3.58	3.58	3.58	3.58	3.58
$\frac{x_n}{MAC_w}$	1.593	1.673	1.593	1.673	1.636	1.623	1.693
$\frac{.25\, L_n}{MAC_w}$	.895	.895	.895	.895	.895	.895	.895
$\frac{x_n}{MAC_w} - \frac{.25\, L_n}{MAC_w}$	.698	.778	.698	.778	.741	.728	.798
$\frac{F_n}{S_w}$	.0407	.0407	.0407	.0407	.0407	.0407	.0407
$\frac{z_n}{MAC_w}$	.1892	.1892	.2549	.2549	.2205	.2620	.1892
$\frac{P_b}{S_w}$	.2159	.2159	.2159	.2159	.2159	.2159	.2159

PART I-G (Cont.)

Dimensionless Ratios

C. G. Location	1	2	3	4	5	6	7
% MAC_w	20.0	28.0	20.0	28.0	24.4	22.6	30.0
					Min. Flying Wt.		
$\frac{L_b}{MAC_w}$	4.861	4.861	4.861	4.861	4.861	4.861	4.861
$\frac{x_b}{MAC_w}$	.987	1.068	.987	1.068	1.031	1.074	1.087
$\frac{.25L_b}{MAC_w}$	1.215	1.215	1.215	1.215	1.215	1.215	1.215
$\frac{x_b}{MAC_w} - \frac{.25\,L_b}{MAC_w}$	-.228	-.147	-.228	-.147	-.184	-.141	-.128
$\frac{F_b}{S_w}$	.0282	.0282	.0282	.0282	.0282	.0282	.0282
$\frac{z_p}{MAC_w} = \frac{z_b}{MAC_w}$	-.1010	-.1010	-.0356	-.0356	.0695	-.0404	-.1010
$\frac{x_{wl}}{MAC_w}$	.2422	.3221	.2422	.3221	.2861	.2679	.3421
$\frac{AC_{wl}}{MAC_w}$	.2274	.2274	.2274	.2274	.2274	.2274	.2274
$\frac{x_{wl}}{MAC_w} - \frac{AC_{wl}}{MAC_w}$	.0148	.0947	.0148	.0947	.0587	.0405	.1147
$\frac{z_{wl}}{MAC_w}$	.0349	.0349	.1006	.1006	.0662	.0958	.0349
$\frac{S_{vt}}{S_w}$	.0690	.0690	.0690	.0690	.0690	.0690	.0690

PART I-G (Cont.)

Dimensionless Ratios

C. G. Location	1	2	3	4	5	6	7
% MAC_w	2.00	28.0	20.0	28.0	24.4	22.6	30.0
					Min. Flying Wt.		
$\frac{z_{vt}}{MAC_w}$	.2416	.2416	.3073	.3073	.2729	.3025	.2416
$\frac{S_{ht}}{S_w}$	.2392	.2392	.2392	.2392	.2392	.2392	.2392
$\frac{l_g}{MAC_w}$	.187	.107	.187	.107	.143	.161	.0873
$\frac{z_g}{MAC_w}$	-.5645	-.5645	-.4992	-.4992	-.5330	.5042	.5645

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 Report R107-013, "Maneuvering Horizontal Tail Loads - Model XC-120", dated 4 March 1949, Volume I.

(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.

(13) Fairchild Engineering Report R110-013, "Maneuvering Horizontal Tail Loads - Model XC-120", dated 12 April 1949, Volume II.

(14) NACA Advanced Confidential Report L5005, "Summary of Airfoil Data", by I. H. Abbott, A. E. von Doenhoff, and L. S. Stivers, dated 19 March 1945.

PART II

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. The method of analysis and method of combination is the same as presented in Volume I of this report, with the crew nacelle being substituted for the fuselage. Crew nacelle is designated by subscripts N or n as C_{L_N} - crew nacelle lift coefficient.

1. Angular Relationships

The angular relationship of each component part of the airplane to the thrust line is the same as presented in Volume I. For convenience they are again presented here with the addition of the angular relationship of the crew nacelle.

$$\alpha_c = \alpha_{TL} - 7.0^\circ$$

$$\alpha_n = \alpha_{TL}$$

$$\alpha_b = \alpha_{TL}$$

$$\alpha_{ht} = \alpha_{TL} + 1.0^\circ - \epsilon_{ave}$$

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 of Volume I of this report.

The aerodynamic characteristics of each 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

$$C_{M_W} = C_{M_{W_{AC}}} + C_{Z_W}(A) + C_{X_W}(B)$$

where $A = \left(\frac{x_W}{MAC_W} - \frac{AC_W}{MAC_W} \right)$

$$B = \frac{z_W}{MAC_W}$$

$C_{M_{W_{AC}}}$, C_{Z_W} and C_{X_W} are from figure 4 of reference (1)

C. G. Location	A	B
1	$A_1 = -.056$	$B_1 = .1123$
2	$A_2 = .024$	$B_2 = .1123$
3	$A_3 = -.056$	$B_3 = .1780$
4	$A_4 = .024$	$B_4 = .1780$
5	$A_5 = -.012$	$B_5 = .1436$
7	$A_7 = .044$	$B_7 = .1123$

Subject:— BASIC FLIGHT CRITERIA PACK OFF

PART II-B-1

PITCHING MOMENT OF WING UNFLAPPED

1	2	3	4	5	6	7	8	9	10	11
				C.G. #1			C.G. #2			
α_{TL}	$C_{M_{WAC}}$	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	-.05685	+.0182	-.00636	-.05116	-.0078	-.00684	-.07716	+.0182
-8		.00398	.00693	-.00022	.00078	-.06244	+.00009	+.00079	-.06212	-.0002
-4		.34820	.03645	-.0195	.00408	-.07812	+.0083	+.00412	-.05058	-.0195
0		.68500	.02700	-.0384	.00303	-.04837	+.01641	+.00305	-.04354	-.0384
+4		1.02270	-.01875	-.0572	-.00210	-.12230	+.0246	-.00213	-.04053	-.0572
+8		1.36180	-.09150	-.0762	-.0104	-.14960	+.0328	-.0104	-.04060	-.0762

12	13	14	15	16	17	18	19	20	21	22
C.G. #3		C.G. #4			C.G. #5			C.G. #7		
$C_{X_W} \cdot B_3$	C_{M_W}	$C_{Z_W} \cdot A_4$	$C_{X_W} B_4$	C_{M_W}	$C_{Z_W} A_5$	$C_{X_W} B_5$	C_{M_W}	$C_{Z_W} \cdot A_7$	$C_{X_W} B_7$	C_{M_W}
-.0104	-.05520	-.0078	-.0104	-.08120	+.0039	-.00818	-.06728	-.0143	-.00637	-.08367
+.00123	-.06197	+.00009	+.00123	-.06168	-.00004	+.00100	-.06200	.000175	.00078	-.06265
+.00650	-.07600	+.0083	+.00650	-.04820	-.0042	+.00525	-.06185	.0153	.00388	-.04282
+.00480	-.09600	+.01641	+.00480	-.04180	-.0082	+.00387	-.06733	.0302	.00303	-.02977
-.00334	-.12354	+.0246	-.00334	-.04174	-.0121	-.00268	-.07778	.0451	-.00210	-.02000
-.01630	-.15550	+.0328	-.01630	-.04650	-.0163	-.01310	-.09240	.0599	-.01030	-.01340

PART II-B-2

2. Wing Flapped $\delta_f = 40°$

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

where $A = \left(\frac{x_W}{MAC_W} - \frac{AC_W}{MAC_W}\right)$

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

$C_{M_{WF_{AC}}}$ $C_{X_{WF}}$ and $C_{Z_{WF}}$ are from figure 8 of reference (1)

C. G. Location

1	$A_1 = -.056$	$B_1 = .1123$
2	$A_2 = +.024$	$B_2 = .1123$
3	$A_3 = -.056$	$B_3 = .1780$
4	$A_4 = +.024$	$B_4 = .1780$
6	$A_6 = .030$	$B_6 = .1732$
7	$A_7 = .044$	$B_7 = .1123$

Subject: BASIC FLIGHT CRITERIA PACK OFF

PART II-B-2

PITCHING MOMENT OF WING FLAPPED $\delta_f = 40^\circ$

1	2	3	4	5	6	7	8	9	10	11
				C.G. #1			C.G. #2			
α_{TL}	$C_{M_{WF_{AC}}}$	$C_{Z_{WF}}$	$C_{X_{WF}}$	$C_{Z_{WF}} \cdot A_1$	$C_{X_{WF}} \cdot B_1$	$C_{M_{WF}}$	$C_{Z_{WF}} \cdot A_2$	$C_{X_{WF}} \cdot B_2$	$C_{M_{WF}}$	$C_{Z_{WF}} A_3$
-12	-.1497	.0046	.0388	-.00026	+.00437	-.14559	.00011	+.00437	-.11522	-.00026
-8		.3403	.0892	-.01900	+.01010	-.15860	.00820	+.01010	-.13140	-.01900
-4		.6806	.1038	-.03800	.01165	-.17605	.01640	.01165	-.12165	-.03800
0		1.0230	.0809	-.05720	.00910	-.19780	.0245	.00910	-.11610	-.05720
4		1.3645	.0198	-.07650	.00224	-.22396	.0326	.00224	-.11486	-.07650
8		1.7036	-.0685	-.09550	-.00770	-.25290	.0409	-.00770	-.11650	-.09550

12	13	14	15	16	17	18	19	20	21	22
C.G. #3		C.G. #4			C.G. #6			C.G. #7		
$C_{X_{WF}} \cdot B_3$	$C_{M_{WF}}$	$C_{Z_{WF}} A_4$	$C_{X_{WF}} \cdot B_4$	$C_{M_{WF}}$	$C_{Z_{WF}} \cdot A_6$	$C_{X_{WF}} \cdot B_6$	$C_{M_{WF}}$	$C_{Z_{WF}} A_7$	$C_{X_{WF}} B_7$	$C_{M_{WF}}$
.00691	-.14305	.00011	.00691	-.14268	-.000138	.00672	-.14312	+.00020	+.00437	-.14513
.01590	-.1528	.00820	.01590	-.1256	-.01021	.01542	-.14449	+.01500	+.01010	-.12460
.0184	-.1693	.01640	.0184	-.11490	-.02040	.01798	-.15212	+.03000	+.01165	-.10865
.0144	-.1925	.0245	.0144	-.1108	-.03060	.01101	-.16629	+.04510	+.00910	-.09550
.00350	-.2227	.0326	.0035	-.1136	-.04100	.00343	-.18727	+.06010	+.00224	-.08736
-.0123	-.2575	.0409	-.0123	-.1211	-.05110	-.01187	-.21267	+.07505	-.00770	-.08235

PART II-B-3

3. <u>Crew Nacelle</u>

$$C_{M_N} = C_{M_{N.25L_n}} (A) + C_{Z_N} (B) + C_{X_N} (C)$$

where

$$A = \left(\frac{P_n}{S_w} \frac{L_n}{MAC_w}\right)$$

$$B = \frac{P_n}{S_w}\left(\frac{x_n}{MAC_w} - \frac{.25\ L_n}{MAC_w}\right)$$

$$C = \left(\frac{F_n}{S_w} \frac{z_n}{MAC_w}\right)$$

$C_{M_{N.25L_n}}$ is from figure 23 reference (1)

C_{Z_N} and C_{X_N} are from figure 22 reference (1)

C. G. Location	A		
1	$A_1 = 1.115$	$B_1 = .2170$	$C_1 = .0077$
2	$A_2 = 1.115$	$B_2 = .2420$	$C_2 = .0077$
3	$A_3 = 1.115$	$B_3 = .2170$	$C_3 = .0104$
4	$A_4 = 1.115$	$B_4 = .2420$	$C_4 = .0104$
5	$A_5 = 1.115$	$B_5 = .2310$	$C_5 = .0090$
6	$A_6 = 1.115$	$B_6 = .2260$	$C_6 = .0107$
7	$A_7 = 1.115$	$B_7 = .2470$	$C_7 = .0077$

Subject: BASIC FLIGHT CRITERIA PACK CDT

PART II-B-3

PITCHING MOMENT OF CREW NACELLE

1	2	3	4	5	6	7	8	9	10	11
				C.G. #1				C.G. #2		
							5+6+7			
α_{TL}	$C_{M_{N.25L_N}}$	C_{X_N}	C_{Z_N}	2 x A_1	4 x B_1	3 x C_1	C_{M_N}	2 x A_2	4 x B_2	3 x C_2
-12	.0076	-.2526	-.2668	.0085	-.0580	-.0019	-.0514	.0085	-.0642	-.0019
-8	.0107	-.0037	-.1956	.0119	-.0424	-.0005	-.0310	.0119	-.0473	-.0005
-4	.0150	.0589	-.1421	.0167	-.0308	.0005	-.0136	.0167	-.0345	.0005
0	.0222	.1229	-.1100	.0247	-.0238	.0009	+.0018	.0247	-.0265	.0009
4	.0200	.1742	-.0710	.0290	-.0154	.0013	+.0149	.0290	-.0172	.0013
8	.0202	.1433	-.0026	.0224	.0006	.0011	+.0241	.0224	-.0006	.0011

12	13	14	15	16	17	18	19	20
CG #2	C.G. #3				C.G. #4			
$C_{M_{11}}$	2 x A_3	4 x B_3	3 x C_3	C_{M_N}	2 x A_4	4 x B_4	3 x C_4	C_{M_N}
-.0581	.0085	-.0580	-.0026	-.0521	.0085	-.0642	-.0026	-.0588
-.0359	.0119	-.0424	-.0007	-.0312	.0119	-.0473	-.0007	-.0361
-.0173	.0167	-.0308	.0006	-.0135	.0167	-.0345	.0006	-.0172
-.0009	.0247	-.0235	.0013	+.0022	.0247	-.0265	.0013	.0005
.0131	.0290	-.0154	.0018	.0154	.0290	-.0172	.0018	.0136
.0229	.0224	.0006	.0015	.0245	.0224	-.0006	.0015	.0233

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MODEL XC-120 PREPARED BY CHECKED BY APPROVED BY

DATE 1 May 1949

REVISED

Subject: BASIC FLIGHT CRITERIA PACK OFF

PART II-B-3

PITCHING MOMENT OF CREW NACELLE (CONT.)

	21	22	23	24	25	26	27	28
	C.G. #5				C.G. #6			
α_{TL}	2 x A_5	4 x B_5	3 x C_5	C_{M_N}	2 x A_6	4 x B_6	3 x C_6	C_{M_N}
-12	.0085	-.0615	-.0023	-.0553	.0085	-.0602	-.0027	-.0544
-8	.0119	-.0450	-.0006	-.0337	.0119	-.0441	-.0007	-.0329
-4	.0167	-.0328	.0005	-.0156	.0167	-.0322	.0006	-.0149
0	.0247	-.0254	.0011	.0004	.0247	-.0248	.0013	.0012
4	.0290	-.0164	.0015	.0141	.0290	-.0160	.0019	.0149
8	.0224	-.0006	.0013	.0231	.0224	-.0006	.0015	.0233

29	30	31	32
C.G. #7			
2 x A_7	4 x B_7	3 x C_7	C_{M_N}
.0085	-.0658	-.0019	-.0592
.0119	-.0487	-.0005	-.0369
.0167	-.0351	.0005	-.0179
.0247	-.0272	.0009	+.0016
.0290	-.0175	.0013	+.0128
.0224	-.0006	.0011	+.0229

PART III-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
$$A = \left(\frac{F_b}{S_w}\frac{L_b}{MAC_w}\right)$$

$$B = \frac{F_b}{S_w}\left(\frac{x_b}{MAC_w} - \frac{.25L_b}{MAC_w}\right)$$

$$C = \left(\frac{F_b}{S_w}\frac{z_b}{MAC_w}\right)$$

C. G. Location

1	$A_1 = 1.0495$	$B_1 = -.0492$	$C_1 = -.00285$
2	$A_2 = 1.0495$	$B_2 = -.0317$	$C_2 = -.00285$
3	$A_3 = 1.0495$	$B_3 = -.0492$	$C_3 = -.001005$
4	$A_4 = 1.0495$	$B_4 = -.0317$	$C_4 = -.001005$
5	$A_5 = 1.0495$	$B_5 = -.0397$	$C_5 = -.00188$
6	$A_6 = 1.0495$	$B_6 = -.0304$	$C_6 = -.00114$
7	$A_7 = 1.0495$	$B_7 = -.0254$	$C_7 = -.00285$

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PART II-B-4

PITCHING MOMENT OF BOOMS

1	2	3	4	5	6	7	8	9	10	11
				C.G. #1				C.G. #2		
α_{TL}	$C_{M_{B.25L_B}}$ (2)	C_{Z_B} (2)	C_{X_B} (2)	② × A_1	$C_{Z_B} B_1$	$C_{X_B} C_1$	C_{M_B}	② × A_2	$C_{Z_B} B_2$	$C_{X_B} C_2$
-12	-.013	-.0694	.358	-.0136	.00342	-.00162	-.01120	-.0136	.0022	-.00102
-8	-.005	-.0330	.378	-.0052	.00162	-.00108	-.00464	-.0052	.00105	-.00105
-4	.006	-.0084	.3792	.0063	.00041	-.00108	-.00455	.0063	.00027	-.00108
0	.0230	0	.378	.0241	0	-.00108	.02302	.0241	0	-.00108
4	.0470	+.0084	.3792	.0492	-.00041	-.00108	.04771	.0492	-.00027	-.00108
8	.0740	+.0330	.378	.0775	-.00162	-.00108	.07480	.0775	-.00105	-.00108

12	13	14	15	16	17	18	19	20
C.G. #2	C.G. #3				C.G. #4			
C_{M_B}	② × A_3	$C_{Z_B} B_3$	$C_{X_B} C_3$	C_{M_B}	② × A_4	$C_{Z_B} B_4$	$C_{X_B} C_4$	C_{M_B}
-.01242	-.0136	.00342	-.00036	-.01054	-.0136	.0022	-.00036	-.01176
-.00523	-.0052	.00162	-.00038	-.00396	-.0052	.00105	-.00038	-.00453
.00549	.0063	.00041	-.00038	.00633	.0063	.00027	-.00038	.00619
.02302	.0241	0	-.00036	.02372	.0241	0	-.00036	.02322
.04785	.0492	-.00041	-.00038	.04841	.0492	-.00027	-.00038	.04855
.07537	.0775	-.00162	-.00038	.07550	.0775	-.00105	-.00038	.07607

PITCHING MOMENT OF BOOMS (CONT.)

C.G. #5 / C.G. #6

	21	22	23	24	25	26	27	28
α_{TL}	② $\times A_5$	$C_{Z_B} B_5$	$C_{X_B} C_5$	C_{M_B}	② $\times A_6$	$C_{Z_B} B_6$	$C_{X_B} C_6$	C_{M_B}
-12	-.0136	.00276	-.00067	-.01151	-.0136	.00211	-.00041	-.0119
-8	-.0052	.00131	-.00071	-.00460	-.0052	.001	-.00043	-.00463
-4	.0063	.00033	-.00071	.00592	.0063	.00025	-.00043	.00612
0	.0241	0	-.00071	.02339	.0241	0	-.00043	.02367
4	.0492	-.00033	-.00071	.04816	.0492	-.00025	-.00043	.04852
8	.0775	-.00131	-.00071	.07649	.0775	-.001	-.00043	.07787

C.G. #7

29	30	31	32
② $\times A_7$	$C_{Z_B} B_7$	$C_{X_B} C_7$	C_{M_B}
-.0136	-.00176	-.00102	-.01638
-.0052	-.00084	-.00108	-.00712
.0063	-.00021	-.00108	+.00501
.0241	0	-.00108	+.02302
.0492	+.00021	-.00108	+.04833
.0775	+.00064	-.00108	+.07726

PART II-B-5

5. Vertical Tail

$$C_{M_{VT}} = 2 \left[C_{X_{VT}} \quad \frac{q_{vt}}{q} \quad \frac{S_{vt}}{S_w} \quad \frac{z_{vt}}{MAC_w} \right]$$

It is assumed that $C_{D_{VT}}$ does not vary with angle of attack so that from pages 234 and 238 of reference (1)

$$C_{X_{VT}} = C_{D_{VT}} = .0081$$

$$\text{for } \alpha_{vt} = 0^\circ \text{ and } \delta_r = 0^\circ$$

PART II-B-8

PITCHING MOMENT OF VERTICAL TAILS

1	2	3	4	5	6	7	8
C.G.	$C_{D_{VT}}$	$2C_{X_{VT}}$	$\frac{q_{vt}}{q}$	$\frac{S_{VT}}{S_W}$	$\frac{Z_{VT}}{MAC_W}$	3 × 5	$C_{M_{VT}}$
1	.0081	.01644	1.0	.0690	.2416	.00114	.00028
2					.2416		.00028
3					.3073		.00031
4					.3073		.00035
5					.2729		.00031
6					.3025		.00035
7					.2416		.00028

PART II-B-6

6. Landing Gear

$$C_{M_G} = 2\left[C_{X_G}(A) - C_{Z_G}(B)\right]$$

where $A = \left(\frac{z_g}{MAC_w}\right)$

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

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

C. G. Location	A	B
1	-.5645	.187
2	-.5645	.107
3	-.4279	.187
4	-.4279	.107
5	-.5330	.143
6	-.5042	.161
7	-.5645	.0873

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PITCHING MOMENT OF LANDING GEAR

1	2	3	4	5	6	7	8	9	10	11
			C.G. #1			C.G. #2			C.G. #3	
α_{FL}	$2C_{X_{LG}}$	$2C_{Z_{LG}}$	(2) x A_1	(3) x B_1	(4) - (5) $C_{M_{LG}}$	(2) x A_2	(3) x B_2	(7) - (8) $C_{M_{LG}}$	(2) x A_3	(3) x B_3
-12	.0642	.0136	-.0362	.0025	-.0387	-.0362	.001455	-.03766	-.0320	.0025
-8	.0650	.0091	-.0366	.0017	-.0383	-.0366	.000974	-.03757	-.0324	.0017
-4	.0654	.0046	-.0369	.0009	-.0378	-.0369	.000492	-.03739	-.0326	.0009
0	.0656	0	-.0370	0	-.0370	-.0370	0	-.0370	-.0327	0
4	.0654	.0046	-.0369	.0009	-.0378	-.0369	.000492	-.03739	-.0326	.0009
8	.0650	.0091	-.0366	.0017	.0383	-.0366	.000974	-.03757	-.0324	.0017

12	13	14	15	16	17	18	19	20	21
C.G. 3	C.G. #4			C.G. #6			C.G. #7		
(2) - (11) $C_{M_{LG}}$	(2) x A_4	(3) x B_4	(13) - (14) $C_{M_{LG}}$	(2) x A_6	(3) x B_6	(16) - (17) $C_{M_{LG}}$	(2) x A_7	(3) x B_7	$C_{M_{LG}}$ (19) - (20)
-.0345	-.0320	.001455	-.03345	-.0324	.0022	-.0346	-.0362	.00119	-.03739
-.0341	-.0324	.000974	-.03337	-.0327	.0015	-.0342	-.0366	.00080	-.03740
-.0335	-.0326	.000492	-.03309	-.0329	.0008	-.0337	-.0369	.00040	-.03730
-.0327	-.0327	0	-.0327	-.0330	0	-.0330	-.0370	0	-.03700
-.0335	-.0326	.000492	-.03309	-.0329	.0008	-.0337	-.0369	.00040	-.03730
-.0341	-.0324	.000974	-.03337	-.0327	.0015	-.0342	-.0366	.00080	-.03740

PART II-B-7

7. Propellers

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

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

C_{Z_p} is 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 6 through 12 .

C. G. Location	ℓ_p	z_p
1	-174.5	-17.0
2	-187.96	-17.0
3	-174.5	- 6.0
4	-187.96	- 6.0
5	-181.9	-11.7
6	-178.88	- 6.8
7	-191.3	-17.0

C_{Z_p} and C_{X_p} for one propeller - Military Rated Power

$V\sigma^{\frac{1}{2}}$	T_c	α_{TL}	C_{Z_p}	C_{X_p}
265	.047	-12	-.058	-.115
		- 8	-.039	
		- 4	-.019	
		0	0	
		4	.019	
		8	.039	
331.3	.024	-12	-.066	-.058
		- 8	-.044	
		- 4	-.022	
		0	0	
		4	.022	
		8	.044	

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PART II-B-7

PITCHING MOMENT OF PROPELLERS

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

q = 25.6

1	2	3	4	5	6	7	8	9	10	11
				C.G. #1				C.G. #2		
α_{TL}	X_p	Z_p	$\frac{2}{q S_w MAC_w}$	$-l_p Z_p$	$z_p X_p$	(5)+(6)	C_{M_p}	$-l_p Z_p$	$z_p X_p$	(9)+(10)
-12	-7400	-246.6	3.21×10^{-7}	-43,032	125,700	82,668	.0265	-46,400	125,700	79,300
-8		-169.6		-29,595		96,105	.0308	-31,900		93,800
-4		-83.4		-14,553		111,147	.0356	-15,670		110,030
0		0		0		125,700	.0404	0		125,700
4		83.4		14,553		140,253	.0450	15,670		141,370
8		169.6		29,595		155,295	.0499	31,900		157,600

12	13	14	15	16	17	18	19	20	21	22
C.G. #2	C.G. #3				C.G. #4				C.G. #5	
C_{M_p}	$-l_p Z_p$	$z_p X_p$	(13)+(14)	C_{M_p}	$-l_p Z_p$	$z_p X_p$	(17)+(18)	C_{M_p}	$-l_p Z_p$	$z_p X_p$
.0254	-43,032	44,400	1,368	.0004	-46,400	44,400	-2000	-.0006	-44,400	86,600
.0301	-29,595		14,805	.0048	-31,900		12,500	.0040	-30,500	
.0353	-14,553		29,847	.0096	-15,670		28,730	.0092	-15,000	
.0401	0		44,400	.0142	0		44,400	.0142	0	
.0454	14,553		58,953	.0189	15670		60,070	.0193	15,000	
.0506	29,595		73,995	.0237	31900		76,300	.0245	30,500	

PART II-B-7

PITCHING MOMENT OF PROPELLERS (Cont.)

$T_c = .639$ $V_{TO} = 100$ mph

$q = 25.6$

23	24	25	26	27	28	29	30	31	32
C.G. 5									
(21)+(22)	C_{m_p}	$-\ell_p Z_p$	$\delta_p X_p$	(25)+(26)	C_{m_p}	$-\ell_p Z_p$	$\delta_p X_p$	(29)+(30)	C_{m_p}
42,200	.0135	-43,600	50,352	6,550	.0021	-47,200	125,700	78,500	.0252
56,100	.0180	-30,100		20,350	.0065	-32,300		93,400	.0301
71,600	.0230	-14,800		35,550	.0114	-15,950		109,750	.352
86,600	.0278	0		50,352	.0162	0		125,700	.0404
[illegible]	.0326	14,800		65,150	.0209	+15,950		141,650	.0455
[illegible]	.0376	30,100		80,450	.0258	+32,300		158,000	.0521

1	2	3	4	5	6	7	8	9	10	11
α_{TL}	X_p	Z_p	$\frac{2}{q S_w MAC_w}$	$-\ell_p Z_p$	$\delta_p X_p$	(5)+(6)	C_{m_p}	$-\ell Z_p$	$\delta_p X_p$	(9)+(10)
-12	-5900	-575.3	1.254×10^{-7}	-100,390	100,300	-90	0	-108,000	100,300	-7,700
-8		-390.9		-68,212		32,088	.0040	-73,500		26,800
-4		-198.4		-34,621		65,679	.0083	-37,300		66,500
0		0		0		100,300	.0126	0		100,300
4		198.4		34,621		134,921	.0169	37,300		137,600
8		390.9		68,212		166,512	.0212	73,500		173,800

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PITCHING MOMENT OF PROPELLERS

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

$q = 65.5$

12	13	14	15	16	17	18	19	20	21	22
C.G. 2	C.G. #3					C.G. #4			C.G. #5	
C_{m_p}	$-l_p Z_p$	$z_p X_p$	(13)+(14)	C_{m_p}	$-l_p Z_p$	$z_p X_p$	(17)+(18)	C_{m_p}	$-l_p Z_p$	$z_p X_p$
-.0015	-100,390	35,400	-64,990	-.0080	-108,000	35,400	-72,600	-.0091	-103,600	69,000
.0031	-68,212		-32,812	-.0041	-73,500		-38,100	-.0045	-70,400	
.0062	-34,600		779	.0001	-37,300		1,900	-.0002	-35,700	
.0126	0		35,400	.0045	0		35,400	.0045	0	
.0177	34,621		70,021	.0088	37,300		72,700	.0091	35,700	
.0218	68,212		103,612	.0130	73,500		108,900	.0137	70,400	

23	24	25	26	27	28	29	30	31	32
C.G. #5		C.G. #6				C.G. #7			
(21)+(22)	C_{m_p}	$-l_p Z_p$	$z_p X_p$	(25)+(26)	C_{m_p}	$-l_p Z_p$	$z_p X_p$	(29)+(30)	C_{m_p}
-34,600	-.0043	-102,000	40,100	-61,900	-.0078	-110500	100 300	-10,200	-.0013
-1,400	-.0002	-69,400		-29,300	-.0037	-74600		+25,700	+.0032
33,200	.0042	-35,200		4,900	.0006	-36,000		+62,300	+.0078
69,000	.0087	0		40,100	.0050	0		+100,300	+.0126
104,700	.0131	35,200		75,300	.0095	+3[illegible]		+138,300	+.0174
139,400	.0175	69,400		109,500	.0137	+74600		+174,900	+.0219

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PART II-B-7

PITCHING MOMENT OF PROPELLER

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

$q = 87.5$

1	2	3	4	5	6	7	8	9	10	11
				C G #1				C.G. #2		
α_{TL}	X_p	Z_p	$\frac{2}{q S_w MAC_w}$	$-l_p Z_p$	$3_p X_p$	(5)+(6)	C_{M_p}	$-l_p Z_p$	$3_p X_p$	(9)+(10)
-12	-5.276	-787.5	9.38x10⁻⁴	-137,300	89,600	-47,700	-.0045	-148,000	89,600	-58,400
-8		-546.0		-95,200		-5,600	-.0005	-102,700		-13,100
-4		-273.0		-47,600		42,000	.0039	-51,300		38,300
0		0		0		89,600	.0084	0		89,600
4		273.0		47,600		137,200	.0129	51,300		140,900
8		546.0		95,200		184,800	.0173	102,700		192,300

12	13	14	15	16	17	18	19	20	21	22
C.G. #2	C.G. #3				C G. #4				C.G. #5	
C_{M_p}	$-l_p Z_p$	$3_p X_p$	(13)+(14)	C_{M_p}	$-l_p Z_p$	$3_p X_p$	(17)+(18)	C_{M_p}	$-l_p Z_p$	$3_p X_p$
-.0053	-137,300	31,600	-105,700	-.0099	-148,000	31,600	-116,400	-.0109	-141,700	61,600
-.0012	-95,200		-63,600	-.0060	-102,700		-71,100	-.0067	-98,200	
.0036	-47,600		-16,000	-.0015	-51,300		-19,700	-.0018	-49,100	
.0084	0		31,600	.0030	0		31,600	.0030	0	
.0132	47,600		79,200	.0074	51,300		82,900	.0078	49,100	
.0180	95,200		126,800	.0119	102,700		134,300	.0126	98,200	

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PART II-B-7

PITCHING MOMENT OF PROPELLERS

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

q = 87.5

23	24	25	26	27	28	29	30	31	32
C.G. #5		C.G. #6				C.G. #7			
(21)+(22)	C_{MP}	$-\ell_p Z_p$	$\bar{z}_p X_p$	(25)+(26)	C_{MP}	$-\ell_p Z_p$	$\bar{z}_p X_p$	(29)+(30)	C_{MP}
-36,100	-.0075	-139,700	35,800	-103,900	-.0098	-150500	89,600	-60,900	-.0057
-36,600	-.0034	-97,000		-61,200	-.0057	-104500		-14,800	-.0014
18,500	.0012	-48,500		-12,700	-.0012	-52000		37600	.0035
61,600	.0058	0		35,800	.0034	0		+89,600	+.0084
110,200	.0104	48,500		84,300	.0079	52000		+141600	.01326
157,600	.0150	97,000		132,800	.0125	+104500		+194,100	+.0182

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PITCHING MOMENT OF PROPELLERS

T_c = .047 $V\bar{\sigma}^{\frac{1}{2}}$ = 265 mph

q = 180

1	2	3	4	5	6	7	8	9	10	11
				C.G. #1				C.G. #2		
α_{TL}	X_P	Z_P	$\bar{q} S_w MAC_w^2$	$-Z_P C_P$	$X_P J_P$	(5)+(6)	C_{M_P}	$-Z_P C_P$	$X_P J_P$	(9)+(10)
-12	-3660	-186	4.53×10^{-8}	-324500	62,500	-262000	-.0119	-349000	62500	-286500
-8		-1252		-218000		-155500	-.0071	-234000		-171500
-4		-610		-106500		-44000	-.0020	-114400		-51,500
0		0		0		62500	.0028	0		62,500
4		610		106500		169000	.0077	114400		175900
8		1252		218000		280000	.0127	234000		296500

12	13	14	15	16	17	18	19	20
C.G. #2	C.G. #3				C.G. #4			
C_{M_P}	$-Z_P C_P$	$X_P J_P$	(13)+(14)	C_{M_P}	$-Z_P C_P$	$X_P J_P$	(17)+(18)	C_{M_P}
-.0131	-324500	22100	-302400	-.0136	-349000	22100	-326900	-.0148
-.0078	-218000		-195900	-.0089	-234000		-211900	-.0097
-.0023	-106500		-84400	-.0038	-114400		-92300	-.0042
.0028	0		22100	.0010	0		22100	.0010
.0081	106500		128600	.0059	114400		136500	+.0062
.0135	218000		240100	.0109	234000		256100	+.0117

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PART II-B-7

PITCHING MOMENT OF PROPELLERS

T_c = .047 $\quad V\sigma^{\frac{1}{2}}$ = 265 mph

q = 180

21	22	23	24	25	26	27	28	29
	C.G. #5				C.G. #6			
α_{TL}	$-Z_p \ell_p$	$X_p \zeta_p$	(22)+(23)	C_{M_P}	$-Z_p \ell_p$	$X_p \zeta_p$	(26)+(27)	C_{M_P}
-12	-334000	43000	-291000	-.0132	-329500	25000	-304500	-.0136
-8	-225000		-182000	-.0083	-222000		-197000	-.0090
-4	-109000		-66600	-.0030	-108000		-83000	-.0038
0	0		43000	.00195	0		25000	.00114
4	109600		152600	.0069	108000		133000	.0061
8	225000		268000	.0122	222000		247000	.0111

30	31	32	33
C.G. #7			
$-Z_p \ell_p$	$X_p \zeta_p$	(30)+(31)	C_{M_P}
-355000	62600	-292400	-.0133
-238000		-175400	-.0079
[illegible]		-53900	-.0024
-		62600	.00285
+116550		+179100	+.0082
+338000		+300600	+.0137

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PART II-B-7

PITCHING MOMENT OF PROPELLERS

$T_c = .024 \qquad V\bar{\sigma}^{\frac{1}{2}} = 331.3$ mph

$q = 281$

1	2	3	4	5	6	7	8	9	10	11
				C.G. #1				C.G. #2		
α_{TL}	X_p	Z_p	$\frac{2}{q S_w MAC_w}$	$-Z_p l_p$	$X_p z_p$	5+6	C_{M_P}	$-Z_p l_p$	$X_p z_p$	9+10
-12	-2910	-3300	2.91×10^{-8}	-572000	49400	-522600	-.0152	-619000	49400	-569600
-8		-2100		-366000		-316600	-.0092	-394000		-344600
-4		-1100		-192000		-142600	-.0041	-206000		[illegible]
0		0		0		49400	.0014	0		49400
4		1100		192000		241400	.0070	206000		255400
8		2100		366000		415400	.0121	394000		443400

12	13	14	15	16	17	18	19	20
C.G. #2	C.G. #3				C.G. #4			
C_{M_P}	$-Z_p l_p$	$X_p z_p$	13+14	C_{M_P}	$-Z_p l_p$	$X_p z_p$	17+18	C_{M_P}
-.0166	-572000	17450	-554550	-.0162	-619000	17450	-601550	-.0175
-.0100	-366000		-348550	-.0101	-394000		-376550	-.0110
-.0045	-192000		-174550	-.0051	-206000		-188550	[illegible]
.0014	0		17450	[illegible]	0		17450	.0205
.0074	192000		209450	.0061	206000		223450	.0065
.0129	366000		383450	.0111	394000		411450	.0120

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PITCHING MOMENT OF PROPELLERS

T_c = .024 $V\sigma^{\frac{1}{2}}$ = 331.3 mph

q = 281

21	22	23	24	25	26	27	28	29
	C.G. #5				C.G. #6			
TL	$-Z_p \ell_p$	$X_p \mathfrak{z}_p$	22 + 23	C_{M_p}	$-Z_p \ell_p$	$X_p \mathfrak{z}_p$	26 + 27	C_{M_p}
-12	-591000	34000	-557000	-.0162	-585000	19800	-565200	-.0165
-8	-377000		-343000	-.0100	-372000		-352200	-.0103
-4	-197000		-163000	-.0048	-195000		-175200	-.0051
0	0		34000	+.0010	0		19800	.0006
4	197000		231000	.0067	195000		21480	.0063
8	377000		411000	.0120	372000		391800	.0114

30	31	32	33
	C.G. #7		
$Z_p \ell_p$	$X_p \mathfrak{z}_p$	30 + 31	C_{M_p}
-630000	49500	-580500	-.0169
-401000		-351500	-.0102
-210000		-160500	-.0047
0		+49500	.0014
+210000		+259500	.0075
+401000		+450500	.0131

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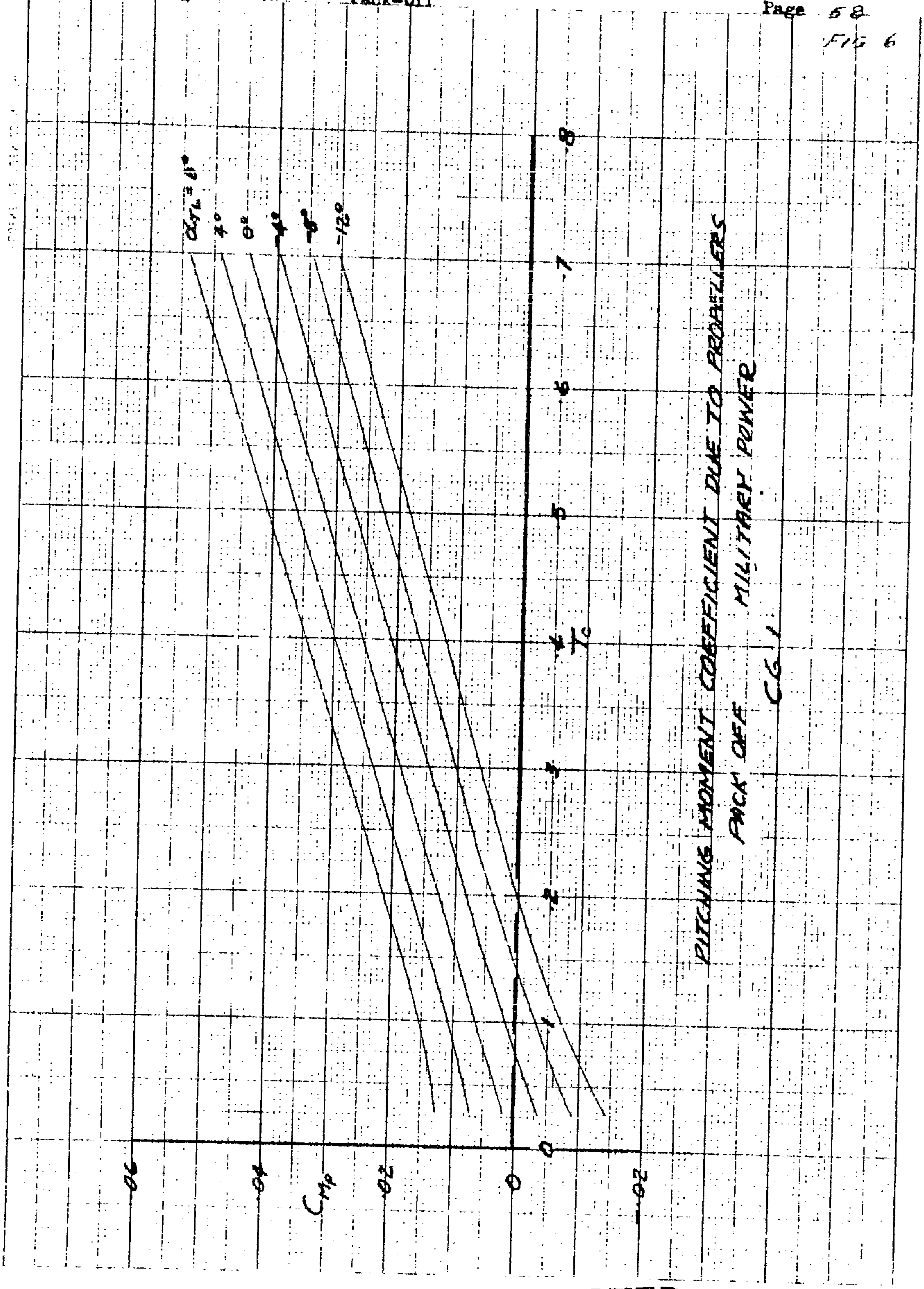
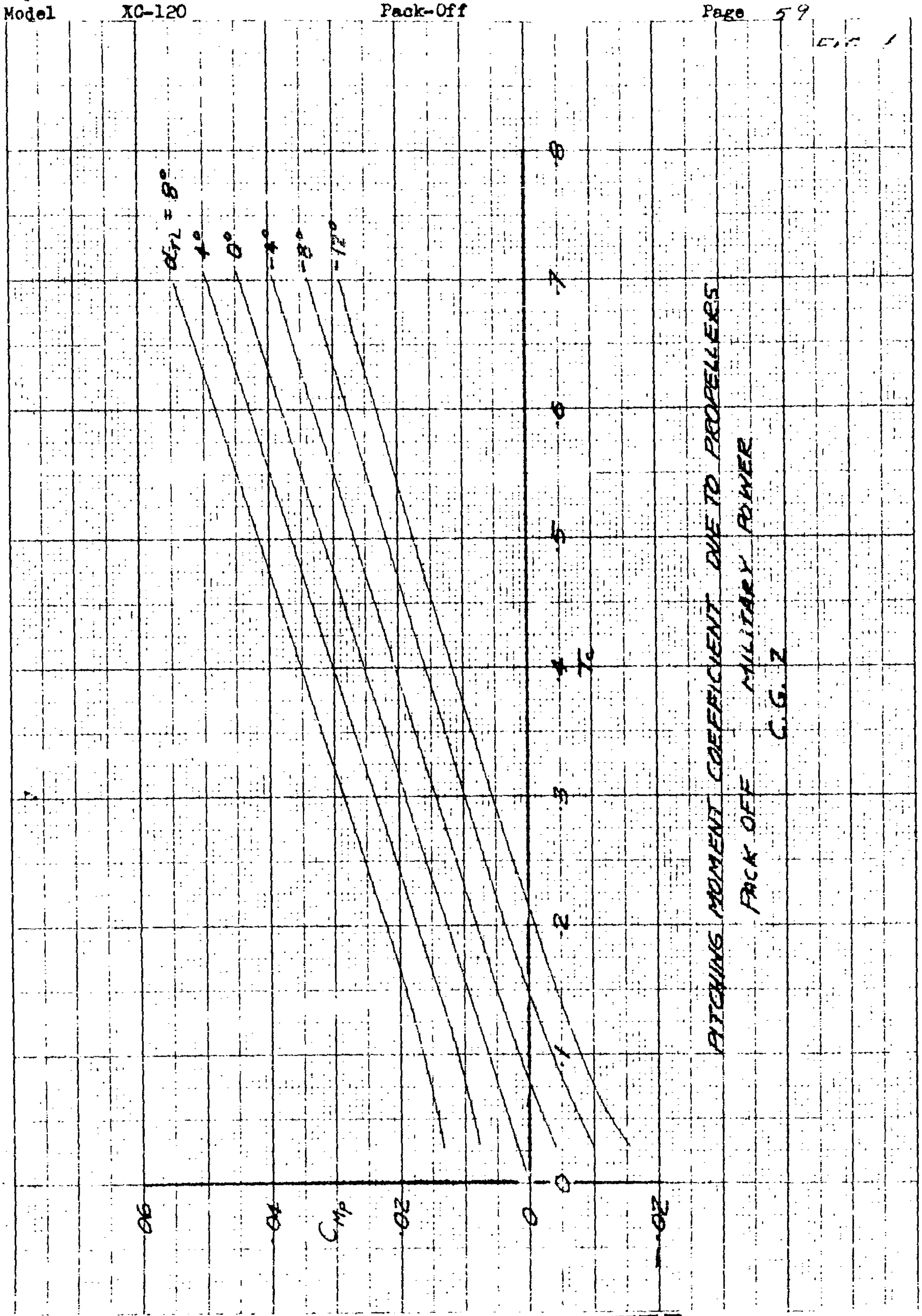
FIG 6


PITCHING MOMENT COEFFICIENT DUE TO PROPELLERS
PACK OFF MILITARY POWER
CG 1
$\alpha_{FL} = 8°$
4°
0°
-4°
-8°
-12°
T_c
C_{M_P}
.06
.04
.02
0
-.02
0
1
2
3
4
5
6
7
8


PITCHING MOMENT COEFFICIENT DUE TO PROPELLERS
PACK OFF MILITARY POWER
C.G. Z
$\alpha_{FL} = 8°$
4°
0°
-4°
-8°
-12°
T_c
C_{M_p}
.06
.04
.02
0
-.02
0
.1
.2
.3
.4
.5
.6
.7
.8

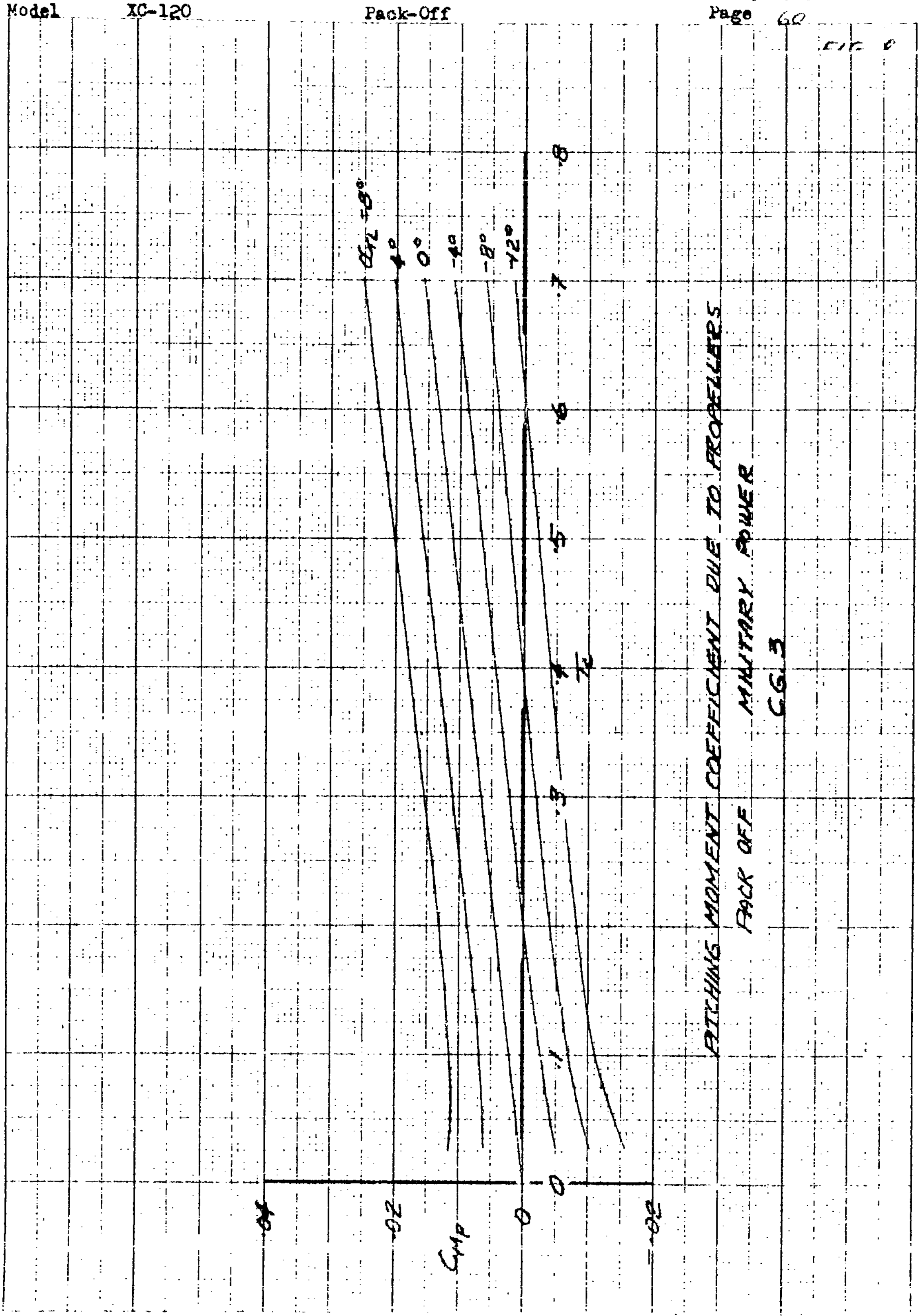

PITCHING MOMENT COEFFICIENT DUE TO PROPELLERS
PACK OFF MILITARY POWER
C.G. 3
αTL 8°
4°
0°
-4°
-8°
-12°
.8
.7
.6
.5
.4
.3
.1
0
Tc
.04
.02
CMp
0
-.02

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FIG. 9

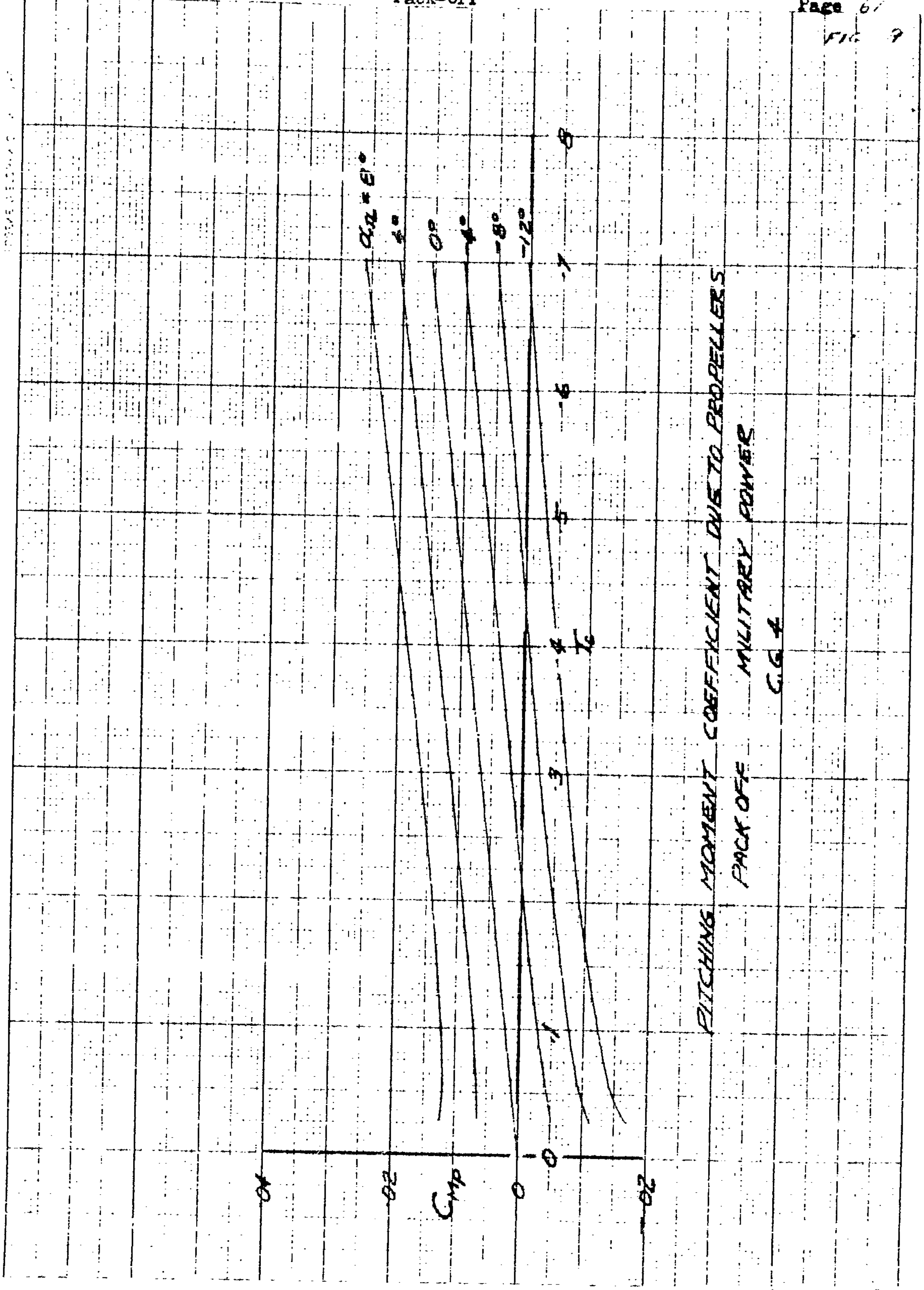

PITCHING MOMENT COEFFICIENT DUE TO PROPELLERS
PACK OFF MILITARY POWER
C.G. 4
$\alpha_R = 8°$
4°
0°
-4°
-8°
-12°
T_c
0
1
3
4
5
6
7
8
.04
.02
C_{mp}
0
-.02

FIG 10

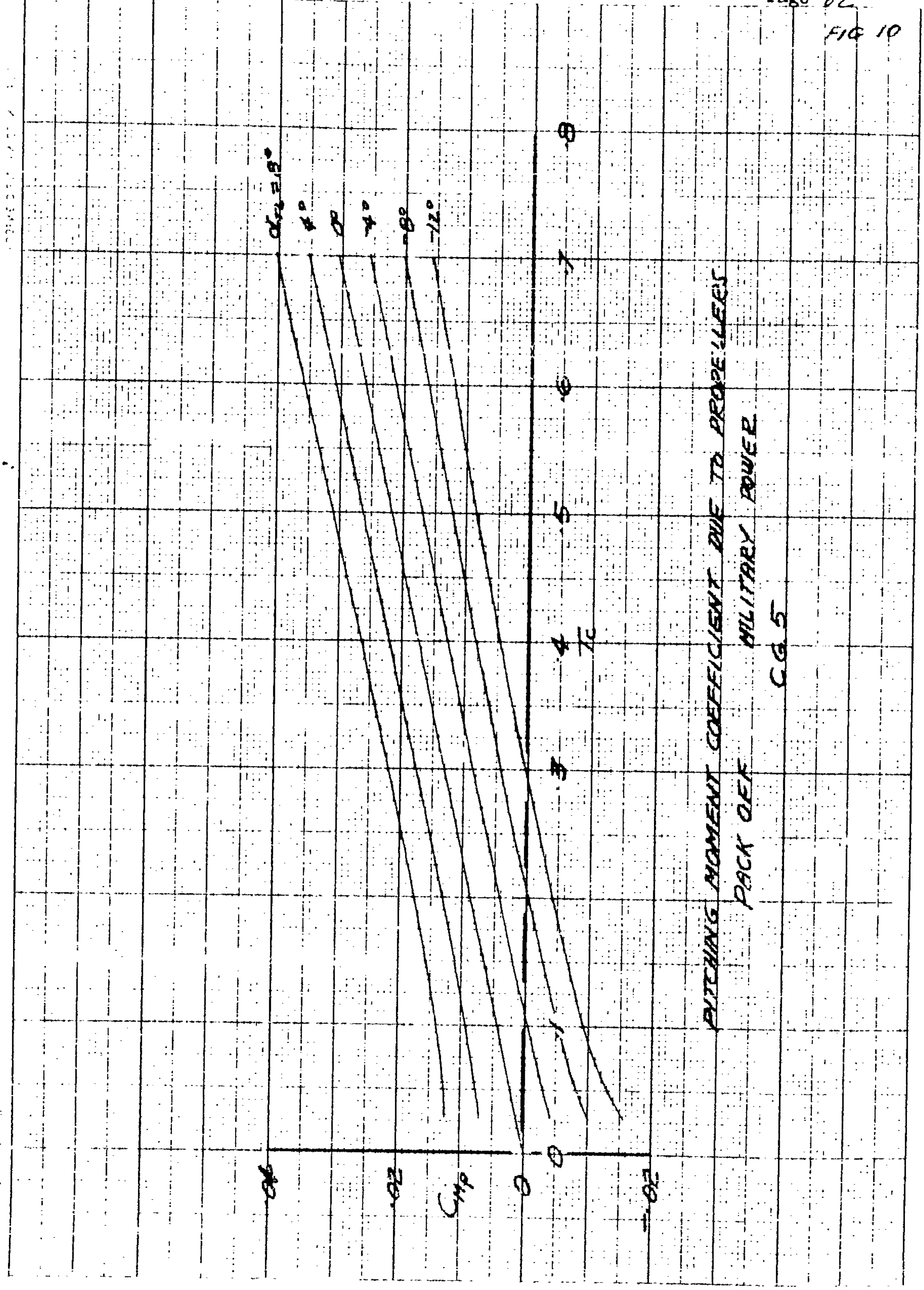
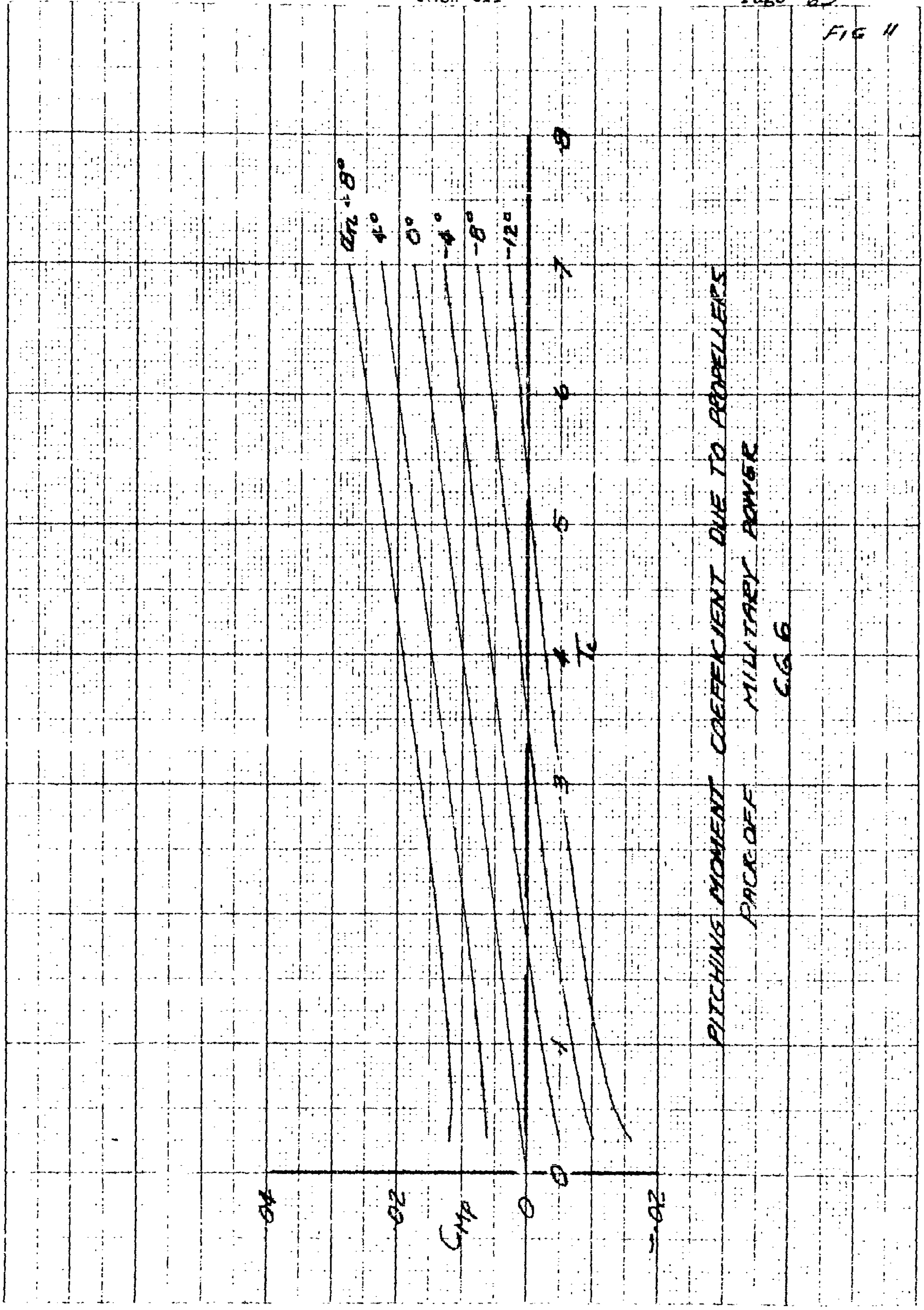

PITCHING MOMENT COEFFICIENT DUE TO PROPELLERS
PACK OFF MILITARY POWER
CG 5
-4°
-8°
-12°
T_c
0
1
3
4
5
6
7
8
C_{MP}
.02
0
-.02

FIG 11


PITCHING MOMENT COEFFICIENT DUE TO PROPELLERS
MILITARY POWER
PACKOFF
C.G. 6
α_{fe} +8°
4°
0°
-4°
-8°
-12°
T_c
C_{M_P}
.04
.02
0
-.02
0
1
2
3
4
5
6
7
8

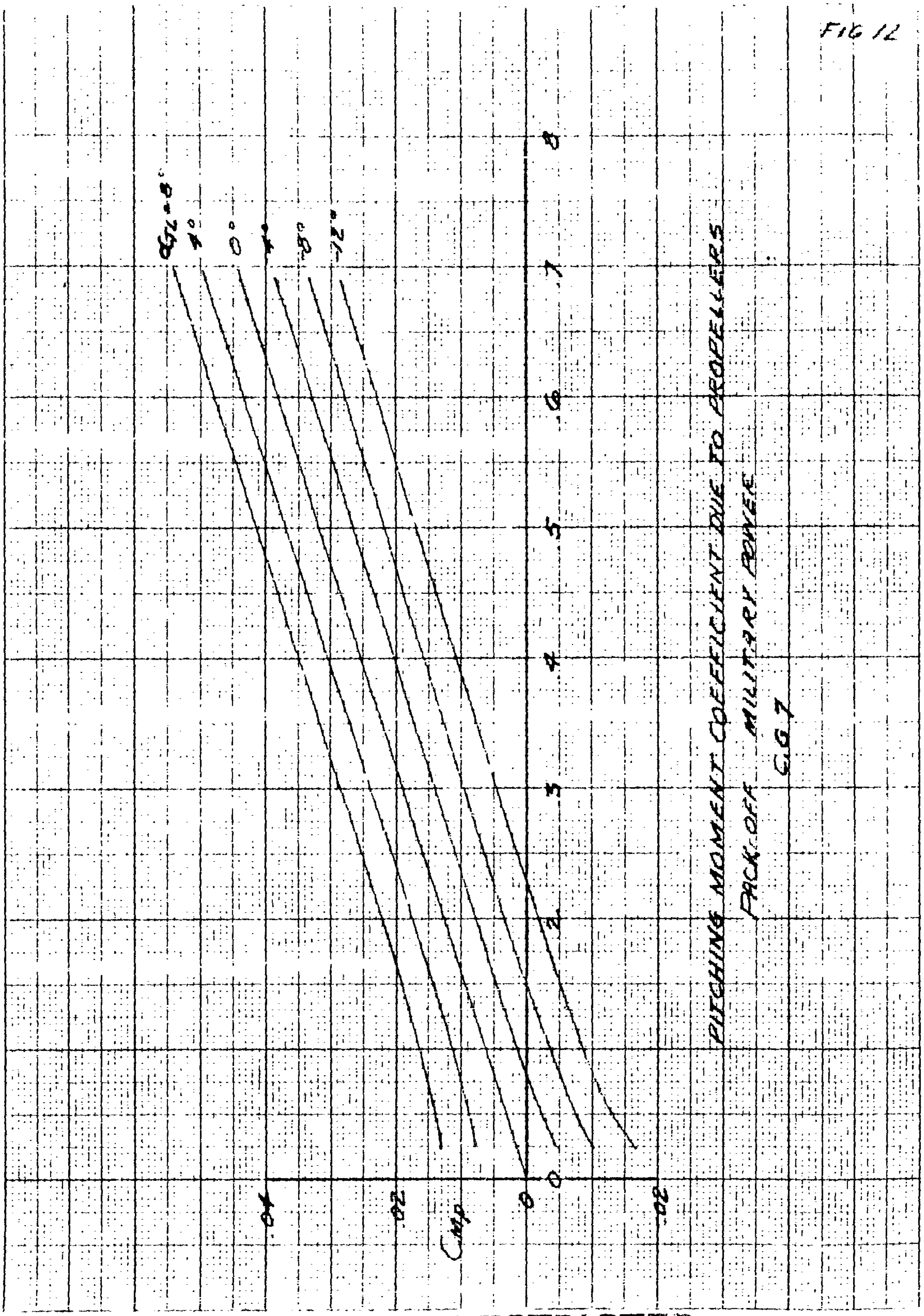


FIG 12
PITCHING MOMENT COEFFICIENT DUE TO PROPELLERS
PACK-OFF MILITARY POWER
C.G. 7
C_{M_P}
.04
.02
0
-.02
0
1
2
3
4
5
6
7
8
$\delta_{GL} = 8°$
4°
0°
-4°
-8°
-12°

PART II-B-8

8a. <u>Slipstream Effects - Wing Unflapped</u>

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

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

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

where

$$A = \left(\frac{x_{wi}}{MAC_w} - \frac{AC_{wi}}{MAC_w} \right)$$

$$B = \frac{z_{wi}}{MAC_w}$$

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

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

C. G. Location	A	B
1	.0148	.0349
2	.0947	.0349
3	.0148	.1006
4	.0947	.1006
5	.0587	.0662
7	.1197	.0349

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PART II-D-6

PITCHING MOMENT DUE TO SLIPSTREAM - 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}	$2\Delta C_{Z_{W1}}$	$2\Delta C_{X_{W1}}$	$A_1 \times (2)$	$B_1 \times (3)$	C_{M_S}	$A_2 \times (2)$	$B_2 \times (3)$	C_{M_S}	$A_3 \times (2)$	$B_3 \times (3)$
-12	-.0052	.0165	-.0001	.0003	.0002	-.0005	.0004	-.0001	-.0001	.0011
-8	.0366	.0067	.0005	.0002	.0007	.0035	.0002	.0037	.0005	.0007
-4	.0736	.0046	.0012	.0002	.0014	.0074	.0002	.0076	.0012	[illegible]
0	.1206	.0058	.0018	.0002	.0020	.0114	.0002	.0116	.0018	.0006
4	.1650	.0090	.0024	.0003	.0027	.0156	.0003	.0159	.0024	.0010
8	.2106	.0153	.0031	.0005	.0036	.0200	.0005	.0205	.0031	.0015

12	13	14	15	16	17	18	19	20	21
C.G. #3	C.G. #4			C.G. #5			C.G. #7		
C_{M_S}	$A_4 \times (2)$	$B_4 \times (3)$	C_{M_S}	$A_5 \times (2)$	$B_5 \times (3)$	C_{M_S}	$A_7 \times (2)$	$B_7 \times (3)$	C_{M_S}
.0010	-.0005	.0011	.0006	-.0003	.0007	.0004	-.0006	.0003	-.0004
.0012	.0031	.0007	.0042	.0021	.0004	.0025	.0044	.0002	.0046
.0017	.0074	.0005	.0079	.0046	.0003	.0049	.0094	.0002	.0096
.0027	.0114	.0006	.0120	.0070	.0004	.0074	.0144	.0002	.0146
.0034	.0156	.0010	.0166	.0097	.0006	.0103	.0197	.0003	.0200
.0046	.0200	.0015	.0215	.0123	.0010	.0133	.0251	.0005	.0256

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PART II-B-8

PITCHING MOMENTS DUE TO SLIPSTREAM - WING UNFLAPPED

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

1	2	3	4	5	6	7	8	9	10	11
			C.G. #1			C.G. #2			C.G. #3	
α_{TL}	$2\Delta C_{Z_{WL}}$	$2\Delta C_{X_{WL}}$	$A_1 \times ②$	$B_1 \times ③$	C_{M_S}	$A_2 \times ②$	$B_2 \times ③$	C_{M_S}	$A_3 \times ②$	$B_3 \times ③$
-12	-.0094	.0071	-.0001	.0001	0	-.0009	.0001	-.0008	-.0001	.0004
-8	.0094	.0019	.0001	.0001	.0002	.0009	.0001	.0010	.0001	.0002
-4	.0284	.0009	.0004	0	.0004	.0027	0	.0027	.0004	.0001
0	.0476	.0019	.0007	0	.0007	.0045	0	.0045	.0007	.0002
4	.0674	.0046	.0010	.0002	.0012	.0064	.0002	.0066	.0010	.0005
8	.0876	.0072	.0013	.0003	.0016	.0083	.0003	.0086	.0013	.0009

12	13	14	15	16	17	18	19	20	21
C.G. #3	C.G. #4			C.G. #5			C.G. #7		
C_{M_S}	$A_4 \times ②$	$B_4 \times ③$	C_{M_S}	$A_5 \times ②$	$B_5 \times ③$	C_{M_S}	$A_7 \times ②$	$B_7 \times ③$	C_{M_S}
.0003	-.0009	.0004	-.0005	-.0006	.0003	-.0003	-.0011	.0001	-.0010
.0003	.0009	.0002	.0011	.0006	.0001	.0007	.0011	.0001	.0012
.0005	.0027	.0001	.0028	.0017	.0001	.0018	.0034	0	.0034
.0007	.0045	.0002	.0047	.0028	.0001	.0029	.0057	0	.0057
.0015	.0064	.0005	.0069	.0040	.0003	.0043	.0081	.0002	.0083
.0022	.0083	.0009	.0089	.0052	.0006	.0058	.0104	.0003	.0107

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PART II-B-7

PITCHING MOMENT DUE TO SLIPSTREAM - WING UNFLAPPED

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

1	2	3	4	5	6	7	8	9	10	11
			C.G. #1			C.G. #2			C.G. #3	
α_w	$2\Delta C_{Z_{WI}}$	$2\Delta C_{X_{WI}}$	$A_1 \times$ (2)	$B_1 \times$ (3)	C_{M_S}	$A_2 \times$ (2)	$B_2 \times$ (3)	C_{M_S}	$A_3 \times$ (2)	$B_3 \times$ (3)
-12	-.0074	.0034	-.0001	.0001	0	-.0007	.0001	-.0006	-.0001	.0003
-8	.0060	.0012	.0001	0	.0001	.0006	0	.0006	.0001	.0001
-4	.0200	.0004	.0003	0	.0003	.0019	0	.0019	.0003	0
0	.0330	.0013	.0005	0	.0005	.0031	0	.0031	.0005	.0001
4	.0476	.0037	.0007	.0001	.0008	.0045	.0001	.0046	.0007	.0004
8	.0624	.0064	.0009	.0003	.0012	.0059	.0003	.006[illegible]	.0009	.0008

12	13	14	15	16	17	18	19	20	21
C.G. #3	C.G. #4			C.G. #5			C.G. #7		
C_{M_S}	$A_4 \times$ (2)	$B_4 \times$ (3)	C_{M_S}	$A_5 \times$ (2)	$B_5 \times$ (3)	C_{M_S}	$A_7 \times$ (2)	$B_7 \times$ (3)	C_{M_S}
.0002	-.0007	.0003	-.0004	-.0004	-.0002	-.0002	-.0009	.0001	-.0008
.0002	.0006	.0001	.0007	.0004	.0001	.0005	.0007	0	.0007
.0003	.0019	0	.0018	.0012	0	.0012	.0024	0	.0024
.0006	.0031	.0001	.0032	.0020	.0001	.0021	.0039	0	.0039
.0011	.0045	.0004	.0049	.0028	.0003	.0031	.0057	.0001	.0058
.0017	.0059	.0008	.0067	.0037	.0006	.0043	.0075	.0003	.0078

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PART II-B-7

PITCHING MOMENT DUE TO SLIPSTREAM - WING UNFLAPPED

T_c = .047 $V\sqrt{\sigma}$ = 265 mph

1	2	3	4	5	6	7	8	9	10	11
α_{TL}	$20C_{Z_{W1}}$	$20C_{X_{W1}}$	$A_1 \times$ (2)	$B_1 \times$ (3)	C_{M_S}	$A_2 \times$ (2)	$B_2 \times$ (3)	C_{M_S}	$A_3 \times$ (2)	$B_2 \times 3$
-12	-.004	+.0024	-.0001	.0001	0	-.0004	.0001	-.0003	-.0001	+.0002
-8	+.001	+.0004	0	0	0	+.0001	0	+.0001	0	0
-4	+.006	-.0002	+.0001	0	+.0001	+.0006	0	+.0006	+.0001	0
0	+.012	+.0006	+.0002	0	+.0002	+.0011	0	+.0011	+.0002	0
4	+.018	+.0030	+.0003	+.0001	+.0004	+.0017	+.0001	+.0017	+.0003	+.0003
8	+.025	+.0074	+.0004	+.0003	+.0007	+.0024	+.0003	+.0024	+.0004	+.0007

12	13	14	15	16	17	18	19	20	21
C.G. 3	CG 4			CG. 5			C.G. 7		
C_{M_S}	$A_4 \times$ (2)	$B_4 \times$ (3)	C_{M_S}	$A_5 \times$ (2)	$B_5 \times$ (3)	C_{M_S}	$A_7 \times$ (2)	$B_7 \times$ (3)	C_{M_S}
+.0001	-.0004	+.0002	-.0002	-.0002	+.0002	0	-.0005	.0001	-.0004
0	+.0001	0	+.0001	+.0001	0	+.0001	.0001	0	.0001
+.0001	+.0006	0	+.0006	+.0004	0	+.0004	.0007	0	.0007
+.0002	+.0011	0	+.0011	+.0006	0	+.0006	.0014	0	.0014
+.0006	+.0017	+.0003	+.0020	+.0011	+.0002	+.0013	.0021	.0001	.0022
+.0011	+.0024	+.0007	+.0031	+.0015	+.0005	+.0020	.0030	.0003	.0033

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PITCHING MOMENT DUE TO SLIPSTREAM – WING UNFLAPPED

T_c = .024 $V\sqrt{\sigma}$ = 331.3 mph

1	2	3	4	5	6	7	8	9	10	11
			C.G. #1			C.G. #2			CG #3	
α_{TL}	$2\Delta C_{Z_{W1}}$	$2\Delta C_{X_{W1}}$	$A_1 \times$ (2)	$B_1 \times$ (3)	C_{M_S}	$A_2 \times$ (2)	$B_2 \times$ (3)	C_{M_S}	$A_3 \times$ (2)	$B_3 \times$ (3)
-12	-.002	+.0024	0	+.0001	+.0001	-.0002	+.0001	-.0001	0	+.0002
-8	+.0005	+.0004	0	0	0	0	0	0	0	0
-4	+.004	-.0002	+.0001	0	+.0001	+.0004	0	+.0004	+.0001	0
0	+.006	+.0006	+.0001	0	+.0001	+.0006	0	+.0006	+.0001	0
4	+.010	+.0030	+.0001	+.0001	+.0002	+.0009	+.0001	+.0009	+.0001	+.0003
8	+.014	+.0074	+.0002	+.0003	+.0005	+.0013	+.0003	+.0016	+.0002	+.0007

12	13	14	15	16	17	18	19	20	21
C.G. #3	C.G. #4			CG. #5			CG. #7		
C_{M_S}	$A_4 \times$ (2)	$B_4 \times$ (3)	C_{M_S}	$A_5 \times$ (2)	$B_5 \times$ (3)	C_{M_S}	$A_7 \times$ (2)	$B_7 \times$ (3)	C_{M_S}
+.0002	-.0002	+.0002	+.0001	-.0001	+.0002	+.0001	-.0002	+.0001	-.0001
0	0	0	0	0	0	0	.0001	0	+.0001
+.0001	+.0004	0	+.0004	+.0002	0	+.0002	.0005	0	+.0005
+.0001	+.0006	0	+.0006	+.0004	0	+.0004	.0007	0	+.0007
+.0004	+.0009	+.0003	+.0012	+.0006	+.0002	+.0008	.0012	+.0001	+.0013
+.0009	+.0013	+.0007	+.0020	+.0006	+.0005	+.0013	.0017	+.0003	+.0020

RESTRICTED

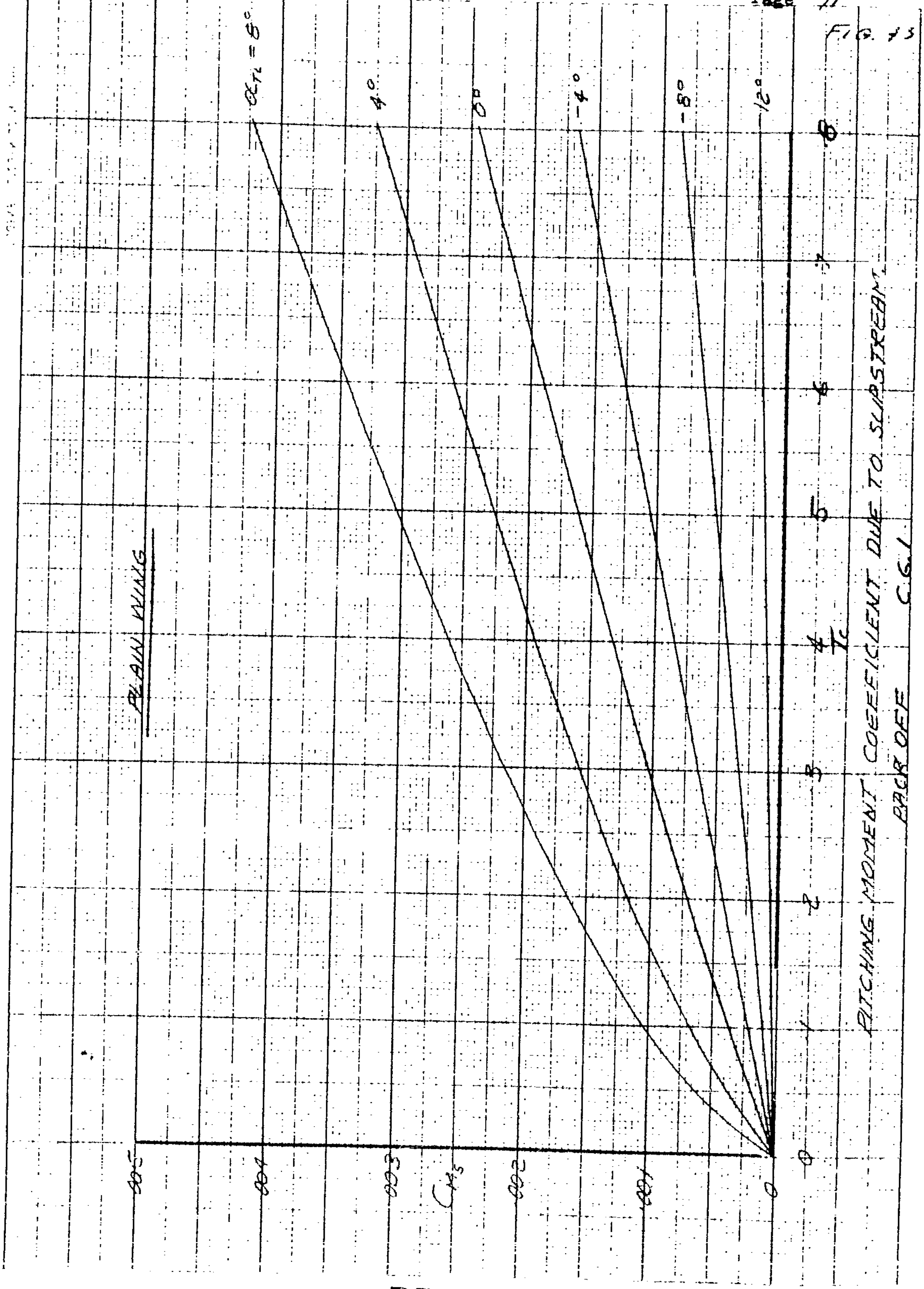

PLAIN WING
$\alpha_{TL} = 8°$
4°
0°
-4°
-8°
-12°
.005
.004
.003
C_{M_S}
.002
.001
0
0
1
2
3
4
T_c
5
6
7
8
PITCHING MOMENT COEFFICIENT DUE TO SLIPSTREAM
PACK OFF C.G.1

FIG. 43

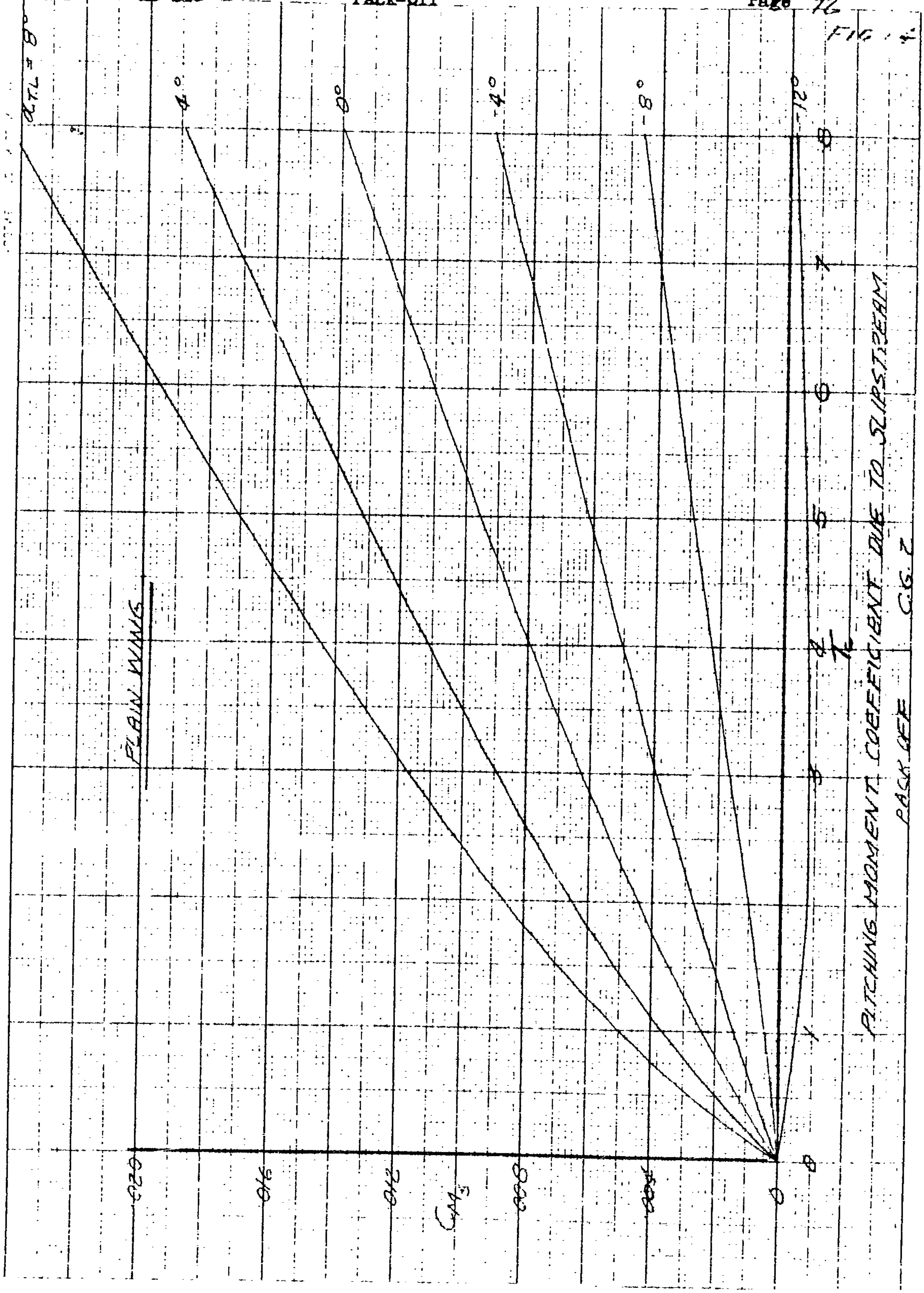


FIG. 4
PITCHING MOMENT COEFFICIENT DUE TO SLIPSTREAM
PACK OFF C.G. 2
PLAIN WING
$\alpha_{T.L.}$ = 8°
4°
0°
-4°
-8°
-12°
C_{M_S}
.020
.016
.012
.008
.004
0
1
2
3
4
5
6
7
8
T_c

FIG 15

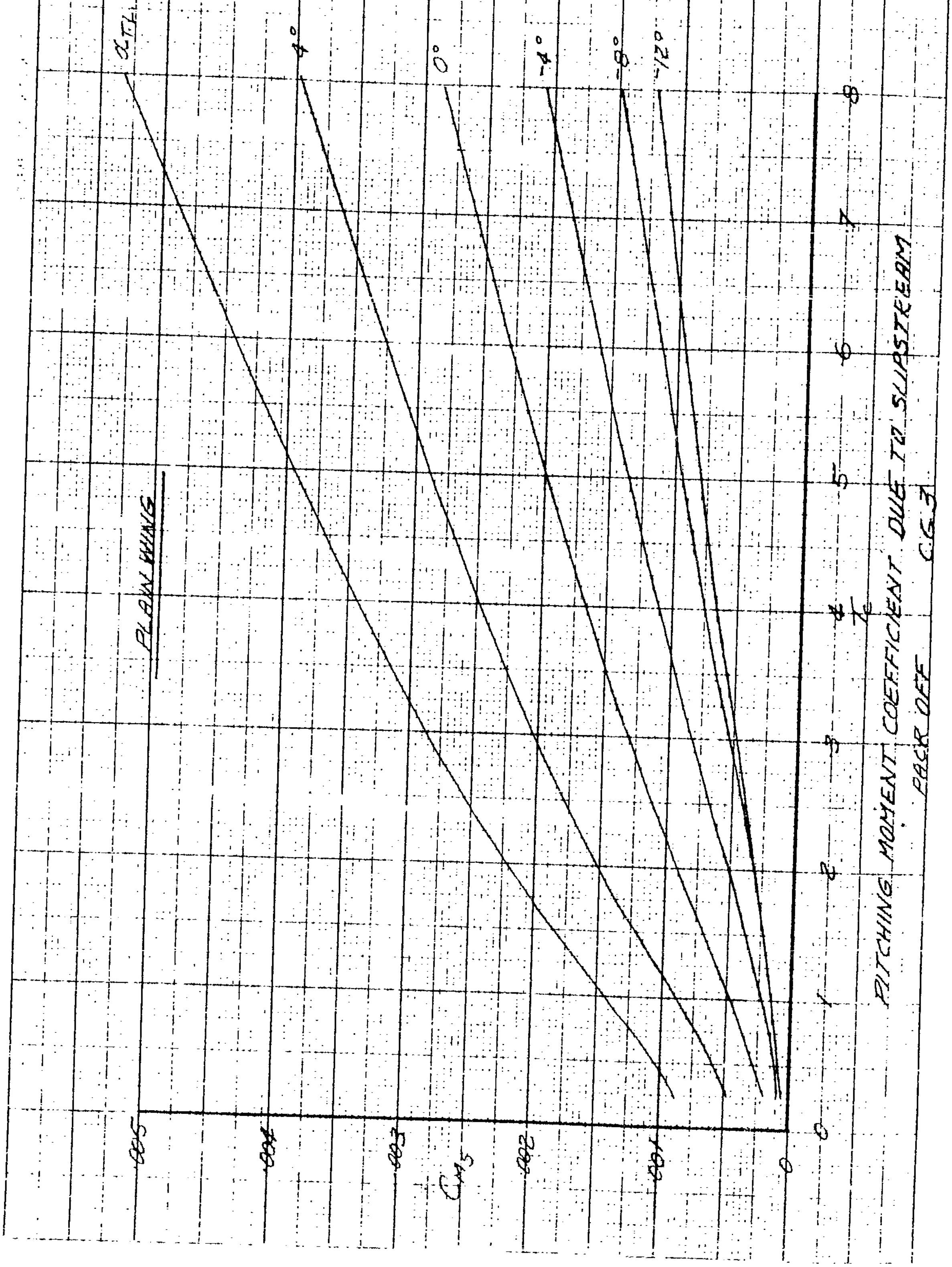

$\alpha_{T.L.}$ = 8°
4°
0°
-4°
-8°
-12°
PLAIN WING
0
1
2
3
4
5
6
7
8
T_c
0
.001
.002
.003
.004
.005
C_{m_s}
PITCHING MOMENT COEFFICIENT DUE TO SLIPSTREAM
PACK OFF
C.G.3

FIG 16

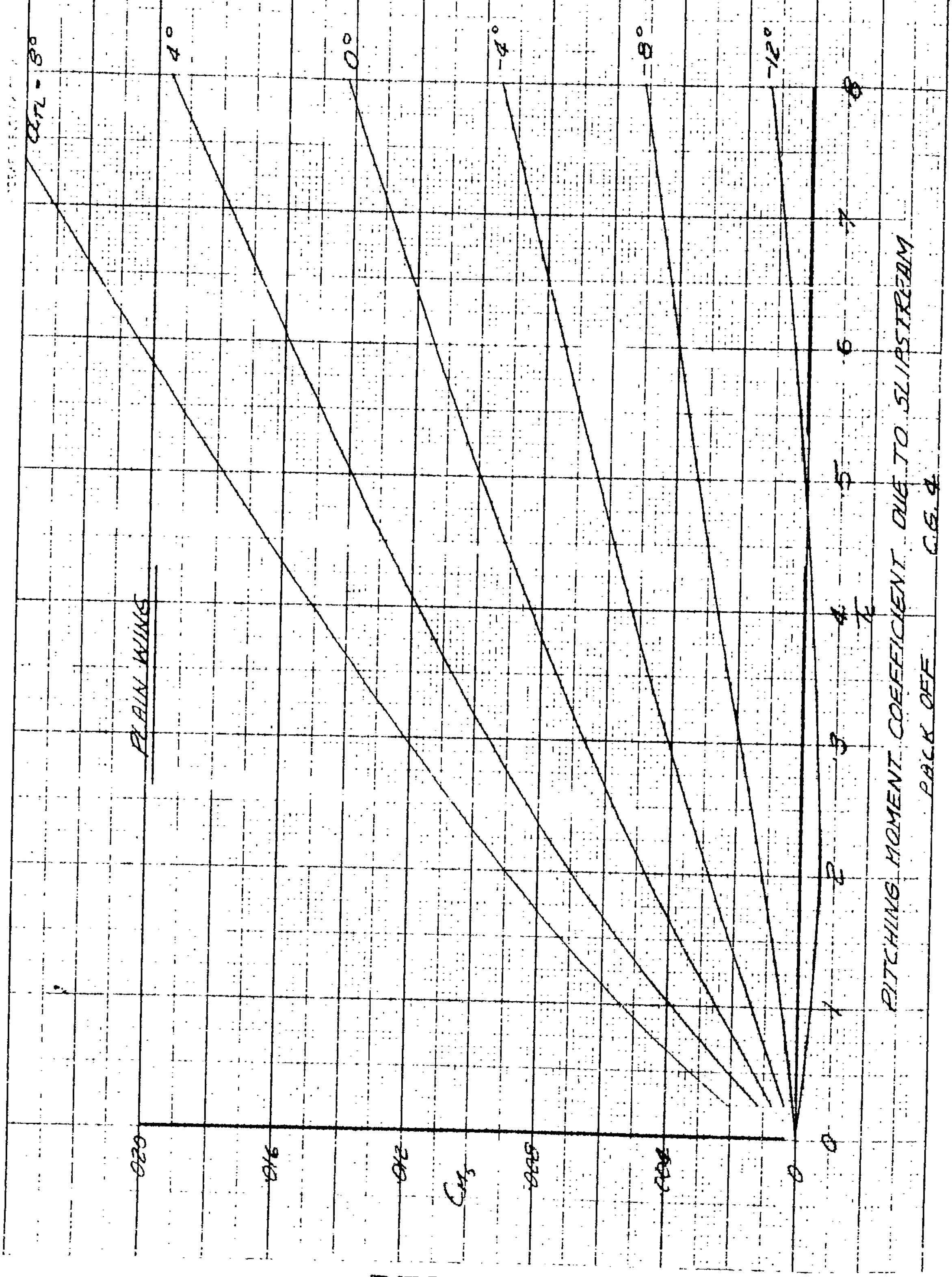
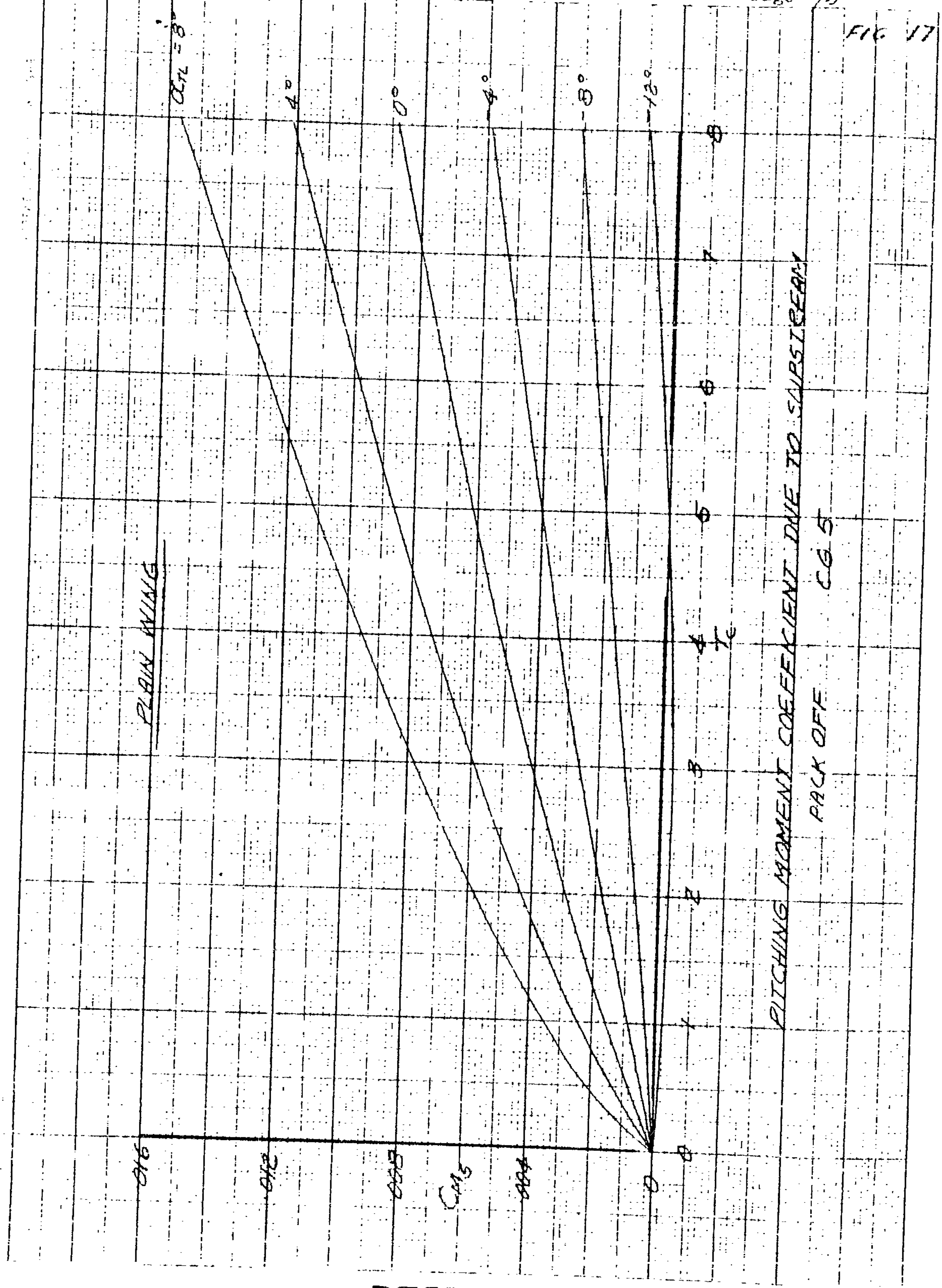

PITCHING MOMENT COEFFICIENT DUE TO SLIPSTREAM
PACK OFF
PLAIN WING
.020
.016
.012
.008
.004
0
0
1
2
3
4
5
6
7
8
-4°
-8°
-12°

FIG 17


α_{FL} = -8°
-4°
0°
4°
8°
12°
0
1
2
3
4
5
6
7
8
T_c
.016
.012
.008
C_{M_S}
.004
0
PLAIN WING
PITCHING MOMENT COEFFICIENT DUE TO SLIPSTREAM
PACK OFF
CG 5

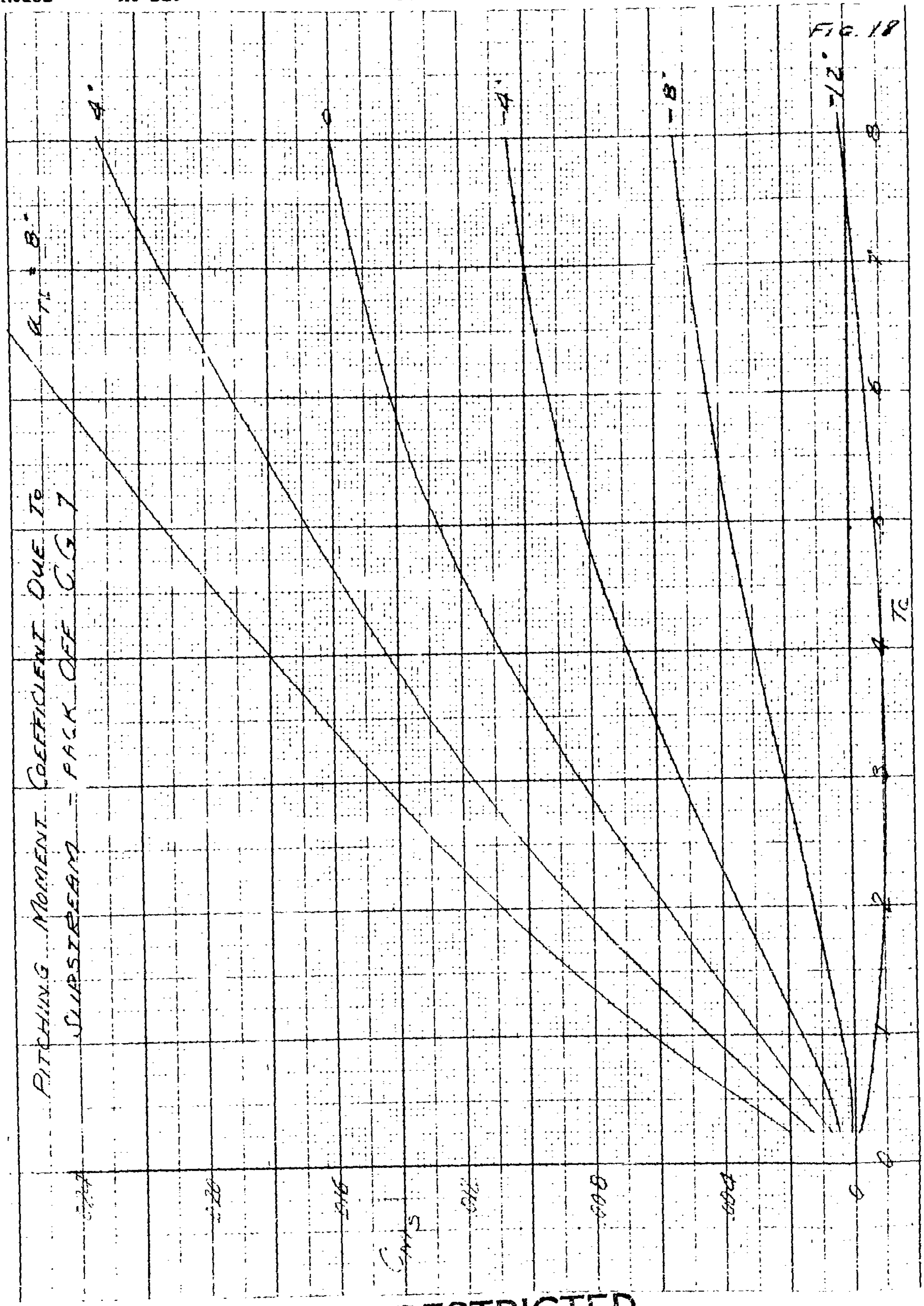


FIG. 18
PITCHING MOMENT COEFFICIENT DUE TO SLIPSTREAM - PACK OFF C.G. 7
α_{TL} = 8°
4°
0
-4°
-8°
-12°
Tc
0
2
4
6
8
C_{m_S}
0
-.004
-.008
-.012
-.016
-.020
-.024

RESTRICTED

PART II-B-8

8b. Slipstream Effects Flapped Wing

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

where $A = \left(\frac{x_{Wi}}{MAC_W} - \frac{AC_{Wi}}{MAC_W}\right)$

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

$\Delta C_{M_{WFI}}$ from figures 52 and 123, reference (1)

$\Delta C_{X_{WFI}}$ from figures 55 and 131, reference (1)

$\Delta C_{Z_{WFI}}$ from figures 56 and 133, reference (1)

C. G. Location	A	B
1	.0148	.0349
2	.0947	.0349
3	.0148	.1006
4	.0947	.1006
6	.0405	.0662
7	.1147	.1006

Subject:— BASIC FLIGHT CRITERIA - PACK OFF

PART II-B-8

PITCHING MOMENT DUE TO SLIPSTREAM - FLAPPED WING

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}	$2\Delta C_{m_{WFI}}$	$2\Delta C_{Z_{WFI}}$	$2\Delta C_{X_{WFI}}$	$A_1 \times ③$	$B_1 \times ④$	C_{m_s}	$A_2 \times ③$	$B_2 \times ④$	C_{m_s}	$A_3 \times ③$
-12	-.1650	.1400	.0208	.0021	.0007	-.1622	.0132	.0007	-.1511	.0021
-8	-.1716	.1804	.0208	.0027	.0007	-.1682	.0171	.0007	-.1538	.0027
-4	-.1770	.2636	.0194	.0039	.0007	-.1724	.0249	.0007	-.1514	.0039
0	-.1800	.3480	.0160	.0052	.0006	-.1742	.0329	.0006	-.1465	.0052
4	-.1816	.4300	.0122	.0064	.0004	-.1748	.0407	.0004	-.1405	.0064
8	-.1822	.5150	.0096	.0076	.0003	-.1743	.0488	.0003	-.1321	.0076

12	13	14	15	16	17	18	19	20	21	22
C.G. #3		C.G. #4			C.G. #6			C.G. #7		
$B_3 \times ④$	C_{m_s}	$A_4 \times ③$	$B_4 \times ④$	C_{m_s}	$A_6 \times ③$	$B_6 \times ④$	C_{m_s}	$A_7 \times ③$	$B_7 \times ④$	C_{m_s}
.0021	-.1608	.0132	.0021	-.1492	.0057	.0020	-.1573	.0161	.0021	-.1468
.0021	-.1668	.0171	.0021	-.1524	.0073	.0020	-.1623	.0208	.0021	-.1487
.0020	-.1711	.0249	.0026	-.1501	.0107	.0019	-.1644	.0302	.0021	-.1441
.0016	-.1732	.0329	.0016	-.1455	.0141	.0015	-.1644	.0399	.0016	-.1385
.0012	-.1740	.0407	.0012	-.1397	.0174	.0012	-.1630	.0493	.0012	-.1311
.0010	-.1736	.0488	.0010	-.1324	.0209	.0009	-.1604	.0592	.0010	-.1220

Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-B-8

PITCHING MOMENT DUE TO SLIPSTREAM - FLAPPED WING

T_c = .400

1	2	3	4	5	6	7	8	9	10	11
				C.G. #1			C.G. #2			C.G. #3
α_{TL}	$2\Delta C_{M_{WF1}}$	$2\Delta C_{Z_{WF1}}$	$2\Delta C_{X_{WF1}}$	$A_1 \times 3$	$B_1 \times 4$	C_{M_S}	$A_2 \times 3$	$B_2 \times 4$	C_{M_S}	$A_3 \times 3$
-12	-.1062	.0912	.0124	.0014	.0004	-.1044	.0086	.0004	-.0972	.0014
-8	-.1106	.1220	.0124	.0018	.0004	-.1084	.0116	.0004	-.0966	.0018
-4	-.1140	.1816	.0116	.0027	.0004	-.1109	.0172	.0004	-.0961	.0027
0	-.1162	.2410	.0094	.0036	.0003	-.1123	.0228	.0003	-.0931	.0036
4	-.1170	.3010	.0072	.0045	.0003	-.1122	.0285	.0003	-.0882	.0045
8	-.1186	.3622	.0060	.0054	.0002	-.1130	.0343	.0002	-.0871	.0054

12	13	14	15	16	17	18	19	20	21	22
C.G. #3		C.G. #4			C.G. #6			C.G. #7		
$B_3 \times 4$	C_{M_S}	$A_4 \times 3$	$B_4 \times 4$	C_{M_S}	$A_6 \times 3$	$B_6 \times 4$	C_{M_S}	$A_7 \times 3$	$B_7 \times 4$	C_{M_S}
.0012	-.1036	.0086	.0012	-.0964	.0037	.0012	-.1013	.0105	.0012	-.0845
.0012	-.1076	.0116	.0012	-.0978	.0049	.0012	-.1045	.0140	.0012	-.0857
.0012	-.1101	.0172	.0012	-.0956	.0074	.0011	-.1056	.0208	.0012	-.0820
.0009	-.1117	.0228	.0009	-.0925	.0098	.0009	-.1055	.0277	.0009	-.0876
.0007	-.1118	.0285	.0007	-.0878	.0122	.0007	-.1041	.0345	.0007	-.0818
.0006	-.1126	.0343	.0006	-.0837	.0147	.0006	-.1033	.0415	.0006	-.0765

Subject:-- BASIC FLIGHT CRITERIA - PACK OFF

PART II-B-8

PITCHING MOMENT DUE TO SLIPSTREAM - FLAPPED WING

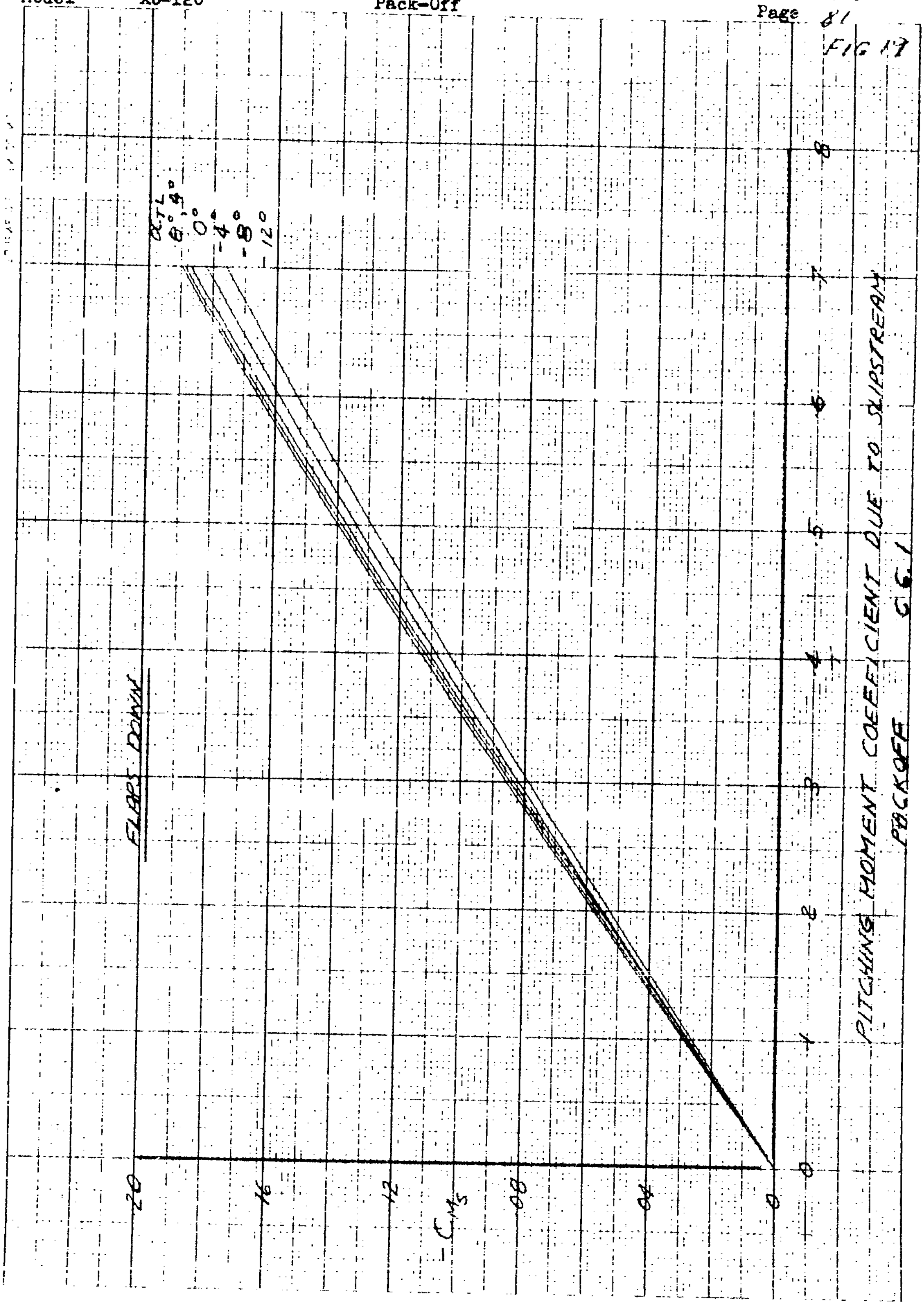
T_c = .198 $V\sqrt{\sigma}$ = 160 mph

1	2	3	4	5	6	7	8	9	10	11
				C.G. #1			C.G. #2			C.G. #3
$\alpha_{T/L}$	$2\Delta C_{M_{WF1}}$	$2\Delta C_{Z_{WF1}}$	$2\Delta C_{X_{WF1}}$	$A_1 \times 3$	$B_1 \times 4$	C_{M_S}	$A_2 \times 3$	$B_2 \times 4$	C_{M_S}	$A_3 \times 3$
-12	-.0532	.0470	.0052	.0007	.0002	-.0523	.0045	.0002	-.0485	.0007
-8	-.0554	.0610	.0052	.0010	.0002	-.0542	.0062	.0002	-.0490	.0010
-4	-.0570	.0990	.0048	.0015	.0002	-.0553	.0094	.0002	-.0474	.0015
0	-.0580	.1350	.0044	.0020	.0002	-.0564	.0128	.0002	-.0452	.0020
4	-.0590	.1700	.0040	.0025	.0001	-.0564	.0161	.0001	-.0426	.0025
8	-.0600	.2030	.0054	.0030	.0002	-.0568	.0192	.0002	-.0406	.0030

12	13	14	15	16	17	18	19	20	21	22
C.G. #3		C.G. #4			C.G. #6			C.G. #7		
$B_3 \times 4$	C_{M_S}	$A_4 \times 3$	$B_4 \times 4$	C_{M_S}	$A_6 \times 3$	$B_6 \times 4$	C_{M_S}	$A_7 \times 3$	$B_7 \times 4$	C_{M_S}
.0005	-.0520	.0045	.0005	-.0482	.0019	.0005	-.0508	.0054	.0005	-.0473
.0005	-.0539	.0062	.0005	-.0487	.0026	.0005	-.0523	.0075	.0005	-.0474
.0005	-.0550	.0094	.0005	-.0471	.0040	.0005	-.0525	.0114	.0005	-.0451
.0004	-.0562	.0128	.0004	-.0454	.0055	.0004	-.0527	.0155	.0004	-.0427
.0004	-.0561	.0161	.0004	-.0425	.0069	.0004	-.0517	.0195	.0004	-.0391
.0005	-.0565	.0192	.0005	-.0403	.0082	.0005	-.0513	.0233	.0005	-.0362

FA-850 2A

FIG 17



RESTRICTED

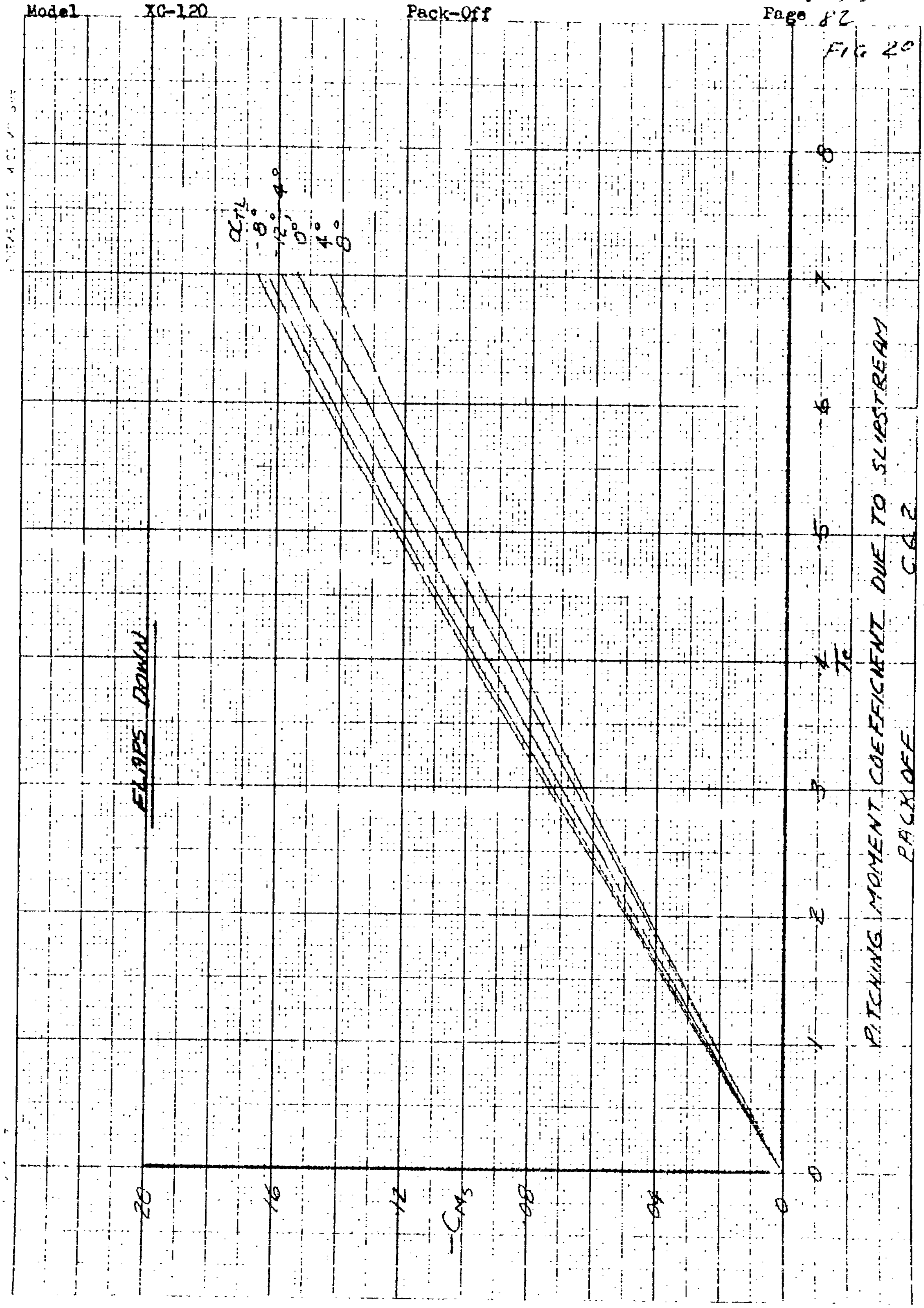
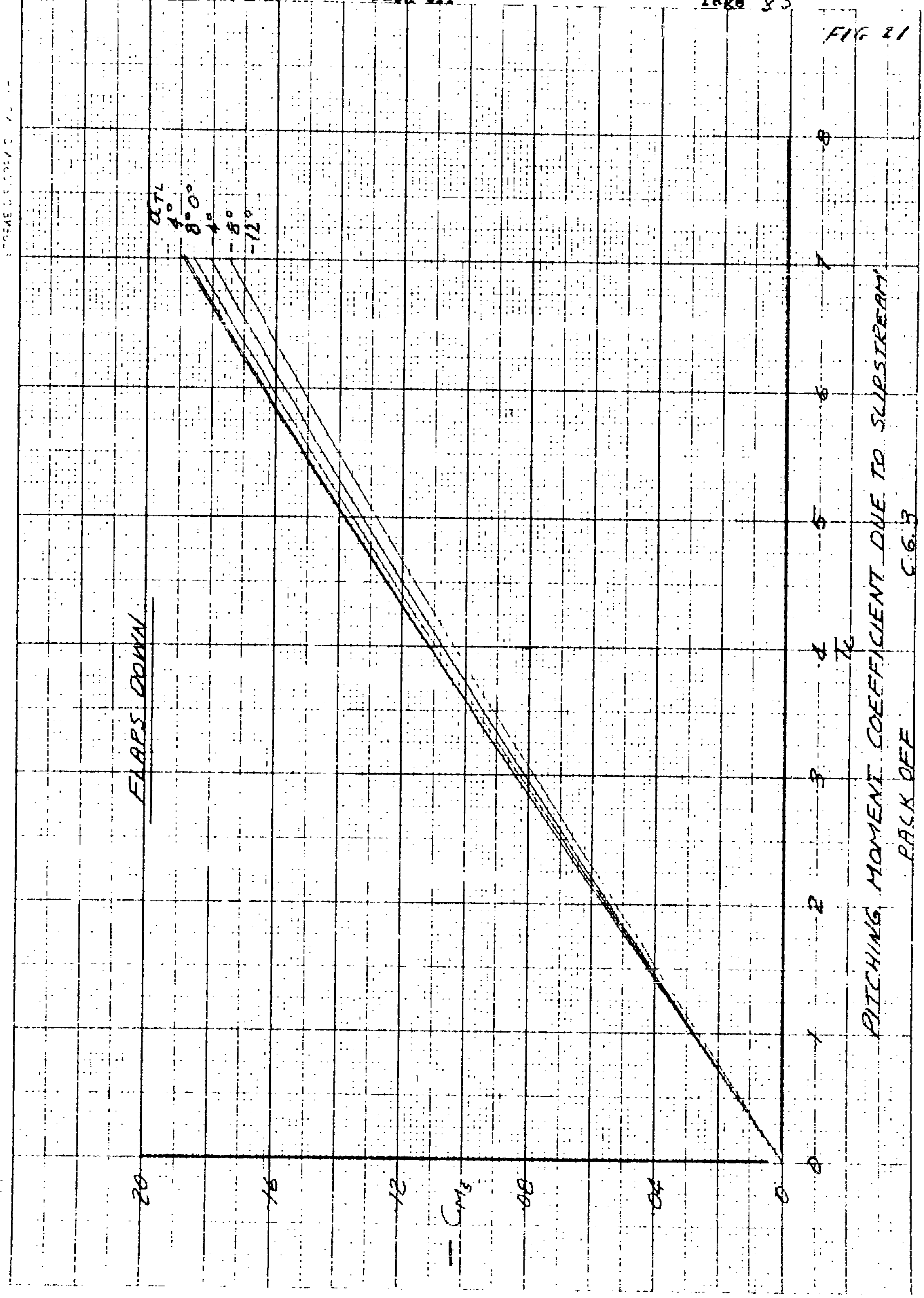


FIG 20
FLAPS DOWN
PITCHING MOMENT COEFFICIENT DUE TO SLIPSTREAM
PACKOFF
CG 2
α_{TL}
T_c
0
1
2
3
4
5
6
7
8
$-C_{M_S}$
0
.04
.08
.12
.16
.20

FIG 21


FLAPS DOWN
α_{TL}
4°
8° 0°
-4°
-8°
-12°
PITCHING MOMENT COEFFICIENT DUE TO SLIPSTREAM
PACK OFF
C.G. 3
T_c
0
1
2
3
4
5
6
7
8
$-C_{m_S}$
20
16
12
08
04
0

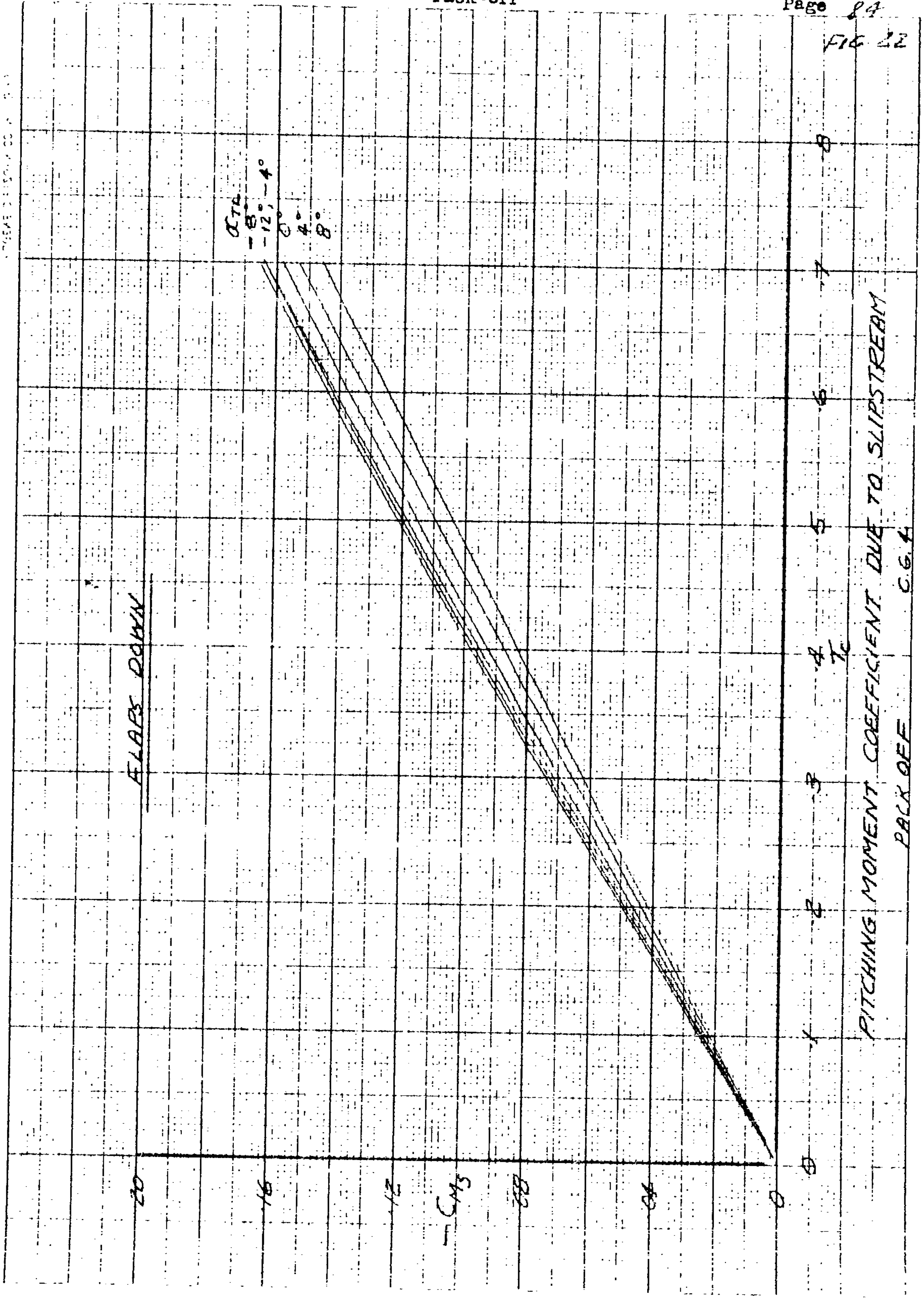


FIG. 22

RESTRICTED

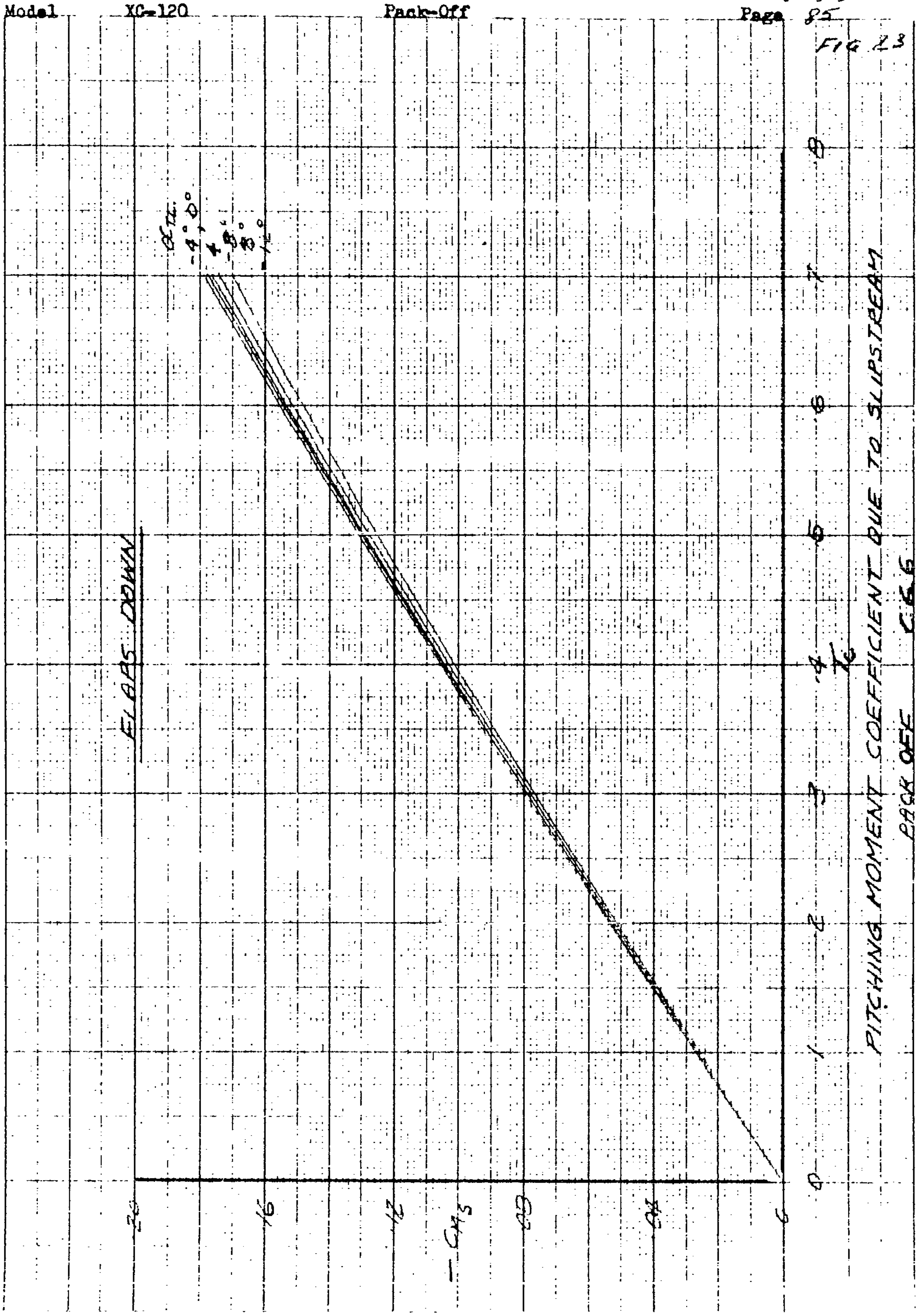
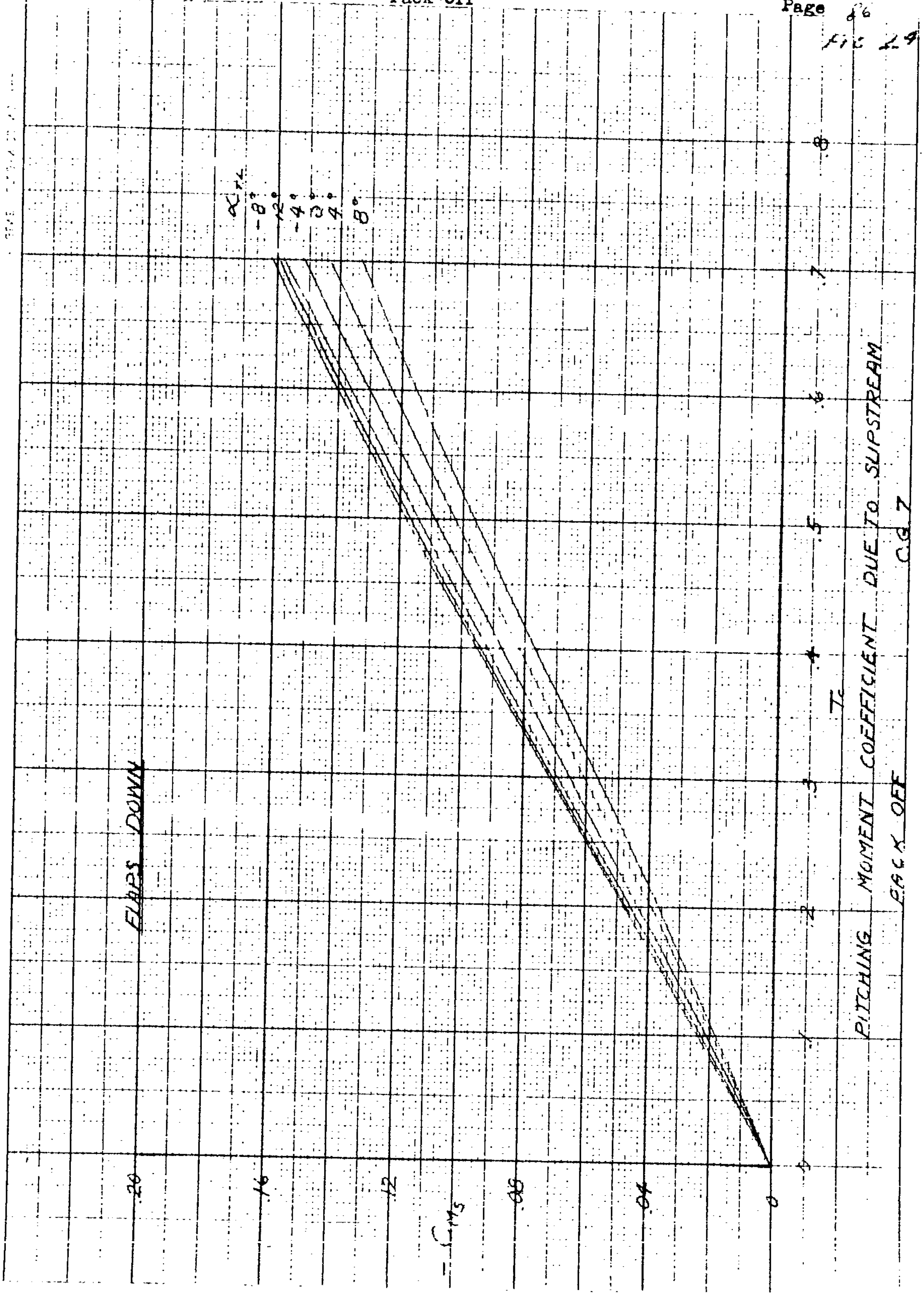


FIG 23

RESTRICTED

Fig 29


PITCHING MOMENT COEFFICIENT DUE TO SLIPSTREAM
BACK OFF C.G. 7
FLAPS DOWN
T_c
$-C_{M_S}$
α
-8°
-12°
-4°
0°
4°
8°
0
.1
.2
.3
.4
.5
.6
.7
.8
0
.04
.08
.12
.16
.20

PART II-C

C. Pitching Moments of Tailless Airplane

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

The characteristics of the tailless airplane will be determined for all of the center of gravity 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 presented in this section are all based on wing area, mean aerodynamic chord of the wing and free stream dynamic pressure.

PART II-C-1-a

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP — POWER OFF

	C.G. ①					C.G. ②				
α_L	C_{M_W}	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{A-T}}$	C_{M_W}	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{A-T}}$
-12	-.05116	-.0514	-.01120	.00028	-.1135	-.07716	-.0581	-.01242	.00028	-.1474
-8	-.06244	-.0310	-.00466	.00028	-.0978	-.06212	-.0359	-.00523	.00028	-.1030
-4	-.07842	-.0136	.00455	.00028	-.0812	-.05058	-.0173	.00549	.00028	-.0621
0	-.09837	.0018	.02302	.00028	-.0733	-.04354	-.0009	.02302	.00028	-.0214
4	-.12230	.0149	.04771	.00028	-.0594	-.04053	.0131	.04785	.00028	.0207
8	-.14960	.0241	.07480	.00028	-.0504	-.04060	.0229	.07537	.00028	.0580

C.G. ③									
C_{M_W}	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{A-T}}$	C_{M_W}	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{A-T}}$
-.05525	-.0521	-.01054	.00035	-.1175	-.08120	-.0588	-.01176	.00035	-.1514
-.06197	-.0312	-.00396	.00035	-.0968	-.06168	-.0361	-.00453	.00035	-.1020
-.07600	-.0135	.00633	.00035	-.0828	-.04820	-.0172	.00619	.00035	-.0589
-.09660	.0022	.02372	.00035	-.0703	-.04180	.0005	.02372	.00035	-.0172
-.12354	.0154	.04841	.00035	-.0594	-.04174	.0136	.04855	.00035	.0208
-.15550	.0245	.07550	.00035	-.0552	-.04650	.0233	.07607	.00035	.0532

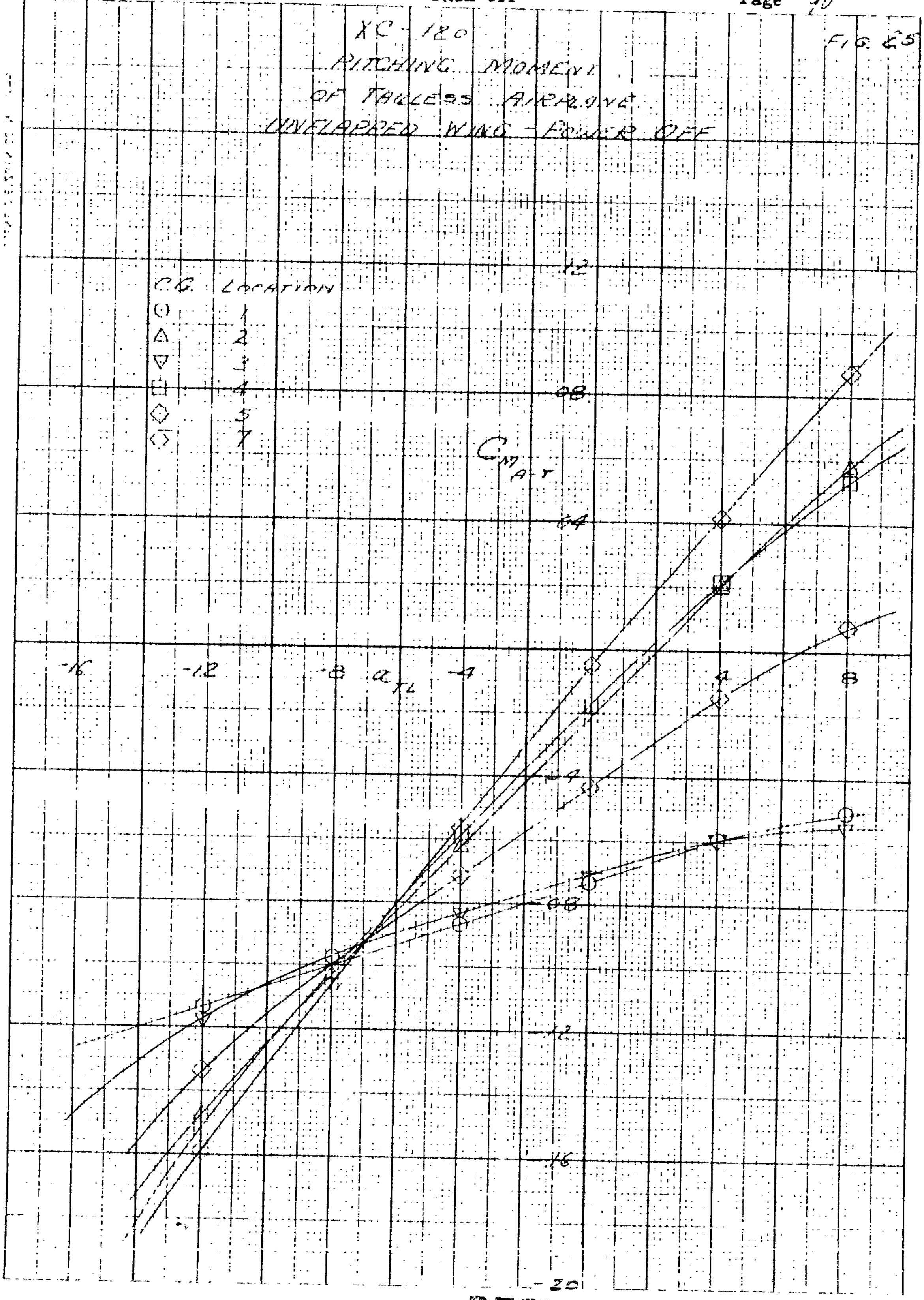
Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-C-1-a

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER OFF

	C.G. 5					C.G. 7				
α_L	C_{M_W}	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{A-T}}$	C_{M_W}	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{A-T}}$
-12	-.06728	-.0553	-.0115	.00031	-.1338	-.08367	-.0592	-.01638	.00028	-.1590
-8	-.06200	-.0337	-.0046	.00031	-.1000	-.06205	-.0369	-.00712	.00028	-.1058
-4	-.06185	-.0156	.0059	.00031	-.0713	-.04382	-.0179	.00501	.00028	-.0564
0	-.06733	.0004	.0234	.00031	-.0432	-.02977	.0016	.02302	.00028	-.0049
4	-.07778	.0141	.0482	.00031	-.0152	-.02000	.0128	.04833	.00028	.0414
8	-.09240	.0231	.0765	.00031	.0075	-.01340	.0229	.07726	.00028	.0870

RESTRICTED


XC-120
PITCHING MOMENT
OF TAILLESS AIRPLANE
UNFLAPPED WING - POWER OFF
FIG. 25
C.G. LOCATION
1
2
3
4
5
7
$C_{M_{A-T}}$
.12
.08
.04
-.04
-.08
-.12
-.16
-.20
-16
-12
-8
α_{FL}
-4
4
8

PART II-C-1-b

PITCHING MOMENTS OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
$V\sigma^{1/2} = 100\,MPH$ $T_c = .639$

C.G. 1

δ_{FL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{*P_{OFF}}$	-.1155	-.0978	-.0872	-.0733	-.0594	-.0504
C_{M_S}	.0002	.0007	.0014	.0020	.0027	.0036
C_{M_P}	.0265	.0308	.0356	.0404	.0450	.0499
$C_{M_{A-T}}^{*P_{ON}}$	-.0868	-.0663	-.0502	-.0309	-.0117	-.0069

C.G. 2

δ_{FL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{*P_{OFF}}$	-.1474	-.1030	-.0621	-.0214	.0207	.0580
C_{M_S}	-.0001	.0037	.0076	.0116	.0159	.0205
C_{M_P}	.0254	.0301	.0353	.0404	.0454	.0506
$C_{M_{A-T}}^{*P_{ON}}$	-.1221	-.0692	-.0192	.0306	.0820	.1291

* POWER

Subject:— BASIC FLIGHT CRITERIA - PACK OFF

PART II-C-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
$V\sigma^{1/2} = 100$ MPH $T_c = 639$

C.G. 3

$\alpha_{T.L.}$	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ P OFF*	-.1175	-.0968	-.0828	-.0703	-.0594	-.0552
C_{M_S}	.0010	.0012	.0017	.0024	.0034	.0046
C_{M_P}	.0004	.0048	.0096	.0142	.0189	.0237
$C_{M_{A-T}}$ P ON*	-.1161	-.0908	-.0715	-.0537	-.0371	-.0269

C.G. 4

$\alpha_{T.L.}$	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ P OFF*	-.1514	-.1020	-.0589	-.0172	.0208	.0532
C_{M_S}	.0006	.0042	.0079	.0120	.0166	.0215
C_{M_P}	-.0006	.0040	.0092	.0142	.0193	.0245
$C_{M_{A-T}}$ P ON*	-.1514	-.0938	-.0418	.0090	.0567	.0992

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

DATE 1 May 1949 — REVISED

Subject:- BASIC FLIGHT CRITERIA - PACK OFF

PART II-C-1-b

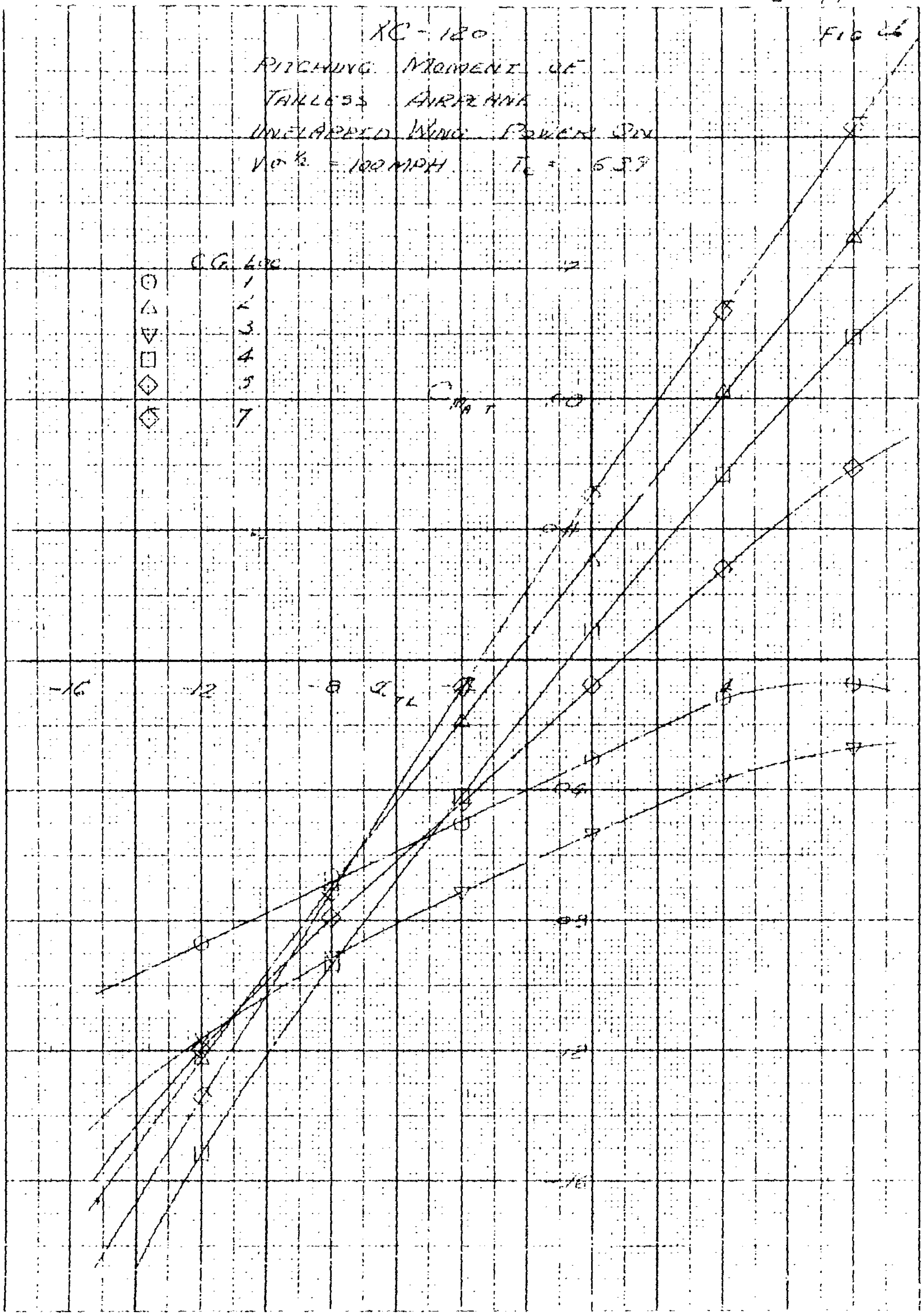
PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP — POWER ON
$V_{0\%} = 100$ MPH $T_c = .639$

C.G. 5

α_{TL}	-12	-8	-4	0	4	8
$C_{MAT}^{P\,OFF}$	-.1338	-.1000	-.0713	-.0432	-.0152	.0075
C_{MS}	.0004	.0025	.0049	.0074	.0103	.0133
C_{MP}	.0135	.0180	.0230	.0278	.0326	.0376
$C_{MAT}^{P\,ON}$	-.1199	-.0795	-.0434	-.0083	.0277	.0584

C.G. 9

α_{TL}	-12	-8	-4	0	4	8
$C_{MAT}^{P\,OFF}$	-.1590	-.1058	-.0564	-.0049	.0414	.0870
C_{MS}	-.0004	.0046	.0096	.0146	.0200	.0256
C_{MP}	.0252	.0301	.0384	.0404	.0455	.0504
$C_{MAT}^{P\,ON}$	-.1342	-.0711	-.0084	.0501	.1069	.1630


XC-120
Pitching Moment of
Tailless Airplane
C.G. Loc.
1
2
3
4
5
7
-16
-12
-8
-4
4

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

Subject: BASIC FLIGHT CRITERIA - PACK OFF — DATE 1 May 1949 — REVISED

PART II-0-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
$V_{0.7}$ = 160 MPH T_c = .198

C.G. ①

α	-12	-8	-4	0	4	8
$C_{M_{A.T}}^{P\,OFF}$	-.1135	-.0978	-.0872	-.0733	-.0594	-.0504
C_{M_S}	0	.0032	.0042	.0027	.0016	.0016
C_{M_P}	0	.0040	.0083	.0126	.0169	.0212
$C_{M_{A.T}}^{P\,ON}$	-.1135	-.0931	-.0745	-.0580	-.0413	-.0216

C.G. ③

$C_{M_{A.T}}^{P\,OFF}$	-.1474	-.1030	-.0621	-.0214	.0207	.0580
C_{M_S}	-.0008	.0010	.0027	.0045	.0066	.0086
C_{M_P}	-.0010	.0034	.0083	.0126	.0173	.0218
$C_{M_{A.T}}^{P\,ON}$	-.1492	-.0986	-.0511	-.0043	.0446	.0884

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

DATE 1 May 1949

REVISED

Subject: **BASIC FLIGHT CRITERIA - PACK OFF**

PART II-C-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE

FLAPS & GEAR UP — POWER ON

$V_{T}^{1/2}$ = 160 MPH T_c = .198

C.G. ③

α_{TL}	-12	-8	-4	0	4	8
$C_{MA.T}$ P OFF	-.1175	-.0968	-.0828	-.0703	-.0594	-.0552
C_{MS}	.0003	.0003	.0005	.0009	.0015	.0022
C_{MP}	-.0082	-.0041	.0001	.0045	.0088	.0130
$C_{MA.T}$ P ON	-.1254	-.1006	-.0822	-.0649	-.0491	-.0400

C.G. ④

α_{TL}	-12	-8	-4	0	4	8
$C_{MA.T}$ P OFF	-.1514	-.1020	-.0589	-.0172	.0208	.0532
C_{MS}	-.0005	.0002	.0001	.0002	.0005	.0009
C_{MP}	-.0091	-.0048	-.0003	.0045	.0091	.0137
$C_{MA.T}$ P ON	-.1610	-.1066	-.0590	-.0125	.0304	.0678

RESTRICTED

PART II-C-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
$V_{T}\sqrt{2}$ = 160 MPH T_c = .198

C.G. 5

a_{R}	-12	-8	-4	0	4	8
$C_{M_{A.T}}^{P.OFF}$	-.1338	-.1000	-.0713	-.0432	-.0152	.0075
C_{M_S}	-.0003	.0007	.0018	.0029	.0043	.0058
C_{M_F}	-.0043	[illegible]	.0046	.0087	.0131	.0175
$C_{M_{A.T}}^{P.ON}$	-.1384	-.0995	-.0653	-.0316	.0022	.0308

C.G. 7

a_{R}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P.OFF}$	-.1590	-.1058	-.0564	-.0099	.0414	.0870
C_{M_S}	-.0010	.0012	.0034	.0057	.0083	.0107
C_{M_F}	-.0013	.0032	.0078	.0126	.0174	.0211
$C_{M_{A-T}}^{P.ON}$	-.1613	-.1014	-.0452	.0134	.0171	.1196

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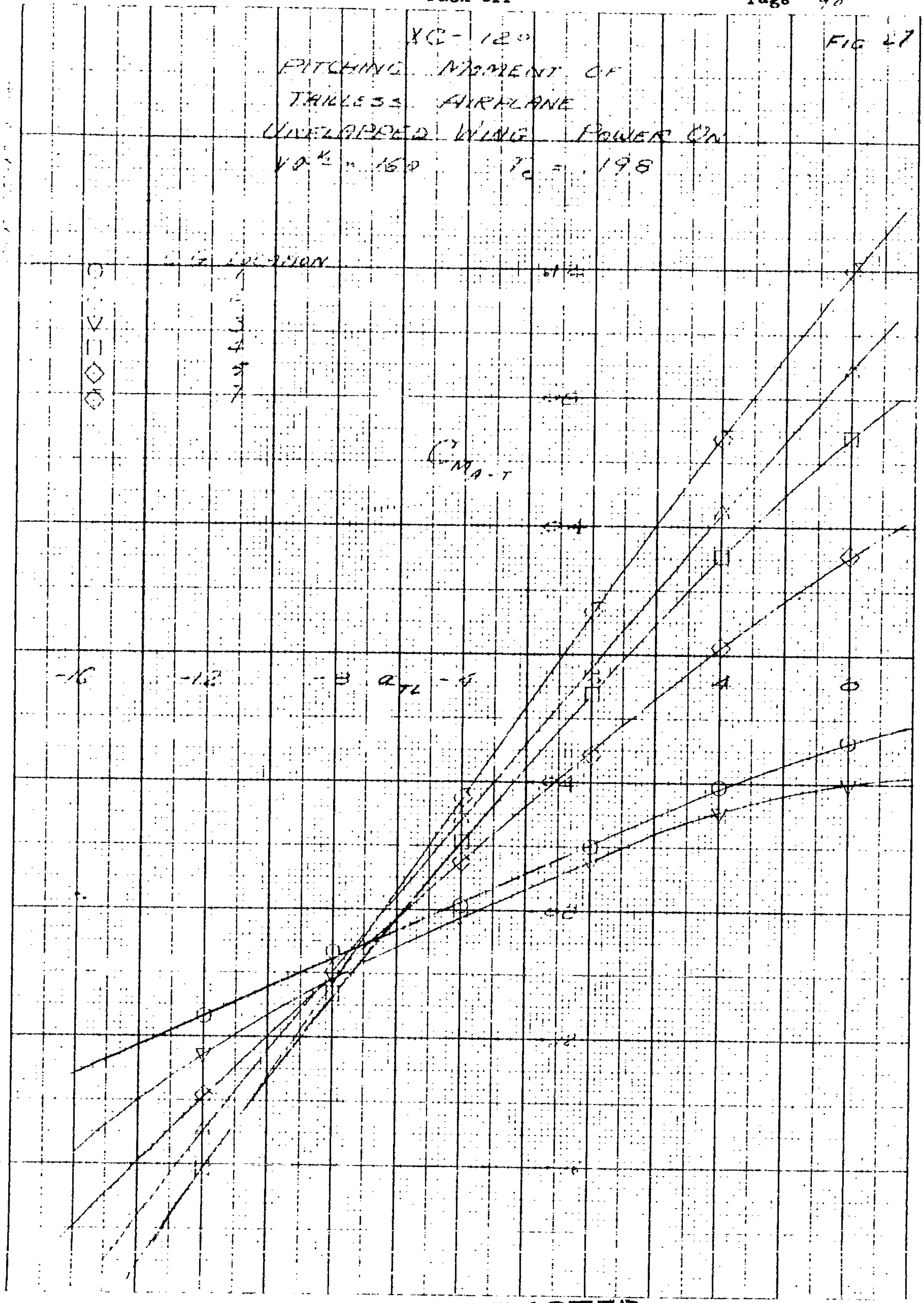

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Subject:- BASIC FLIGHT CRITERIA - PACK OFF

PART II-C-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
V 7½ - 185 MPH T_c = .132

C.G. ①

α_{TC}	-12	-8	-4	0	4	8
$C_{M_{A\,T}}$ P OFF	-.1125	-.0978	-.0872	-.0733	-.0594	-.0504
C_{M_S}	0	.0001	.0003	.0005	.0008	.0012
C_{M_P}	-.0045	[illegible]	.0059	.0044	.0169	.0173
C_{M_T} P ON	-.1130	[illegible]	[illegible]	-.0644	-.0457	-.0511

C.G. ②

α_{TC}	-12	-8	-4	0	4	8
$C_{M_{A\,T}}$ P OFF	-.1474	-.1030	-.0621	-.0214	.0207	.0580
C_{M_S}	-.0006	.0006	.0019	.0031	.0046	.0062
C_{M_P}	-.0055	-.0012	.0036	.0084	.0136	.0180
C_{M_T} P ON	-.1535	-.1036	-.0576	-.0099	.0385	.0822

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Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-C-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
$V^2/2 = 185$ $T_c = .132$

C.G. 3

α_{TL}	-12	-8	-4	0	4	8
C_{M_A} P OFF	-.1175	-.0968	-.0828	-.0703	-.0594	-.0552
C_{M_S}	.0002	.0002	.0003	.0006	.0011	.0017
C_{M_P}	-.0047	-.0060	-.0015	.0030	.0074	.0119
$C_{M_{A-T}}$ P ON	-.1272	-.1026	-.0840	-.0667	-.0509	-.0416

C.G. 4

$C_{M_{A-T}}$ P OFF	-.1514	-.1020	-.0589	-.0172	.0208	.0532
C_{M_S}	-.0004	.0007	.0019	.0032	.0049	.0067
C_{M_P}	-.0109	-.0067	-.0018	.0030	.0078	.0126
$C_{M_{A-T}}$ P ON	-.1627	-.1080	-.0588	-.0110	.0335	.0725

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PART II-C-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP — POWER ON
$V\sigma^{1/2}$ = 185 MPH T_c = .132

C.G 5

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{a.T}}$ P.OFF	-.1338	-.1000	-.0713	-.0432	-.0152	.0075
C_{M_S}	-.0002	.0005	.0012	.0021	.0031	.0043
C_{M_D}	-.0075	-.0034	.0012	.0058	.0104	.0150
$C_{M_{a.T}}$ P.ON	-.1415	[illegible]	[illegible]	[illegible]	[illegible]	.0268

C.G 7

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{a.T}}$ P.OFF	-.1590	-.1058	-.0564	-.0049	.0419	.0870
C_{M_S}	-.0008	.0007	.0024	.0037	.0058	.0078
C_{M_D}	-.0054	-.0014	[illegible]	.0084	.0151	.0182
$C_{M_{a.T}}$ P.ON	-.1652	-.1065	-.0523	.0074	.0623	.1130

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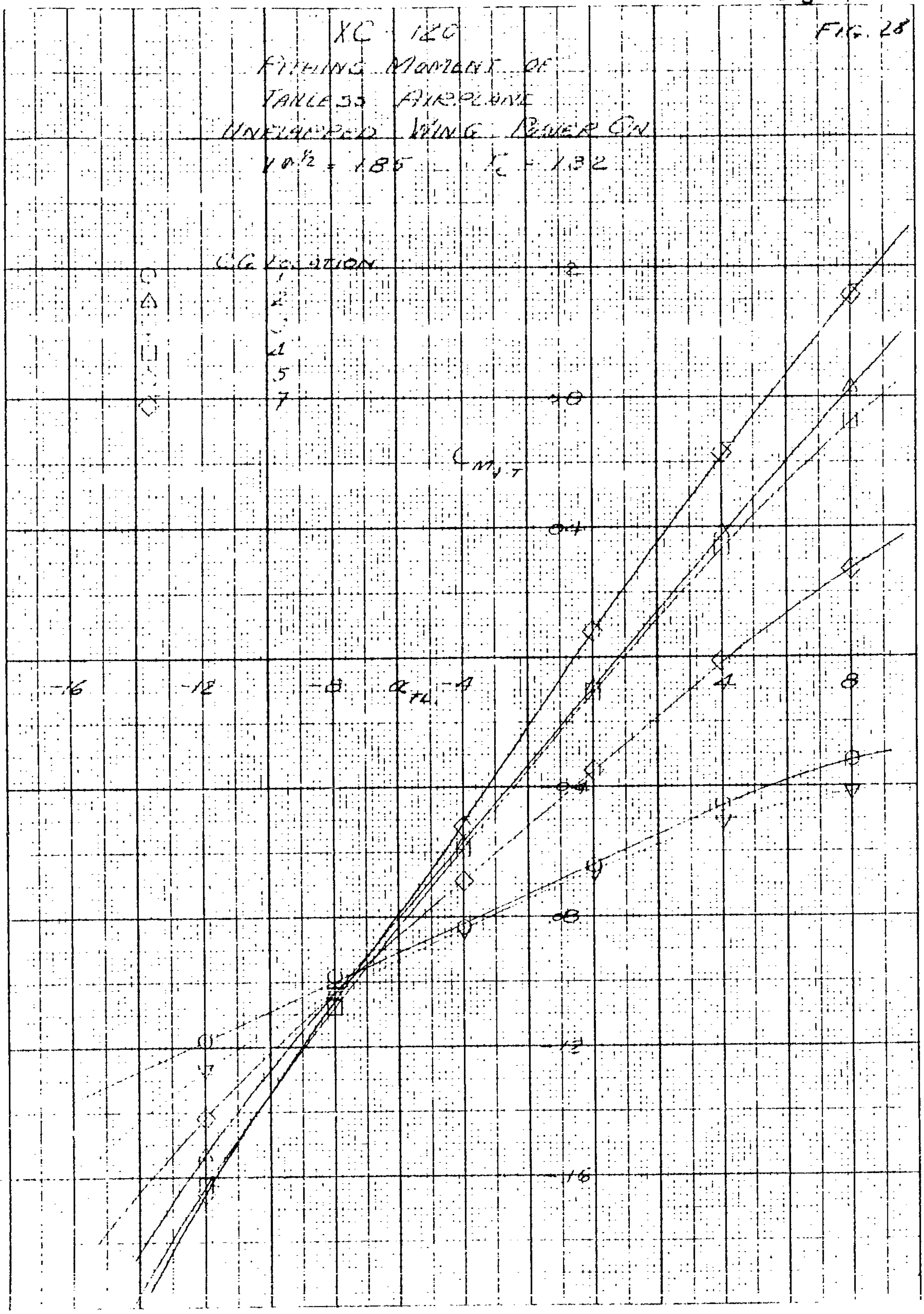
XC 120
Fig. 28
C.G. Location
1
2
4
5
7
$C_{m,T}$
α_{TL}
-16
-12
-8
-4
4
8
.08
.04
.04
.08
.12
.16

Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-C-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
V_{e} = 265 MPH T_c = .047

C. G. 1

$\alpha_{T.L.}$	-12	-8	-4	0	4	8
$C_{M_{A.T.}}^{P.OFF}$	-.1135	-.0978	-.0872	-.0733	-.0594	-.0504
C_{M_S}	0	0	.0001	.0002	.0004	.0007
C_{M_P}	-.0119	-.0011	-.0020	0	.0077	.0127
$C_{M_{A.T.}}^{P.ON}$	-.1254	-.1049	-.0891	-.0731	-.0513	-.0370

C. G. 2

$\alpha_{T.L.}$	-12	-8	-4	0	4	8
$C_{M_{A.T.}}^{P.OFF}$	-.1474	-.1030	-.0621	-.0214	.0207	.0580
C_{M_S}	-.0003	.0001	.0006	.0011	.0017	.0024
C_{M_P}	-.0131	-.0078	-.0023	0	.0081	.0135
$C_{M_{A.T.}}^{P.ON}$	-.16[illegible]	-.1167	-.0628	-.0203	.0305	.0739

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PART II-C-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP — POWER ON
$V_{0}^{1/2}$ = 265 MPH T_c = .047

C. G. 3

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P\ OFF}$	-.1175	-.0968	-.0828	-.0703	-.0594	-.0552
C_{M_S}	.0001	0	.0001	.0002	.0006	.0011
C_{M_P}	-.0138	-.0089	-.0038	0	.0059	.0109
$C_{M_{A-T}}^{P\ ON}$	-.1312	-.1057	-.0865	-.0701	-.0529	-.0432

C. G. 4

$C_{M_{A-T}}^{P\ OFF}$	-.1514	-.1020	-.0589	-.0172	.0208	.0532
C_{M_S}	-.0002	.0001	.0006	.0011	.0020	.0031
C_{M_P}	-.0148	-.0097	-.0042	0	.0062	.0117
$C_{M_{A-T}}^{P\ ON}$	-.1664	-.1116	-.0625	-.0161	.0290	.0680

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PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
$V_0{}^{1/2}$ = 265 MPH T_c = .047

C. G. 5

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A.T}}$ POFF	-.1338	-.1000	-.0713	-.0432	-.0152	.0075
C_{M_S}	0	.0001	.0004	.0006	.0013	.0020
C_{M_P}	-.0136	-.0083	-.0030	0	.0069	.0122
$C_{M_{A.T}}$ PON	-.1470	-.1056	-.0737	-.0426	-.0070	.0217

C. G. 7

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A.T}}$ POFF	-.1590	-.1058	-.0564	-.0097	.0414	.0870
C_{M_S}	-.0014	.0001	.0007	.0014	.0022	.0033
C_{M_P}	-.0133	-.0079	-.0064	0	.0082	.0137
$C_{M_{A.T}}$ PON	-.1727	-.1136	-.0581	-.0035	.0518	.1040

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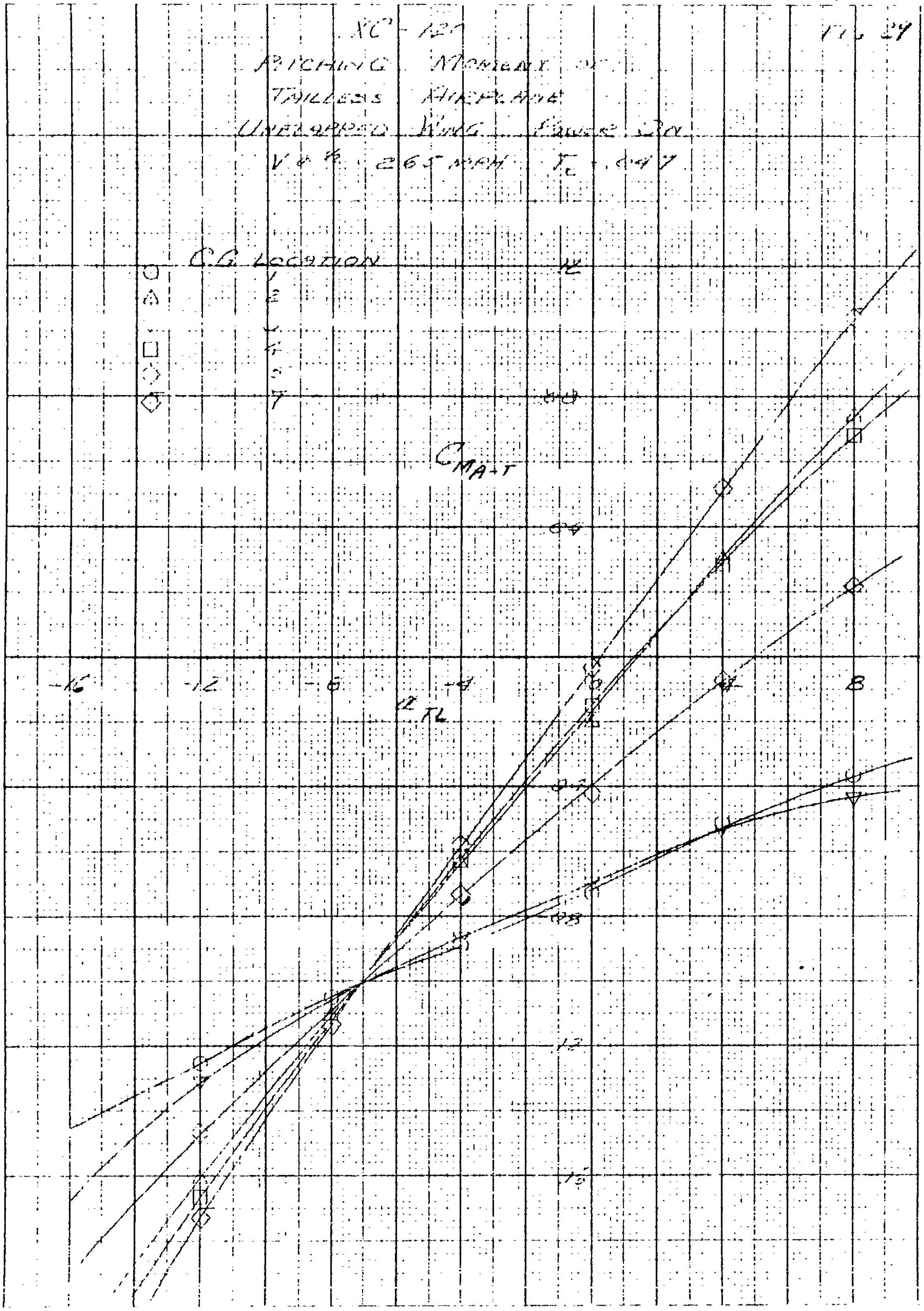
XC-120
FIG. 24
PITCHING MOMENT OF
TAILLESS AIRPLANE
C.G. LOCATION
$C_{M_{A+T}}$
-16
-12
-8
-4
8
α_{TL}

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
$V_{0.6}^{1/2}$ 331.3 MPH $T_c = .024$

C. G. ①

X_{TC}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P\ OFF}$	-.1135	-.0978	-.0812	-.0733	-.0594	-.0504
C_{M_S}	.0001	0	.0001	.0001	.0002	.0005
C_{M_P}	-.0152	-.0092	-.0341	0	.0070	.0121
$C_{M_{A-T}}^{P\ ON}$	-.1286	-.1070	-.0912	-.0732	-.0522	-.0377

C. G. ②

X_{TC}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P\ OFF}$	-.1474	-.1030	-.0621	-.0214	.0607	.0580
C_{M_S}	-.0001	0	.0004	.0006	.0007	.0016
C_{M_P}	-.0166	-.0100	-.0045	0	.0074	.0129
$C_{M_{A-T}}^{P\ ON}$	-.1641	-.1130	-.0662	-.0208	.0290	.0725

Subject:— BASIC FLIGHT CRITERIA - PACK OFF

PART II-C-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
$V\sigma^{1/2}$ = 331.3 MPH T_c = .024

C. G. 3

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P\ OFF}$	-.1175	-.0968	-.0828	-.0703	-.0594	-.0552
C_{M_S}	.0002	0	.0001	.0001	.0004	.0009
C_{M_P}	-.0162	-.0101	-.0051	0	.0061	.0111
$C_{M_{A-T}}^{P\ ON}$	-.1335	-.1069	-.0878	-.0702	-.0529	-.0432

C. G. 4

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P\ OFF}$	-.1514	-.1020	-.0589	-.0172	.0208	.0532
C_{M_S}	.0001	0	.0004	.0006	.0012	.0020
C_{M_P}	-.0175	-.0110	-.0055	0	.0065	.0120
$C_{M_{A-T}}^{P\ ON}$	-.1688	-.1130	-.0640	-.0166	.0285	.0672

PART II-C-1-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR UP - POWER ON
$V\sigma^{1/2}$ = 331.3 MPH T_c = .024

C.G. 5

α_R	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P_{OFF}}$	-.1338	-.1000	-.0713	-.0432	-.0152	.0075
C_{M_S}	.0001	0	.0002	.0004	.0008	.0013
C_{M_P}	-.0162	-.0100	-.0048	0	.0067	.0120
$C_{M_{A-T}}^{P_{ON}}$	-.1499	-.1100	-.0759	-.0428	-.0077	.0208

C.G. 7

α_R	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P_{OFF}}$	-.1590	-.1058	-.0564	-.0049	.0414	.0870
C_{M_S}	-.0001	.0001	.0005	.0007	.0013	.0020
C_{M_P}	-.0169	-.0102	-.0047	.0014	.0075	.0131
$C_{M_{A-T}}^{P_{ON}}$	-.1760	-.1159	-.0606	-.0028	.0502	.1021

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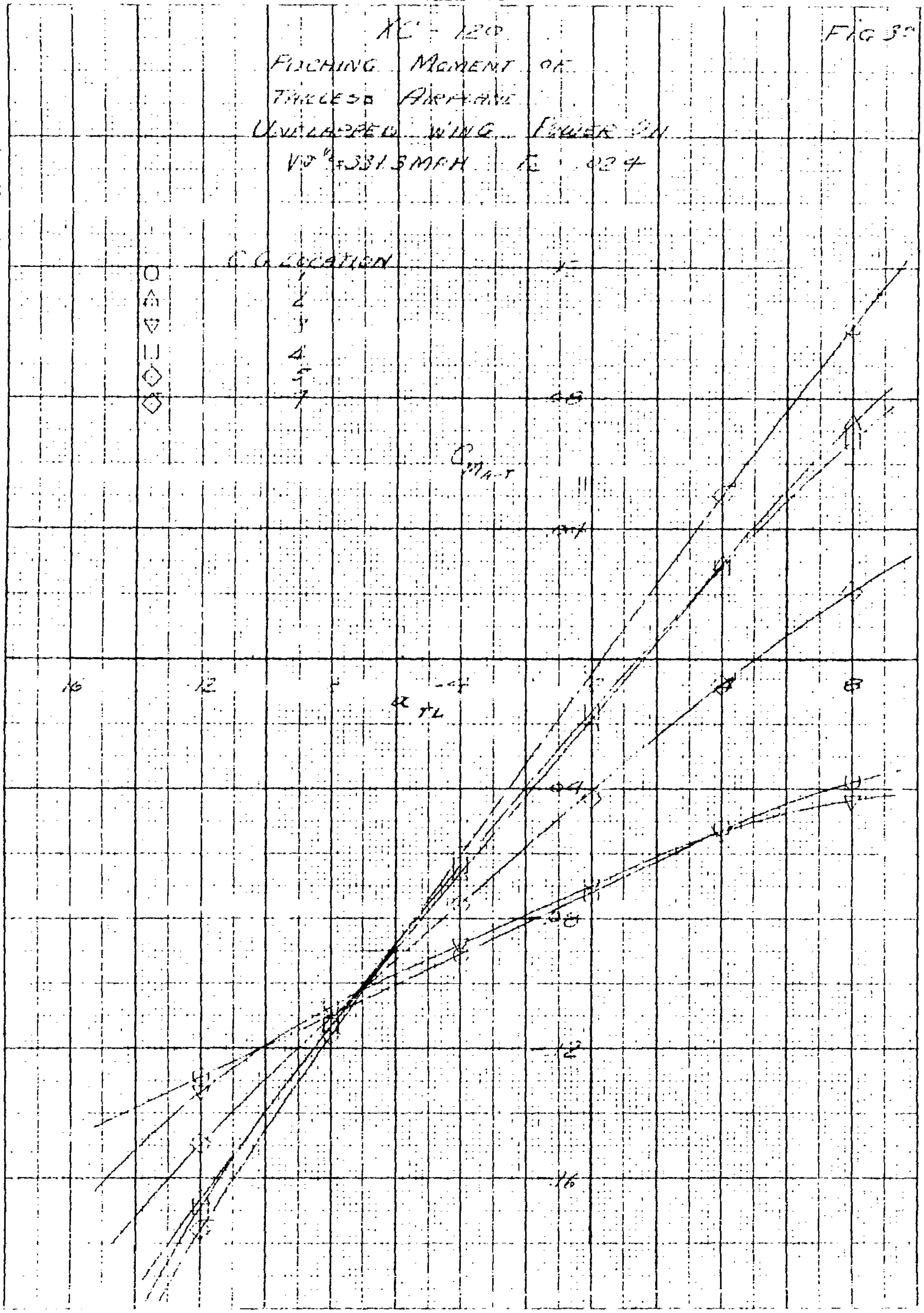

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Subject: – BASIC FLIGHT CRITERIA – PACK OFF

PART II-C-2-a

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR DOWN – POWER OFF

	C.G. 1					
α_{TL}	$C_{M_{WF}}$	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{LG}}$	$C_{M_{A-T}}$
-12	-.14559	-.0514	-.0112	.00028	-.03870	-.2466
-8	-.15860	-.0310	-.00466	.00028	-.03820	-.2323
-4	-.17605	-.0136	.00455	.00028	-.03780	-.2226
0	-.19780	.0018	.02302	.00028	-.03700	-.2097
4	-.22396	.0149	.04771	.00028	-.03780	-.1989
8	-.25290	.0241	.07480	.00028	-.03830	-.1720

	C.G. 2					
α_{TL}	$C_{M_{WF}}$	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{LG}}$	$C_{M_{A-T}}$
-12	-.14522	-.0581	-.01292	.00028	-.03766	-.2532
-8	-.13140	-.0359	-.06523	.00028	-.03757	-.2099
-4	-.12165	-.0173	.00549	.00028	-.03739	-.1706
0	-.11610	.0009	.02302	.00028	-.03700	-.1289
4	-.11496	.0131	.04755	.00028	-.03739	-.0913
8	-.11650	.0229	.07537	.00028	-.03757	-.0555

	C.G. 3					
α_{TL}	$C_{M_{WF}}$	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{LG}}$	$C_{M_{A-T}}$
-12	-.14305	-.0521	-.01054	.00035	-.03450	-.2378
-8	-.15280	-.0312	-.00396	.00035	-.0341	-.2217
-4	-.16930	-.0135	.00633	.00035	-.0335	-.2096
0	-.19250	.0022	.02372	.00035	-.0327	-.1989
4	-.22270	.0154	.04841	.00035	-.03350	-.1920
8	-.25750	.0245	.07550	.00035	-.03410	-.1913

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR DOWN - POWER OFF

C.G. (4)

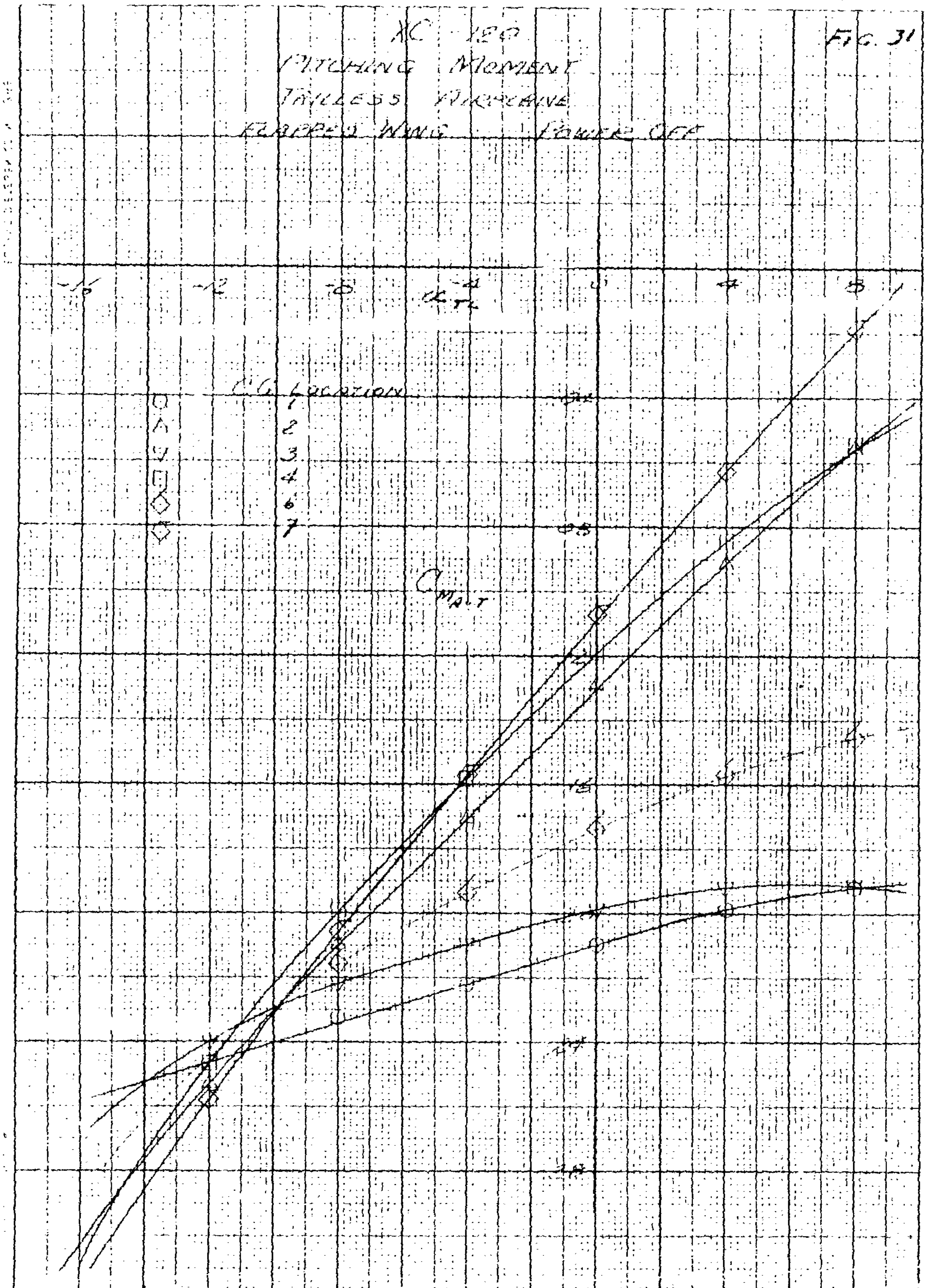
α_{TL}	$C_{M_{WF}}$	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{LG}}$	$C_{M_{A-T}}$
-12	-.14268	-.0588	-.01176	.00035	-.03346	-.2464
-8	-.1256	-.0361	-.00453	.00035	-.03337	-.1993
-4	-.11490	-.0172	.00619	.00035	-.03309	-.1587
0	-.1108	.0095	.02372	.00035	-.03270	-.1189
4	-.11360	.0136	.04855	.00035	-.03309	-.0842
8	-.12116	.0233	.07607	.00035	-.03337	-.0548

(6)

$C_{M_{WF}}$	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{LG}}$	$C_{M_{A-T}}$
-.14312	-.0544	-.0117	.00035	-.03460	-.2443
-.14449	-.0329	-.00463	.00035	-.03420	-.2158
-.15212	-.0149	.00612	.00035	-.03370	-.1942
-.16629	.0012	.02367	.00035	-.03300	-.1742
-.18727	.0149	.04852	.00035	-.03370	-.1572
-.21267	.0233	.07787	.00035	-.03420	-.1454

(7)

$C_{M_{WF}}$	C_{M_N}	C_{M_B}	$C_{M_{VT}}$	$C_{M_{LG}}$	$C_{M_{A-T}}$
-.14513	-.0592	-.01638	.00028	-.03739	-.2578
-.12460	-.0369	-.00712	.00028	-.03740	-.2057
-.10805	-.0179	.00501	.00028	-.03730	-.1580
-.09550	.0016	.02302	.00028	-.03700	-.1076
-.08736	.0128	.04833	.00028	-.03730	-.0633
-.08235	.0229	.07726	.00028	-.03740	-.0173


XC 120
Pitching Moment
Tailless Airplane
Flapped Wing Power Off
Fig. 31
-16
-12
-8
-4
0
4
8
C.G. Location
1
2
3
4
6
7

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR DOWN — POWER ON
$V\sigma^{1/2} = 100$ MPH $T_c = .639$

C. G. 1

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P\,OFF}$	-.2466	-.2323	-.2226	-.2097	-.1989	-.1920
C_{M_S}	-.1622	-.1682	-.1724	-.1742	-.1748	-.1743
C_{M_P}	.0265	.0308	.0356	.0404	.0450	.0499
$C_{M_{A-T}}^{P\,ON}$	-.3823	-.3697	-.3594	-.3435	-.3287	-.3162

C. G. 2

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P\,OFF}$	-.2532	-.2099	-.1706	-.1289	-.0913	-.0555
C_{M_S}	-.1511	-.1538	-.1514	-.1465	-.1405	-.1321
C_{M_P}	.0254	.0301	.0353	.0404	.0454	.0506
$C_{M_{A-T}}^{P\,ON}$	-.3787	-.3336	-.2867	-.2350	-.1864	-.1370

PART II-C-2-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR DOWN - POWER ON
$V\sigma^{1/2}$ = 100 MPH T_c = .639

C.G. 3

α_{TL}	-12	-8	-4	0	4	8
$C_{MA\text{-}T}^{ROFF}$	-.2398	-.2217	-.2096	-.1989	-.1920	-.1913
C_{Ms}	-.1608	-.1668	-.1711	-.1732	-.1740	-.1736
C_{Mp}	.0004	.0048	.0096	.0142	.0189	-.0237
$C_{MA\text{-}T}^{RON}$	-.4002	-.3837	-.3711	-.3579	-.3471	-.3411

C.G. 4

$C_{MA\text{-}T}^{ROFF}$	-.2464	-.1993	-.1587	-.1189	-.0842	-.0548
C_{Ms}	-.1497	-.1524	-.1501	-.1455	-.1397	-.1324
C_{Mp}	-.0006	.0040	.0092	.0142	.0193	.0245
$C_{MA\text{-}T}^{RON}$	-.3967	-.3477	-.2998	-.2510	-.2046	-.1627

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PART II-C-2-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR DOWN — POWER ON
$V\sigma^{1/2}$ = 100 MPH $\quad T_c$ = .639

C. G. 6

[illegible]	-12	-8	-4	0	4	8
$C_{M_{y-T}}$ P OFF	-.2443	-.2158	-.1942	-.1742	-.1572	-.1454
C_{M_S}	-.1572	-.1623	-.1644	-.1644	-.1630	-.1604
C_{M_P}	.0021	.0065	.0114	.0162	.0209	.0258
$C_{M_{y-T}}$ P.ON	-.3975	-.3716	-.3472	-.3224	-.2993	-.2800

C. G. 7

	-12	-8	-4	0	4	8
$C_{M_{y-T}}$ P OFF	-.2578	-.2057	-.1580	-.1076	-.0633	-.0193
C_{M_S}	-.1476	-.1494	-.1448	-.1379	-.1300	-.1204
C_{M_P}	.0252	.0301	.0384	.0404	.0455	.0504
$C_{M_{y-T}}$ P ON	-.3802	-.3250	-.2694	-.2051	-.1378	-.0893

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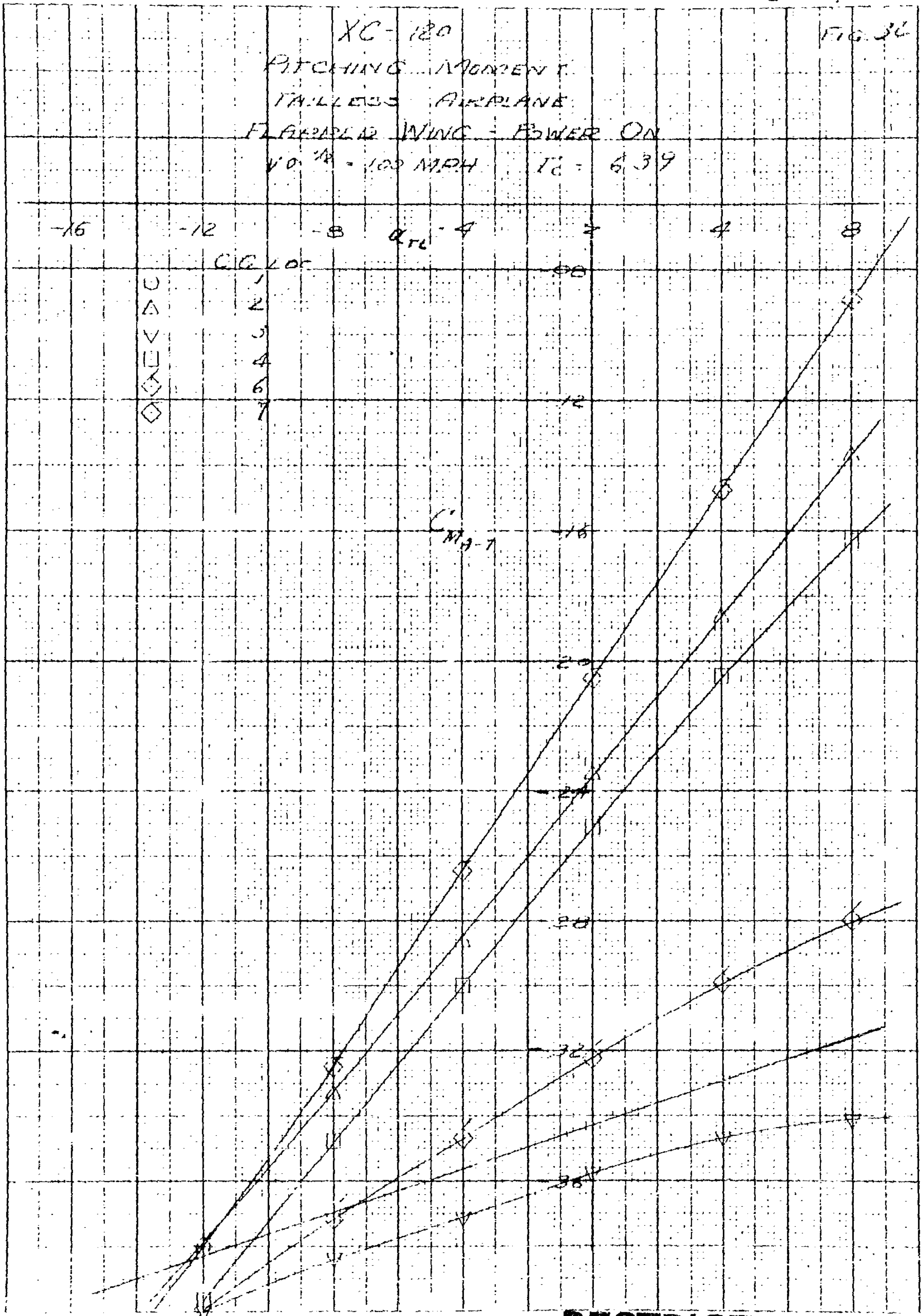
XC-120
Fig. 36
Pitching Moment
Tailless Airplane
Flapped Wing - Power On
-16
-12
-8
-4
0
4
8
C.G. Loc
1
2
3
4
6
7
.08
.12
.16
.20
.24
.28
.32
.36

PART II-C-2-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR DOWN — POWER ON
V_{CAL} = 160 MPH T_c = .198

C.G. ①

α	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ P-OFF	-.2456	-.2323	-.2226	-.2097	-.1989	-.1920
C_{M_S}	-.0523	-.0542	-.0553	-.0564	-.0564	-.0568
C_{M_P}	-	.0040	.0093	.0126	.0169	.0212
$C_{M_{A-T}}$ P-ON	[illegible]	-.2825	-.2676	-.2535	-.2384	-.2276

C.G. ②

$C_{M_{A-T}}$ P-OFF	-.2532	-.2099	-.1706	-.1287	-.0913	-.0555
C_{M_S}	-.0435	-.0490	-.0474	-.0456	-.0428	-.0406
C_{M_P}	.0010	.0034	.0083	.0126	.0173	.0218
$C_{M_{A-T}}$ P-ON	-.3027	-.2555	-.2097	-.1619	-.1166	-.0743

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PART II-C-2-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR DOWN – POWER ON
$V_{\sigma^{1/2}} = 160$ MPH $T_c = .178$

C. G. 3

$\alpha_{F.L.}$	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ P-OFF	-.2398	-.2217	-.2096	-.1987	-.1920	-.1913
C_{M_Z}	-.0520	-.0539	-.0550	-.0566	-.0561	-.0565
C_{M_E}	-.0096	-.0041	.0001	.0045	.0093	.0130
$C_{M_{A-T}}$ P-ON	-.3000	-.2797	-.2645	-.2506	-.2393	-.2347

C. G. 4

$\alpha_{F.L.}$	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ P-OFF	-.2464	-.1993	-.1587	-.1189	-.0842	-.0548
C_{M_Z}	-.0492	-.0487	-.0471	-.0454	-.0425	-.0403
C_{M_E}	-.0091	-.0048	-.0002	.0045	.0091	.0137
$C_{M_{A-T}}$ P-ON	-.3037	-.2528	-.2060	-.1598	-.176	-.0814

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PART II-C-2-b

PITCHING MOMENT OF TAILLESS AIRPLANE
FLAPS & GEAR DOWN - POWER ON
V_0 "L" = 160 MPH T_c = .193

	C. G. ⑥					
α_{TC}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{P\ OFF}$	-.2443	-.2158	-.1942	-.1742	-.1572	-.1454
C_{M_S}	-.2578	-.0523	-.0525	-.0527	-.0517	-.0513
C_{M_P}	-.0078	-.0037	.0036	.0050	.0095	.0137
$C_{M_{A-T}}^{P\ ON}$	-.3029	-.2718	-.2451	-.2219	-.1994	-.1830
$C_{M_{A-T}}^{P\ OFF}$	-.2578	-.2057	-.1580	-.1076	-.0633	-.0193
C_{M_S}	-.0474	-.0474	-.0450	-.0422	.0385	-.0355
C_{M_e}	-.0013	.0032	.0078	.0126	.0174	.0219
$C_{M_{A-T}}^{P\ ON}$	-.3065	-.2499	-.1952	-.1372	-.0844	-.0329

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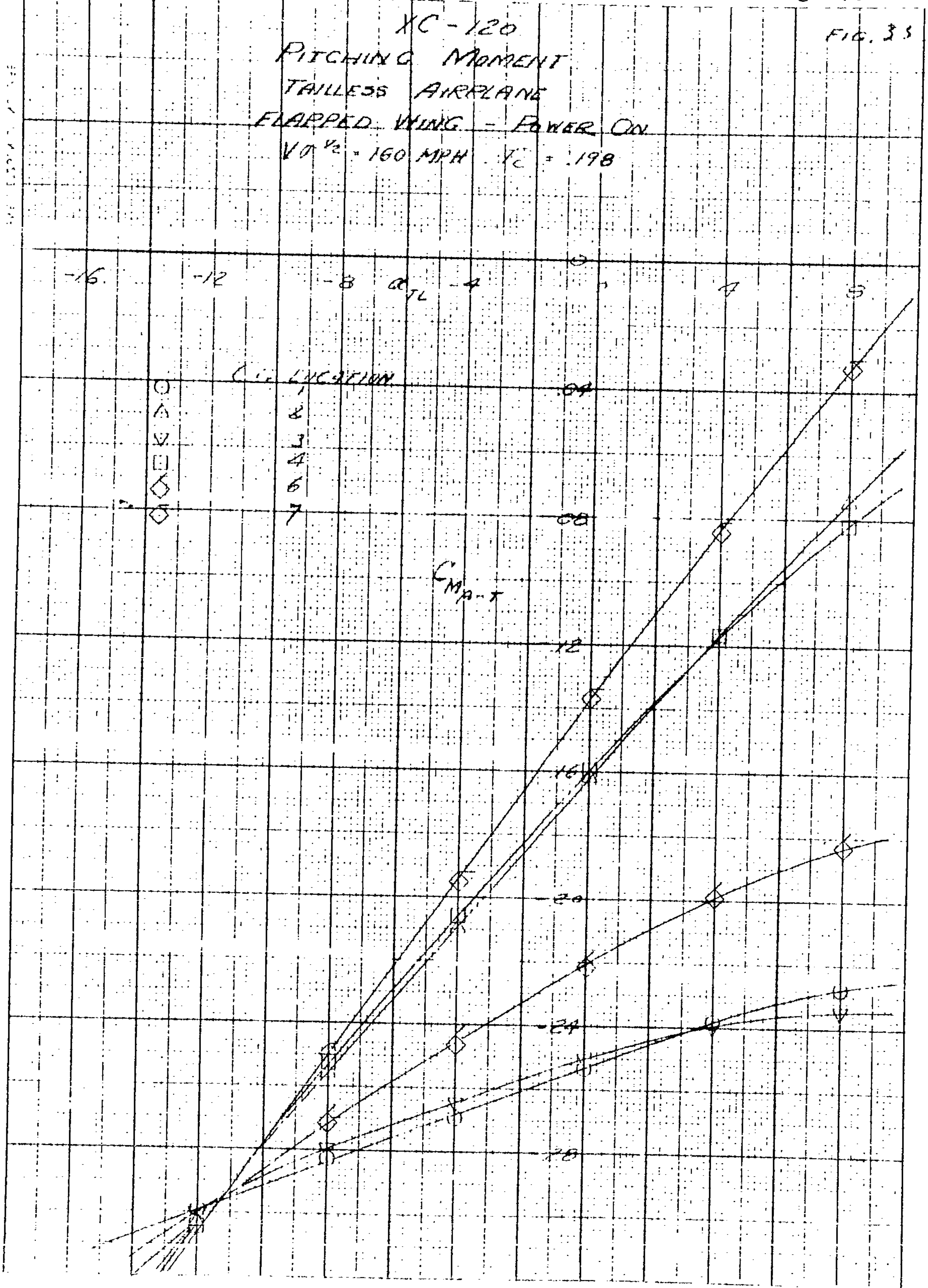
XC-120
PITCHING MOMENT
TAILLESS AIRPLANE
FLAPPED WING - POWER ON
$V\sigma^{1/2}$ = 160 MPH T_c = .198
FIG. 35
-16
-12
-8
α_{TL}
-4
0
4
8
C.G. LOCATION
1
2
3
4
6
7
.04
.08
$C_{M_{A-T}}$
.12
.16
.20
.24
.28

PART II-D

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 $C_{M_A} = 0$. 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 negligible small fraction of the total tail length

So that to balance the airplane

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

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

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

where η_{ht} = .85 lift efficiency of the horizontal tail

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

$$K = \frac{S_{ht}}{S_w} \frac{l_{ht}}{MAC_w} \frac{q_{ht}}{q}$$

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

$$K = .00142\, l_{ht}$$

The effective tail lift is then $\eta_{ht}\, C_{L_{HT}} = \dfrac{C_{M_{A-T}}}{K \frac{q_{ht}}{q}}$

based on $q_{ht}\, S_{ht}$

l_{ht} is the distance from the c.g. to the elevator hinge line.

PART II-D-1

1. Flaps and Gear Up

a. Power Off

$\frac{q_{ht}}{q} = 1.00$ page 212 reference (1)

C. G. Location	l_{ht}	K
1	621.10	.8820
2	607.64	.8628
3	621.10	.8820
4	607.64	.8628
5	613.7	.8715
6	616.7	.8757
7	604.2	.8580

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DATE 1 May 1949

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Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-D-1-a

BALANCING TAIL LIFT
FLAPS & GEAR UP POWER OFF

	C.G. 1			C.G. 2			C.G. 3		
α_{TL}	$C_{M_{A-T}}$	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	-.1135	.8820	-.1287	-.1474	.8628	-.1710	-.1175	.8820	-.1332
-8	-.0978		-.1109	-.1030		-.1194	-.0968		-.1097
-4	-.0872		-.0988	-.0621		-.0720	-.0828		-.0939
0	-.0733		-.0831	-.0214		-.0248	-.0703		-.0798
4	-.0594		-.0674	.0207		.0240	-.0594		-.0674
8	-.0504		-.0572	.0580		.0673	-.0552		-.0626

C.G. 4			C.G. 5			C.G. 7		
$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$
-.1514	.8628	-.1756	-.1938	.8715	-.1537	-.1590	.8580	-.1853
-.1020		-.1182	-.1000		-.1148	-.1058		-.1233
-.0539		-.0683	-.0713		-.0819	-.0564		-.06577
-.0172		-.0199	-.0432		-.0996	-.0049		-.00567
.0208		.0241	-.0152		-.0175	.0414		+.04826
.0532		.0616	.0075		.0086	.0870		+.1014

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Pack-Off

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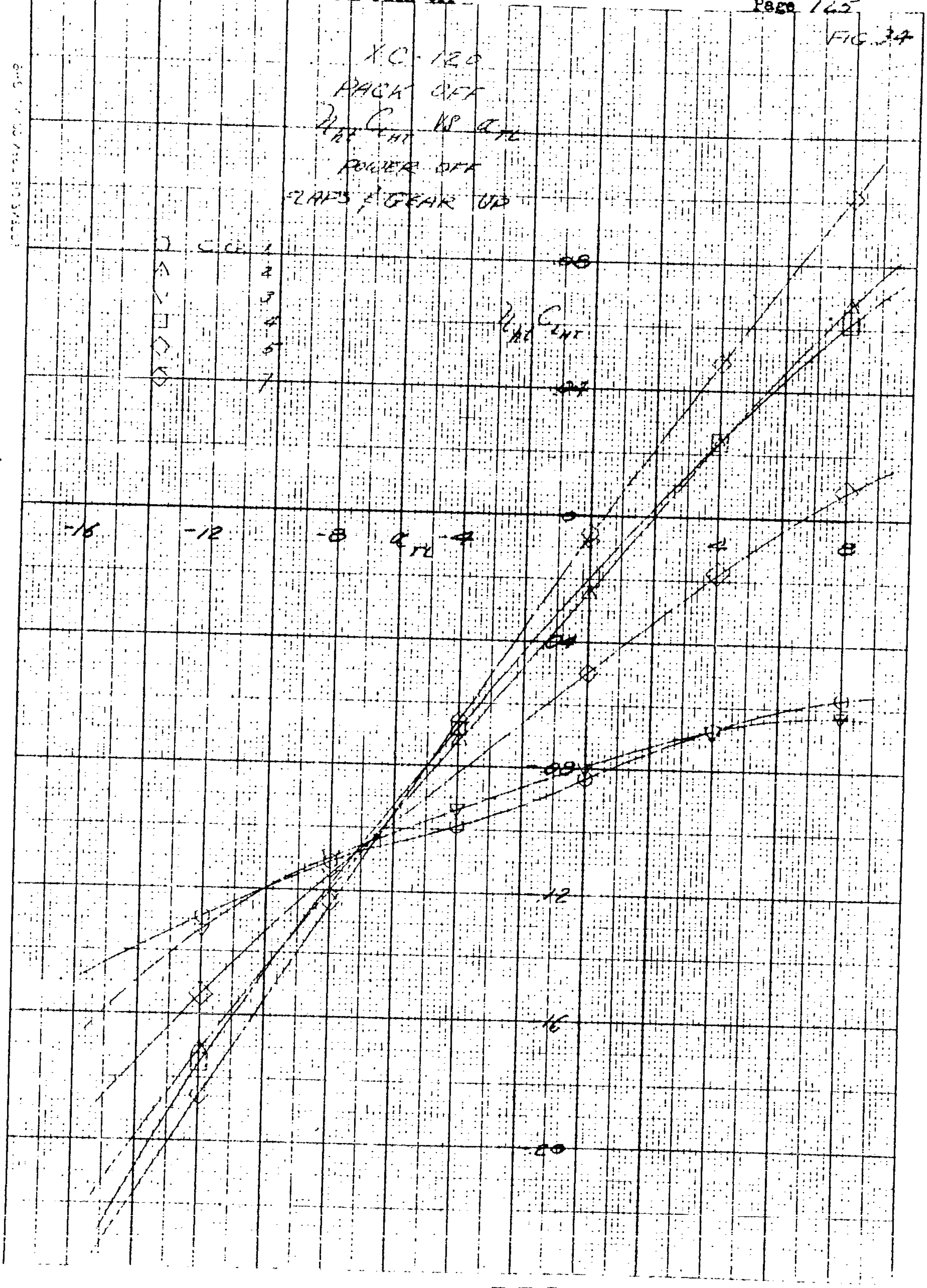
FIG. 34



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PART II-D-1

b. Power On

For power on conditions values of $\frac{q_{ht}}{q}$ will very with angle of attack and with thrust coefficient

$\frac{q_{ht}}{q}$ is obtained from figure 136 reference (1)

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PART II-D-1-b

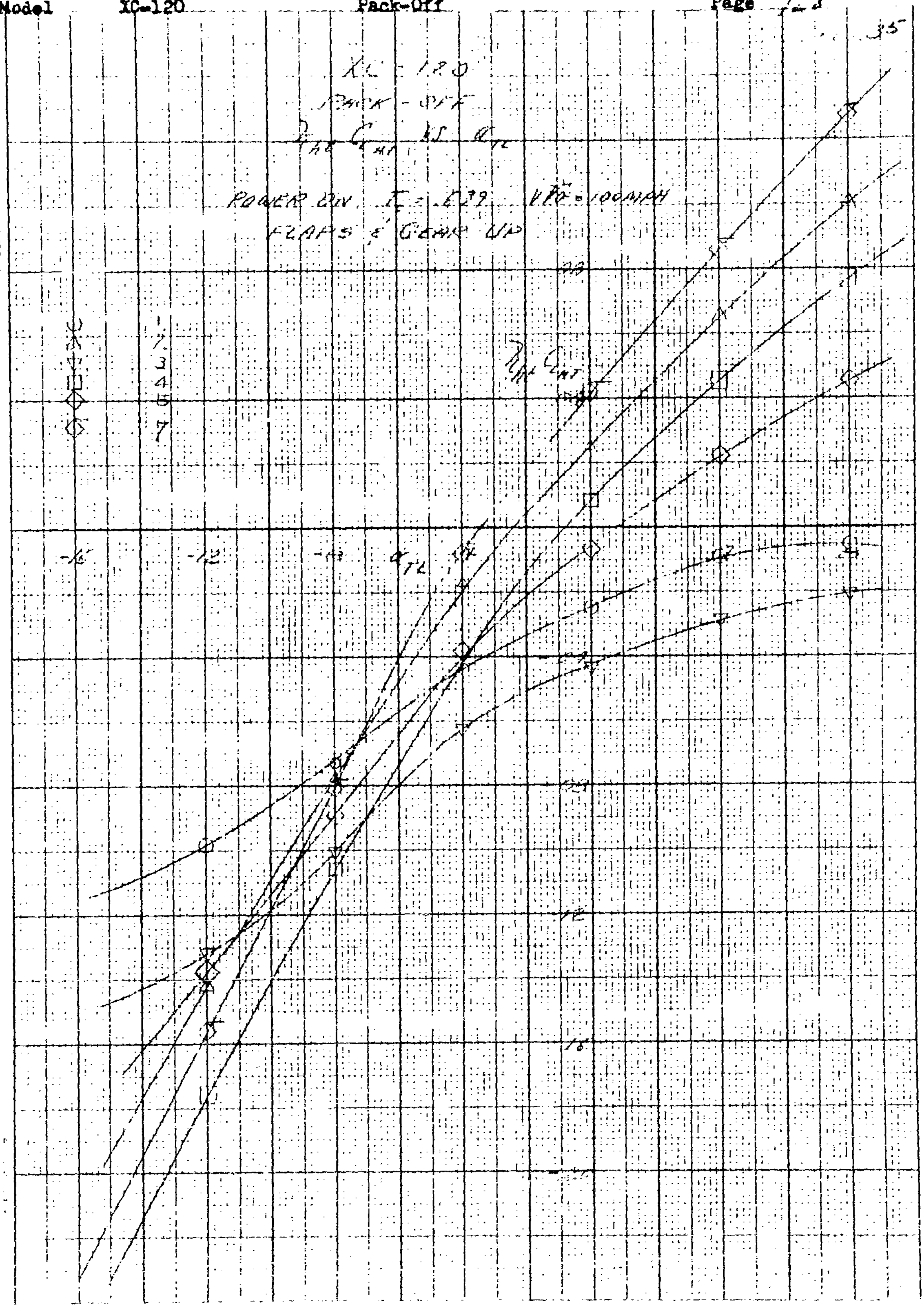
BALANCING TAIL LIFT

FLAPS & GEAR UP—POWER ON

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

		C.G. 1			C.G. 2			C.G. 3		
α_{TL}	q_{ht}/q	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$
-12	1.000	-.0868	.882	-.0985	-.1221	.8628	-.1417	-.1161	.882	-.1318
-8	1.029	-.0663	.908	-.0731	-.0692	.8878	-.0780	-.0908	.9078	-.1001
-4	1.300	-.0562	1.147	-.0438	-.0192	1.1216	-.0171	-.0715	1.147	-.0623
0	1.409	-.0309	1.238	-.0250	.0306	1.2114	.0253	-.0537	1.238	-.0434
4	1.452	-.0117	1.281	-.0091	.0820	1.2528	.0654	-.0371	1.281	-.0290
8	1.490	-.0069	1.314	-.00525	.1291	1.2856	.1005	-.0269	1.314	-.0205

C.G. 4			C.G. 5			C.G. 7		
$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$
-.1514	.8623	-.1757	-.1197	.8715	-.1378	-.1342	.8580	-.1564
-.0936	.8878	-.1058	-.0795	.8968	-.0888	-.0711	.8829	-.0805
-.0418	1.1216	-.0373	-.0434	1.1329	-.0383	-.0084	1.1154	-.0076
.0090	1.2114	.0081	-.0080	1.224	-.0065	.0501	1.2046	.0416
.0567	1.2528	.04525	.0277	1.265	.0219	.1069	1.2458	.0858
.0992	1.2856	.0772	.0584	1.299	.0450	.1630	1.2784	.1275


XC-120
PACK-OFF
POWER ON
FLAPS & GEAR UP
-16
-12
-8
-4
4
8
12
16

PART II-D-1-b

BALANCING TAIL LIFT
FLAPS & GEAR UP—POWER ON
$V\sqrt{\sigma}$ = 160 mph T_c = .198

		C.G. 1			C.G. 2			C.G. 3		
X_{TL}	q_{ht}/q	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$
-12	1.000	-.1135	.882	-.1288	-.1492	.8628	-.1730	-.1254	.882	-.1422
-8	1.060	-.0936	.882	-.1061	-.0986	.8628	-.1142	-.1006	.882	-.1141
-4	1.058	-.0785	.933	-.0842	-.0511	.9135	-.0560	-.0822	.933	-.0882
0	1.148	-.0600	1.013	-.0592	-.0043	.9900	-.0043	-.0649	1.013	-.0641
4	1.178	-.0413	1.039	-.0398	.0446	1.0160	.0439	-.0491	1.039	-.0473
8	1.194	-.0276	1.053	-.0262	.0884	1.0365	.0858	-.0400	1.053	-.0380

C.G. 4			C.G. 5			C.G. 7		
$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$
-.1610	.8628	-.1868	-.1364	.8715	-.1590	-.1613	.8580	-.1880
-.1066	.8628	-.1236	-.0975	.8715	-.1141	-.1014	.8580	-.1182
-.0590	.9135	-.0646	-.0653	.9215	-.0709	-.0452	.9078	-.0498
-.0125	.9900	.01262	-.0316	1.000	-.0316	.0134	.9850	.0136
.0304	1.0160	.02970	.0022	1.027	.00214	.0671	1.0107	.0664
.0678	1.0365	.0658	.0308	1.041	.0296	.1196	1.0245	.1168

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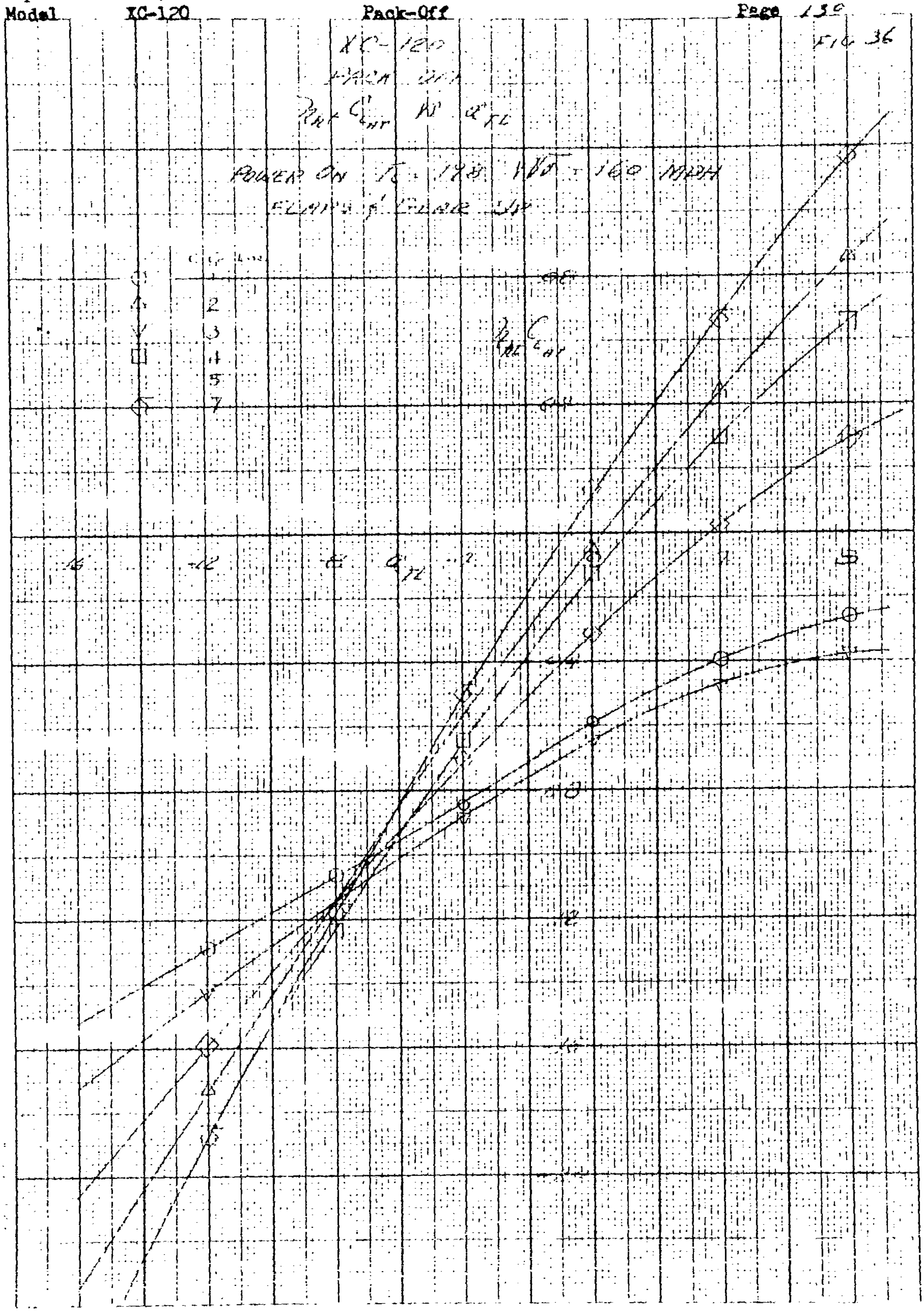

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PART II-D-1-b

BALANCING TAIL LIFT

FLAPS & GEAR UP—POWER ON

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

		C.G. 1			C.G. 2			C.G. 3		
α_{TL}	q_{Ht}/q	$C_{M_{A-T}}$	$K \cdot q_{Ht}/q$	$\eta_t C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{Ht}/q$	$\eta_t C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{Ht}/q$	$\eta_t C_{L_{HT}}$
-12	1.000	-.1180	.882	-.1338	-.1535	.8628	-.1780	-.1272	.882	-.1493
-8	1.000	-.0782	.882	-.1113	-.1036	.8628	-.1200	-.1026	.882	-.1162
-4	1.027	-.0830	.906	-.0916	-.0576	.8861	-.0650	-.0840	.906	-.0928
0	1.100	-.0644	.970	-.0664	-.0099	.9411	-.0104	-.0667	.970	-.0688
4	1.123	-.0457	.990	-.0462	.0385	.9689	.0398	-.0509	.990	-.0514
8	1.135	-.0319	1.001	-.0319	.0822	.9793	.0840	-.0416	1.001	-.0416

C.G. 4			C.G. 5			C.G. 7		
$C_{M_{A-T}}$	$K \cdot q_{Ht}/q$	$\eta_t C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{Ht}/q$	$\eta_t C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{Ht}/q$	$\eta_t C_{L_{HT}}$
-.1627	.8628	-.1886	-.1415	.8715	-.1625	-.1652	.8580	-.1925
-.1080	.8628	-.1251	-.1029	.8715	-.1181	-.1065	.8580	-.1240
-.0582	.8861	-.0665	-.0689	.8950	-.0770	-.0523	.8812	-.0594
-.0110	.9411	-.01065	-.0353	.9587	-.0368	.0074	.9438	.0079
.0335	.9689	.0346	-.0017	.9787	-.0017	.0623	.9635	.0647
.1725	.9793	.0741	+.0268	.9892	.0271	.1130	.9736	.1161

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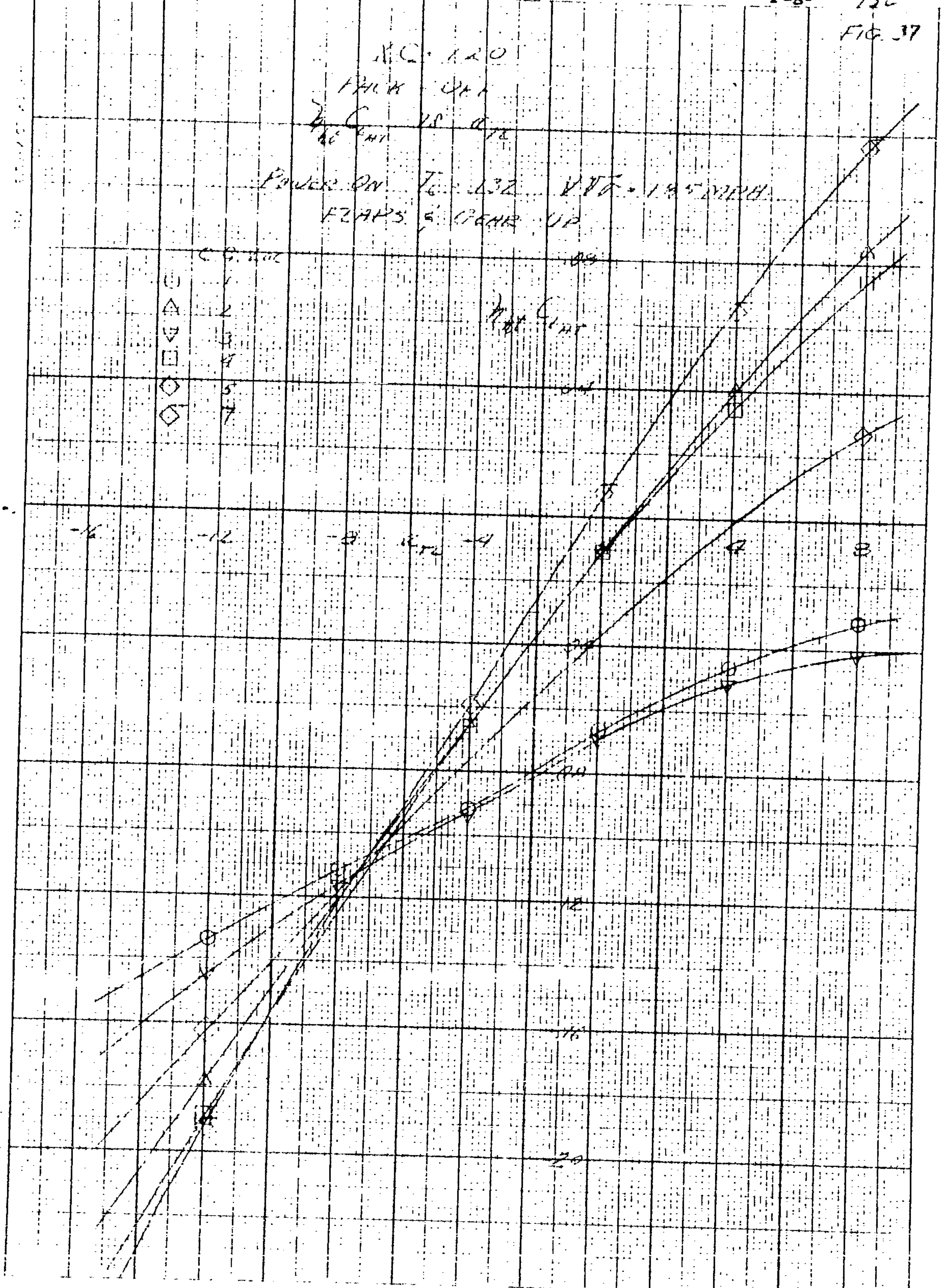
FIG. 37



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PART II-D-1-b

BALANCING TAIL LIFT
FLAPS & GEAR UP~POWER ON
$V\sqrt{\sigma}$ = 265 mph T_c = .047

		C.G. 1			C.G. 2			C.G. 3		
α_{TL}	$\frac{q_{Ht}}{q}$	$C_{M_{A-T}}$	$K\cdot\frac{q_{Ht}}{q}$	$\eta_{Ht} C_{LHT}$	$C_{M_{A-T}}$	$K\frac{q_{Ht}}{q}$	$\eta_{Ht} C_{LHT}$	$C_{M_{A-T}}$	$K\cdot\frac{q_{Ht}}{q}$	$\eta_{Ht} C_{LHT}$
-12	1.000	-.1254	.882	-.1416	-.1608	.8628	-.1866	-.1312	.882	-.1483
-8	1.000	-.1049	.882	-.1187	-.1107	.8628	-.1288	-.1057	.882	-.1190
-4	1.000	-.0891	.882	-.1010	-.0638	.8628	-.0734	-.0865	.882	-.0968
0	1.035	-.0731	.912	-.0800	-.0203	.892	-.0224	-.0701	.912	-.0768
4	1.046	-.0531	.926	-.0555	+.0305	.903	+.0338	-.0529	.926	-.0571
8	1.050	-.0370	.928	-.0398	+.0737	.906	+.0815	-.0432	.928	-.0466

C.G. 4			C.G. 5			C.G. 7		
$C_{M_{A-T}}$	$K\frac{q_{Ht}}{q}$	$\eta_{Ht} C_{LHT}$	$C_{M_{A-T}}$	$K\frac{q_{Ht}}{q}$	$\eta_{Ht} C_{LHT}$	$C_{M_{A-T}}$	$K\cdot\frac{q_{Ht}}{q}$	$\eta_{Ht} C_{LHT}$
-.1664	.8628	-.1930	-.1470	.8715	-.1685	-.1727	.8580	-.2012
-.1116	.8628	-.1290	-.1082	.8715	-.1240	-.1136	.8580	-.1324
-.0625	.8628	-.0725	-.0739	.8715	-.0846	-.0581	.8580	-.0678
-.0161	.892	-.0180	-.0426	.891	-.0430	-.0035	.8880	-.0039
+.0290	.903	+.0321	-.0070	.903	-.0078	.0518	.8975	.0577
+.0680	.906	+.0750	+.0217	.906	+.0240	.1040	.9009	.1155

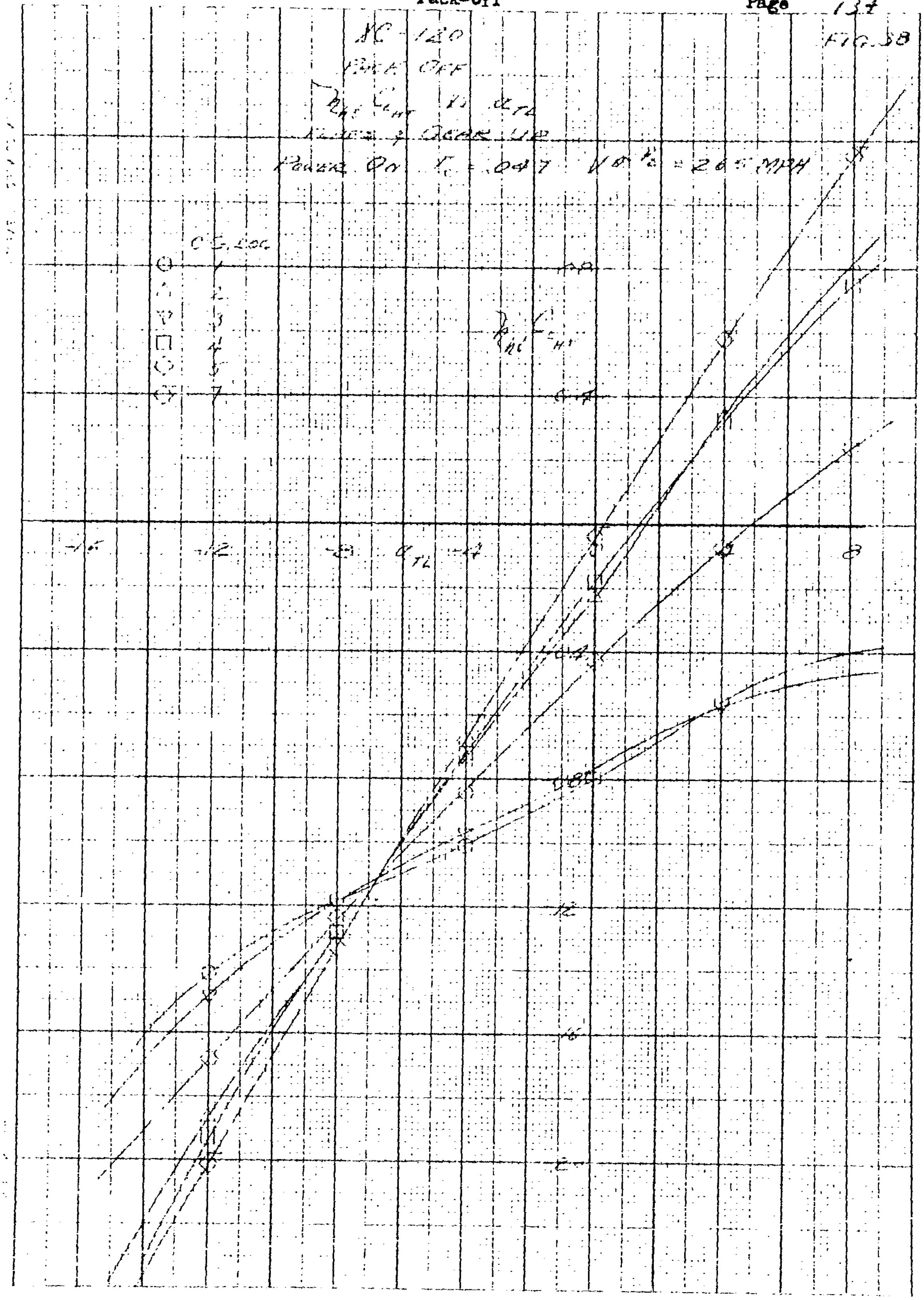


FIG. 38
XC-120
PACK OFF
C.G. LOC
POWER ON
MPH

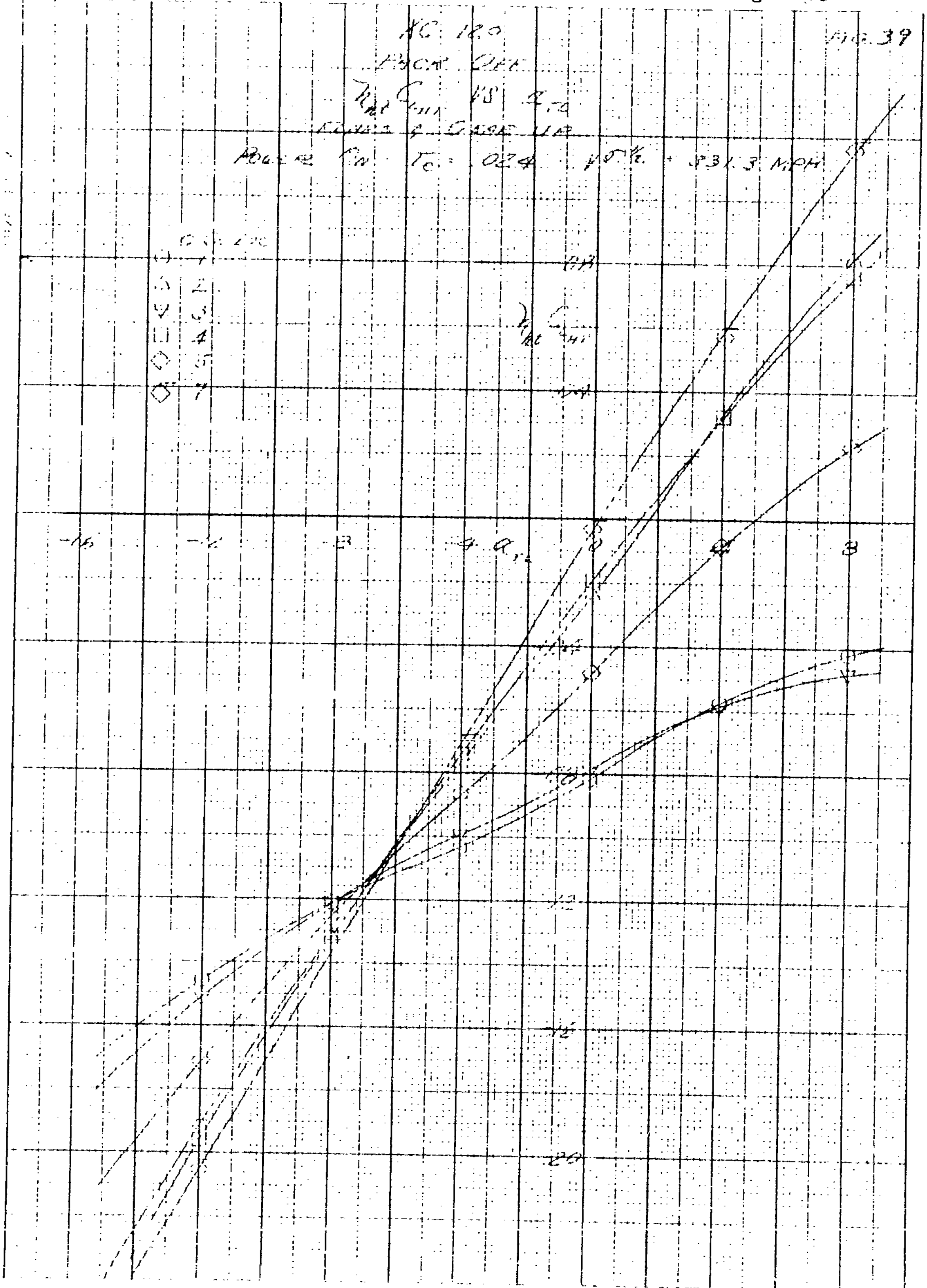
Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-D-1-b

BALANCING TAIL LIFT
FLAPS & GEAR UP~POWER ON
$V\sqrt{\sigma}$ = 331.3 mph T_c = .024

		C.G. 1			C.G. 2			C.G. 3		
α_T	q_{At}/q	$C_{M_{A-T}}$	$K \cdot q_{At}/q$	$\eta_{At} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{At}/q$	$\eta_{At} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{At}/q$	$\eta_{At} C_{L_{HT}}$
-12	1.000	-.1286	.882	-.1458	-.1641	.8628	-.1910	-.1335	.882	-.1510
-8	1.000	-.1070	.882	-.1210	-.1130	.8628	-.1311	-.1069	.882	-.1210
-4	1.000	-.0712	.882	-.1034	-.0662	.8628	-.0768	-.0818	.882	-.0922
0	1.018	-.0732	.896	-.0317	-.0208	.876	-.0238	-.0702	.896	-.0796
4	1.025	-.0522	.905	-.0578	+.0290	.885	+.0328	-.0522	.905	-.0563
8	1.026	-.0378	.906	-.0417	+.0725	.886	+.0819	-.0432	.906	-.0473

C.G. 4			C.G. 5			C.G. 7		
$C_{M_{A-T}}$	$K \cdot q_{At}/q$	$\eta_{At} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{At}/q$	$\eta_{At} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{At}/q$	$\eta_{At} C_{L_{HT}}$
-.1686	.8628	-.1960	-.1499	.8715	-.1715	-.1760	.8580	-.2051
-.1130	.8628	-.1310	-.1100	.8715	-.1256	-.1159	.8580	-.1351
-.0640	.8628	-.0743	-.0759	.8715	-.0868	-.0606	.8580	-.0707
-.0156	.876	-.0190	-.0428	.886	-.0484	-.0028	.8734	-.0032
+.0285	.885	+.0322	-.0077	.895	-.0086	.0502	.8795	.0571
+.0672	.886	+.0761	+.0208	.896	+.0232	.1021	.8803	.1160



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PART II-D-2

2. **Flaps and Gear Down**

a. Power Off

For flaps and gear down, power off

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

Values of K are the same as presented in Part II-D-1

PART II-D-2-a

BALANCING TAIL LIFT
FLAPS & GEAR DOWN~POWER OFF

		C.G. 1			C.G. 2			C.G. 3		
α_{TL}	q_{ht}/q	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$
-12	1.000	-.2466	.8820	-.2795	-.2532	.8628	-.2935	-.2398	.8820	-.2720
-8		-.2323		-.2640	-.2099		-.2433	-.2217		-.2515
-4		-.2226		-.2525	-.1706		-.1977	-.2096		-.2375
1		-.2097		-.2380	-.1289		-.1494	-.1989		-.2255
4		-.1969		-.2255	-.0913		-.1056	-.1920		-.2179
8		-.1920		-.2175	-.0555		-.0643	-.1913		-.2170

C.G. 4			C.G. 6			C.G. 7		
$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{L_{HT}}$
-.2464	.8628	-.2856	-.2443	.8715	-.2803	-.2578	.8580	-.3005
-.1993		-.2310	-.2158		-.2492	-.2057		-.2398
-.1587		-.1839	-.1942		-.2230	-.1580		-.1841
[illegible]		-.1379	-.1742		-.2002	-.1076		-.1254
-.0842		-.0976	-.1572		-.1804	-.0633		-.0737
-.0548		-.0635	-.1954		-.1670	-.0193		-.0225

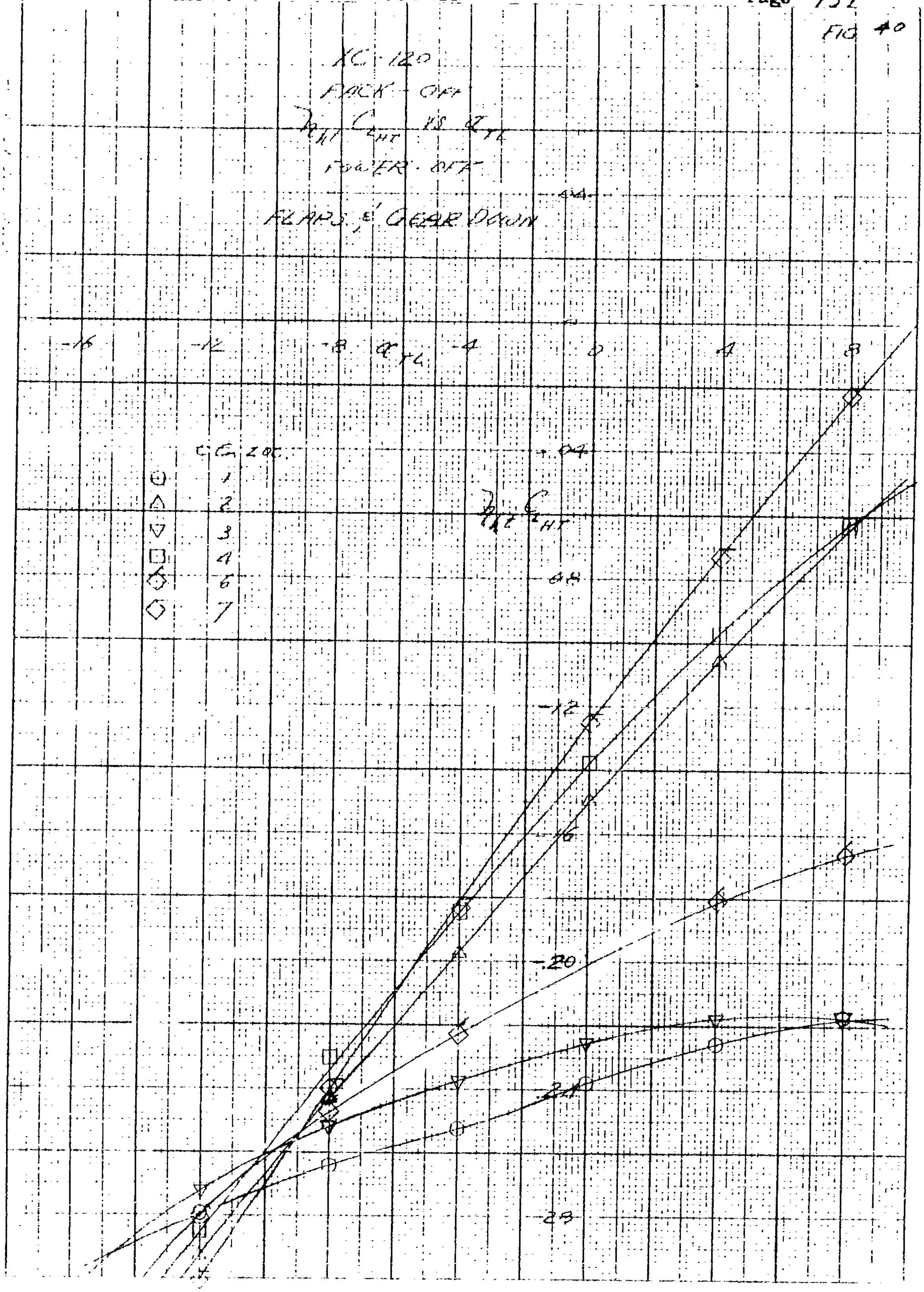


FIG 40
XC-120
PACK-OFF
POWER-OFF
FLAPS & GEAR DOWN
-16
-12
-8
-4
0
4
8
.04
C.G. %
1
2
3
4
6
7
-.12
-.16
-.20
-.24
-.28

PART II-D-2

b. Power On

For flaps and gear down, power on $\frac{q_{ht}}{q} = 1.00$

from page 321 reference (1)

Values of K are the same as presented in Part II-D-1

PART II-D-2-b

BALANCING TAIL LIFT
FLAPS & GEAR DOWN POWER ON
$V\sqrt{\sigma}$ = 100 mph T_c = .639

		C.G. 1			C.G. 2			C.G. 3		
α_{TL}	q_{ht}/q	C_{MA-T}	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$	C_{MA-T}	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$	C_{MA-T}	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$
-12	1.000	-.3823	.8820	-.4340	-.3789	.8628	-.4391	-.4002	.8820	-.4540
-8		-.3697		-.4190	-.3336		-.3866	-.3837		-.4350
-4		-.3594		-.4075	-.2867		-.3323	-.3711		-.4210
0		-.3435		-.3897	-.2350		-.2724	-.3579		-.4060
4		-.3287		-.3730	-.1864		-.2160	-.3471		-.3939
8		-.3162		-.3585	-.1370		-.1588	-.3411		-.3870

C.G. 4			C.G. 6			C.G. 7		
C_{MA-T}	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$	C_{MA-T}	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$	C_{MA-T}	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$
-.3967	.8628	-.4598	-.3995	.8715	-.4580	-.3802	.8580	-.4430
-.3477		-.4030	-.3716		-.4260	-.3250		-.3790
-.2998		-.3475	-.3472		-.3980	-.2644		-.3083
-.2510		-.2909	-.3229		-.3700	-.2051		-.2391
-.2046		-.2371	-.2993		-.3430	-.1478		-.1720
-.1627		-.1886	-.2800		-.3210	-.0893		-.1041

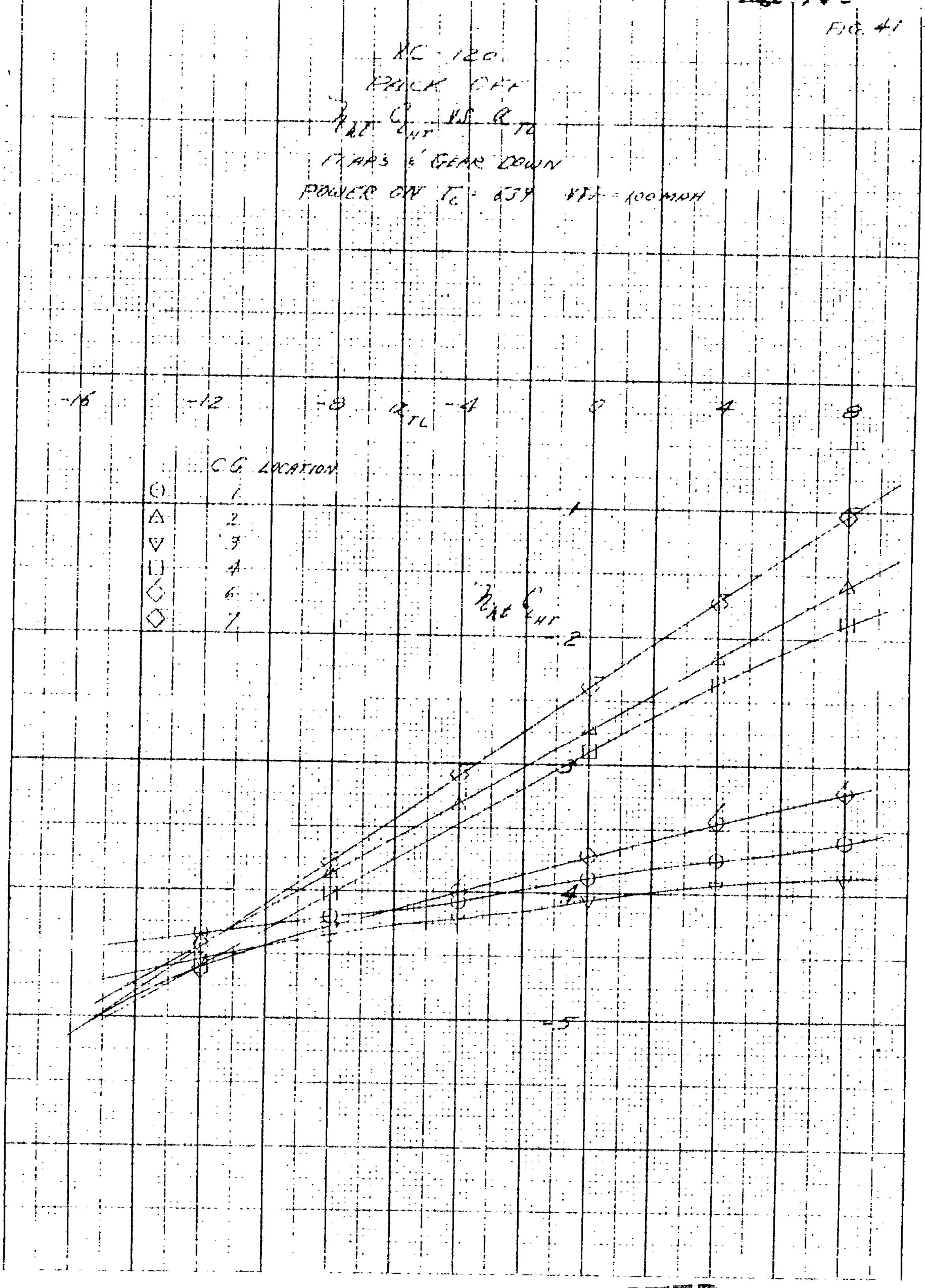

XC-120
PACK OFF
FLAPS & GEAR DOWN
-16
-12
-8
-4
0
4
8
C.G. LOCATION
1
2
3
4
6
7
-.1
-.2
-.4
-.5

FIG. 41

PART II-D-2-b

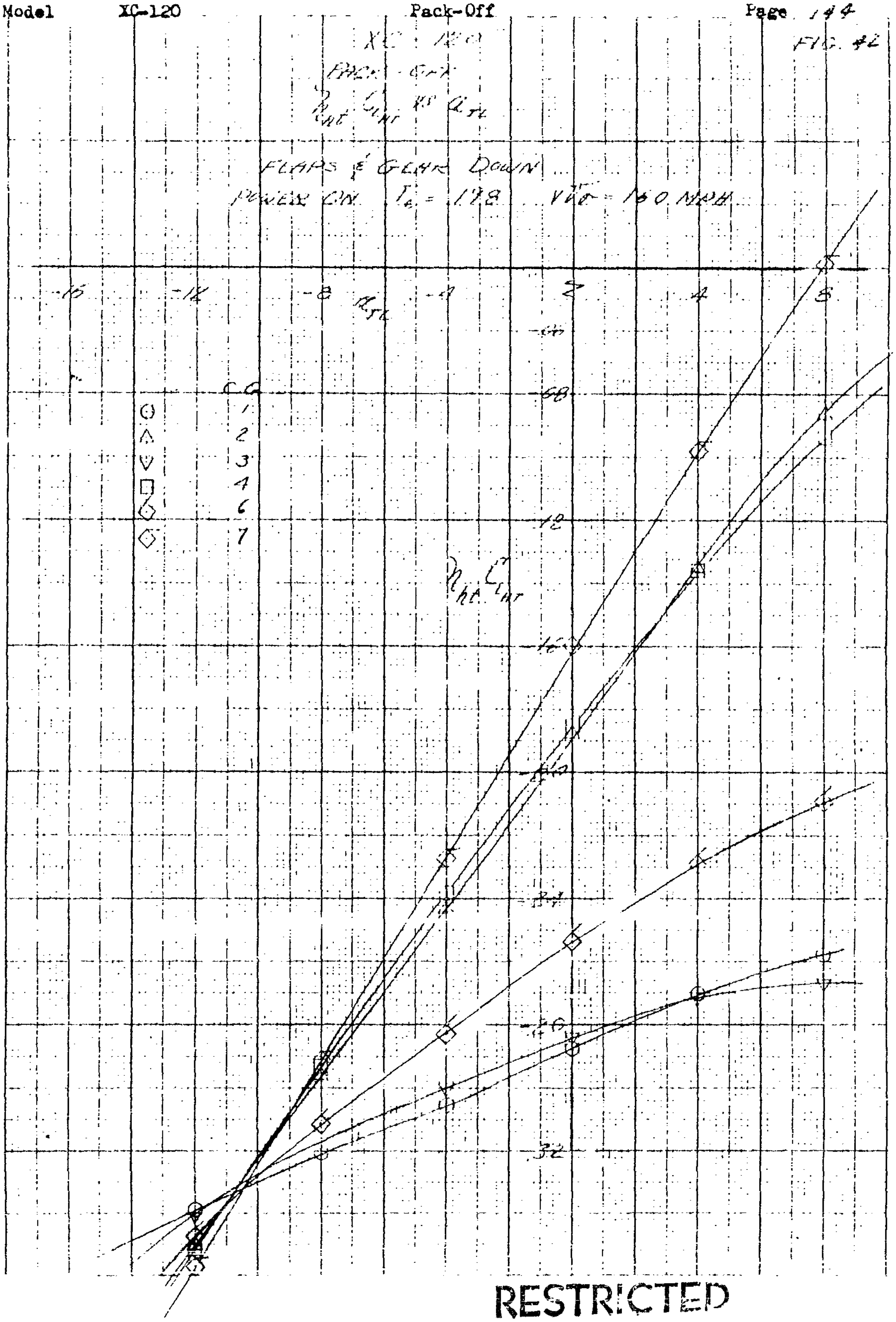
BALANCING TAIL LIFT

FLAPS & GEAR DOWN POWER ON

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

		C.G. 1			C.G. 2			C.G. 3		
α_{TL}	q_{ht}/q	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$
-12	1.000	-.2989	.8820	-.3390	-.3027	.8628	-.3508	-.3000	.8820	-.3401
-8		-.2825		-.3205	-.2555		-.2961	-.2797		-.3170
-4		-.2696		-.3055	-.2097		-.2430	-.2645		-.3000
0		-.2535		-.2875	-.1617		-.1876	-.2506		-.2840
4		-.2384		-.2705	-.1168		-.1354	-.2373		-.2715
8		-.2276		-.2560	-.0743		-.0861	-.2347		-.2665

C.G. 4			C.G. 6			C.G. 7		
$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$	$C_{M_{A-T}}$	$K \cdot q_{ht}/q$	$\eta_{ht} C_{LHT}$
-.3037	.8628	-.3520	-.3029	.8715	-.3478	-.3065	.8580	-.3575
-.2528		-.2930	-.2718		-.3118	-.2499		-.2913
-.2000		-.2368	-.2461		-.2824	-.1952		-.2278
-.1598		-.1852	-.2219		-.2540	-.1372		-.1600
-.1176		-.1363	-.1994		-.2288	-.0844		-.0983
-.0814		-.0943	-.1830		-.2100	-.0329		-.0384


XC-120
PACK-OFF
FIG. 42
FLAPS & GEAR DOWN
POWER ON
-16
-12
-8
-4
0
4
8
CG
1
2
3
4
6
7
-.04
-.08
-.12
-.16
-.20
-.24
-.28
-.32

RESTRICTED

PART II-E

E. COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

This section covers the determination of the normal force coefficients.

As the horizontal tail normal force coefficients will vary with the center of gravity location 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 coefficients 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.

1. Flaps and Gear Up

a. Power Off

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

$$C_{Z_A} = C_{Z_W} + \underline{C_{Z_N}} + \underline{C_{Z_B}} + \underline{C_{Z_{VT}}} + \underline{C_{Z_{HT}}}$$

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-c, reference (1) are all based on wing area no corrections are necessary.

The values of C_{Z_N} for the nacelle, power off, as obtained from Part II-A-5A-e are based on crew nacelle planform area and are corrected to wing area as follows:

$$\underline{C_{Z_N}} = C_{Z_N} \times \frac{P_n}{S_w}$$

$$\underline{C_{Z_N}} = C_{Z_N} \times \frac{450}{1447.25} = C_{Z_N} \times .3109$$

PART II-E-1-a (Cont.)

α_{TL} degrees	C_{Z_N}	$\underline{C_{Z_N}}$
-12	-.2668	-.0830
- 8	-.1956	-.06076
- 4	-.1421	-.0442
0	-.1100	-.0342
4	-.0710	-.02205
8	-.0026	-.00081

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

$$\underline{C_{Z_B}} = 2\left(C_{Z_B} \times \frac{P_b}{S_w}\right)$$

$$\underline{C_{Z_B}} = 2\left(C_{Z_B} \times \frac{312.5}{1447.25}\right) = C_{Z_B} \times .4318$$

α_{TL} degrees	C_{Z_B}	$\underline{C_{Z_B}}$
-12	-.0347	-.01495
- 8	-.0166	-.00711
- 4	-.0042	-.00181
0	0	0
4	.0042	.00181
8	.0166	.00711

$\underline{C_{Z_{VT}}} = 0$ as $C_{Z_{VT}} = 0$ from Part II-A-9-e reference (1)

PART II-E-1-a (Cont.)

The values of $C_{Z_{HT}}$ will of course vary with center of gravity location 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 of this report

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

substituting

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

$C_{M_{A-T}}$ is from Part II-C-1-a.

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR UP POWER OFF

						C.G. #1				C.G. 2
α_{TL}	C_{Z_W}	C_{Z_N}	C_{Z_B}	$C_{Z_{VT}}$	$C_{Z_{A-T}}$	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$
-12	-.32550	-.083	-.01495	0	-.42345	-.1135	.2709	-.03074	-.45419	-.1474
-8	.00398	-.06076	-.00711	0	-.06389	-.0978		-.02653	-.09042	-.1030
-4	.34820	-.0442	-.001811	0	+.30219	-.0872		-.02362	.27847	-.0621
0	.68500	-.0342	0	0	.6508	-.0733		-.01992	.63088	-.0214
4	1.02270	-.02205	.001811	0	1.00246	-.0594		-.01610	.98636	.0207
8	1.3618	-.000808	.00711	0	1.3681	-.0504		-.01366	1.35444	.0580

				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}
-.2769	-.0408	-.46425	-.1175	.2709	-.03183	-.45528	-.1514	.2769	-.04192	-.46537
	-.0285	-.09223	-.0968		-.02622	-.09011	-.1020		-.02824	-.09213
	-.01720	.28499	-.0828		-.02243	+.27976	-.0589		-.01631	.28588
	-.00593	.64407	-.0703		-.01904	.63176	-.0172		-.00476	.64604
	-.00573	.99673	-.0594		-.01609	.98637	.0208		.00575	1.00821
	+.01605	1.38415	-.0552		-.01494	1.35316	.0532		.01474	1.38284

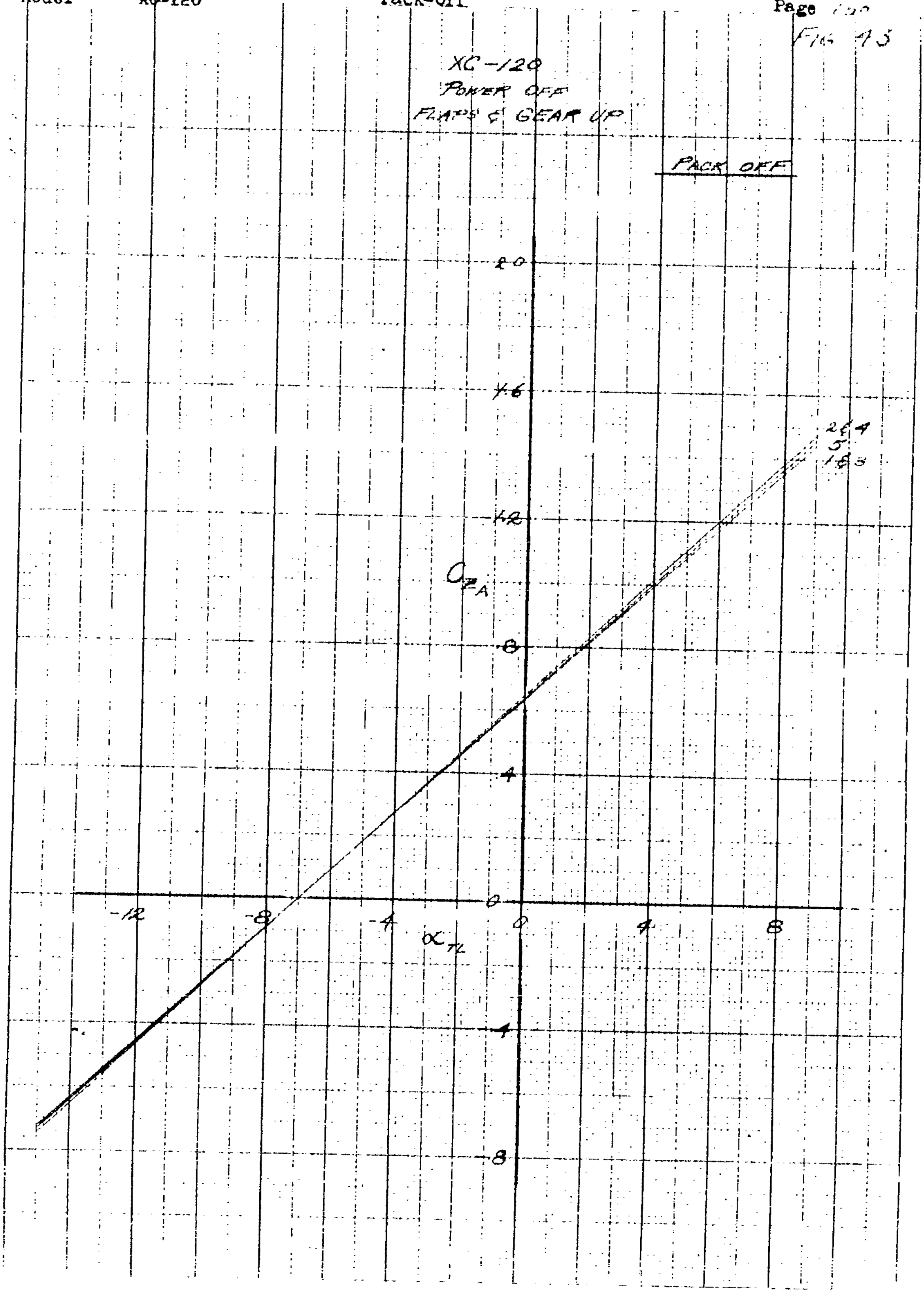
PART II-E-1-a

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR UP POWER OFF

C.G. #5				C.G. #7			
$C_{M_{A-T}}$	$\frac{MAC_W}{\ell_{ht}}$	$C_{Z_{NT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{\ell_{ht}}$	$C_{Z_{NT}}$	C_{Z_A}
-.1338	.2742	-.03669	-.46014	-.1590	.2780	-.0442	-.46765
-.1000		-.02472	-.08861	-.1458		-.0283	-.09219
-.0713		-.01955	.28264	-.0564		-.0157	+.28649
-.0432		-.01185	.63895	-.0049		-.00136	+.64944
-.0152		-.0042	.99826	.0414		.0115	+1.01396
.0075		.00205	1.37015	.0870		.0242	+1.3923

Fig 43



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PART II-E-1-b

b. Power On

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

$$C_{Z_A} = C_{Z_{A-T}\ \text{power off}} + 2\ \triangle C_{Z_{WI}} + 2\ C_{Z_P} + \underline{C_{Z_{HT}}}$$

where

$C_{Z_{A-T}\ \text{power off}}$ is the same as determined in Part II-E-1-a.

The values of $\triangle C_{Z_{WI}}$ due to propeller slipstream effects on the portion of the wing immersed in the propeller slipstream as obtained from Part II-B-1-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 other correction.

The values of C_{Z_P} 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

$$C_{Z_P} = \frac{Z_P}{q\ F_P} \qquad \text{Figure 60 reference (1)}$$

The values of $\underline{C_{Z_{HT}}}$ are determine the same as outlined in Part II-E-1-a with values of $C_{M_{A-T}}$ being obtained from Part II-C-1-b.

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MODEL XO-120 PREPARED BY CHECKED BY APPROVED BY

DATE 1 May 1949

REVISED

Subject:— BASIC FLIGHT CRITERIA - PACK OFF

PART II-E-1-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR UP 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_{Z_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	-.42345	-.0052	-.0541	-.0134	-.44205	-.0868	+.2709	-.0235	-.46555	-.1221
-8	-.00389	.0366	-.0372	-.0092	-.03649	-.0663		-.01795	-.05444	-.0692
-4	+.30219	.0786	-.0183	-.0045	.37629	-.0502		-.01360	.36269	-.0192
0	.6508	.1206	0	0	.77140	-.0309		-.00836	.76304	.0306
4	1.00246	.1652	.0183	.0045	1.17196	-.0117		-.00317	1.16879	.0920
8	1.3651	.2166	.0372	.0092	1.5879	-.0069		-.00187	1.58603	.1291

C.G. #2										
$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	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}
+.2769	-.0338	-.47585	-.1161	+.2709	-.0315	-.47355	-.1514	.2769	-.0420	-.48405
	-.01918	-.05567	-.0968		-.0246	-.06109	-.0938		-.0260	-.06249
	-.00531	.37098	-.0715		-.01938	.35691	-.0418		-.01156	.36473
	.00846	.77956	-.0537		-.01452	.75684	.0090		.002494	.77389
	.0227	1.19466	-.0371		-.01005	1.16191	.0567		.01570	1.18766
	.0358	1.6237	-.0269		-.00229	1.5806	.0992		.02747	1.61537

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Subject: BASIC FLIGHT CRITERIA - PACK OFF			DATE 1 May 1949 REVISED	

PART II-E-1-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS
FLAPS & GEAR UP POWER ON
$V\sqrt{\sigma} = 100$ mph $T_c = .639$

C.G. 5				C.G. 7			
$C_{M_{A-T}}$	$\frac{MAC_W}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
-.1199	+.2742	-.0329	-.47495	-.1342	.2780	-.0375	-.47955
-.0795		-.0218	-.05829	-.0711		-.0196	-.05609
-.0434		-.0119	.36439	-.0084		-.00235	+.37394
-.0080		-.002197	.769203	.0501		+.01395	.78535
+.0277		.0076	1.17956	.1069		+.0297	1.20166
.0584		.016	1.60390	.1630		+.0453	1.6332

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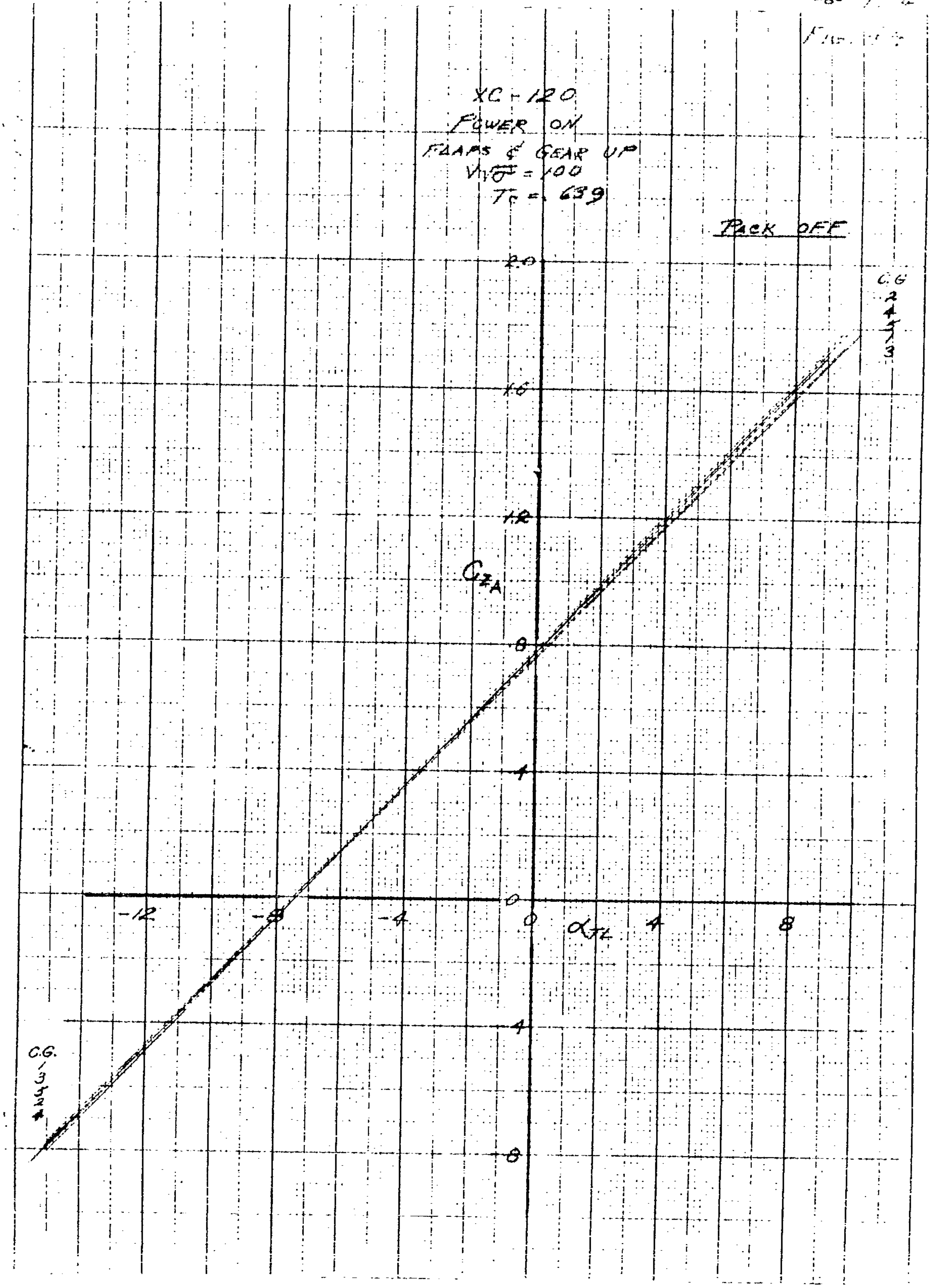
XC-120
POWER ON
FLAPS & GEAR UP
$V\sqrt{\sigma} = 100$
$T_c = .639$
PACK OFF
C.G
C_{Z_A}
α_{FL}
20
16
12
8
4
0
-4
-8
-12
-8
-4
0
4
8
C.G.

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COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR UP POWER ON

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

						C.G. #1				C.G. #2
α_{TL}	$C_{Z_{A-T}}$ POWER OFF.	$2\Delta C_{Z_{W1}}$	C_{Z_P}	$2 C_{Z_P}$	$C_{Z_{A-T}}$ POWER ON	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$
-12	-.42345	-.0094	-.0493	-.0122	-.44505	-.1135	+.2709	-.03073	-.47578	-.1192
-8	-.06389	.0094	-.0335	-.0083	-.06279	-.0936		-.02540	-.08819	-.0986
-4	[illegible]	.0284	-.0170	-.0042	.32639	-.0785		-.02123	.30516	-.0511
0	.6565	.0476	0	0	.69840	-.0600		-.01625	.68215	-.0043
4	1.02246	.0674	.0170	.0042	1.07406	-.0413		-.0111	1.06287	+.0446
8	1.3681	.0876	.0335	.0083	1.4640	-.0276		-.00744	1.45653	.0586

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}
+.2769	-.04140	-.48645	-.1254	.2709	-.0340	-.47905	-.1610	.2769	-.04452	-.48957
	-.02732	-.09011	-.1006		-.0272	-.08999	-.1066		-.02950	-.09229
	-.01415	.31224	-.0822		-.02227	.30412	-.0596		-.01633	.31006
	-.01190	.69721	-.0649		-.0176	.6808	-.0125		-.00346	.69494
	-.01235	1.68644	-.0491		-.0133	1.06076	.0304		.00841	1.08247
	-.02445	1.45645	-.0400		-.01082	1.45318	.0678		.01878	1.46278

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PART II-E-1-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR UP POWER ON

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

C.G. 5				C.G. 7			
$C_{M_{A-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_B}	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.1384	.2742	-.0380	-.48305	-.1613	.2780	-.0446	-.48985
-.0995		-.0273	-.09009	-.1014		-.0282	-.09099
-.0553		-.0179	.30549	-.0452		-.0126	+.31379
-.0316		-.00666	.68974	.0134		+.00374	+.70214
.0022		.00064	1.074664	.0671		+.0187	+1.09276
.0308		.00845	1.47245	.1196		+.0332	+1.4972

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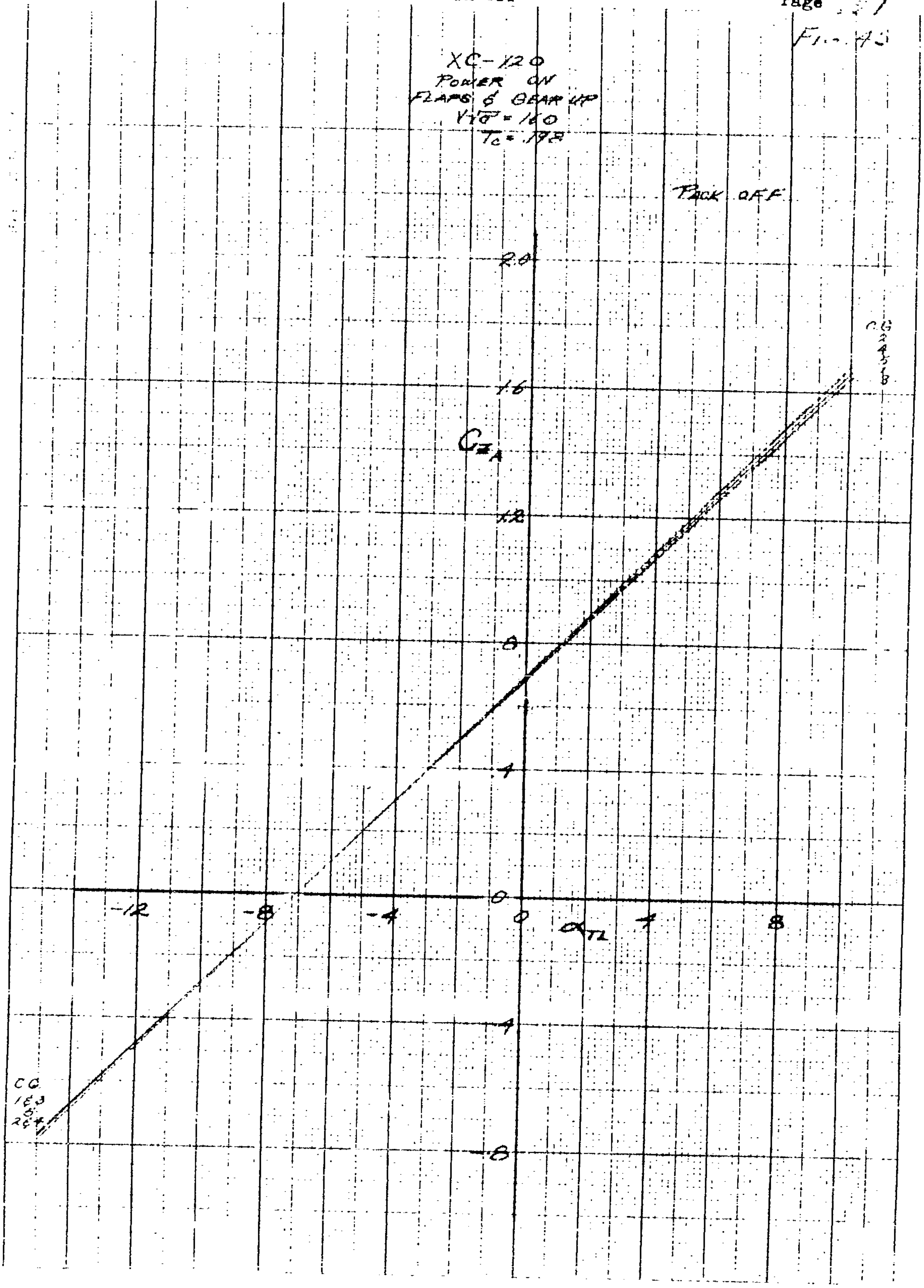

XC-120
POWER ON
FLAPS & GEAR UP
$V\sqrt{\sigma} = 160$
$T_c = .192$
PACK OFF
C_{Z_A}
α_{TL}
C.G.
C.G.

PART II-E-1-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR UP POWER ON

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

						C.G. #1				C.G. #2
α_{TL}	$C_{Z_{A-T}}$ POWER OFF	$2\Delta C_{Z_{W1}}$	C_{Z_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	-.42345	-.0074	-.0505	-.0125	-.44335	-.1160	+.2709	-.0320	-.473535	-.1535
-8	-.06389	.0060	-.0350	-.0086	-.06649	-.0982		-.0266	-.09309	-.1036
-4	.30219	.0200	-.0175	-.0043	+.31789	-.0830		-.0225	.29539	-.0576
0	.6508	.0330	0	0	.68380	-.0644		-.0174	.66640	-.0099
4	1.00246	.0472	.0175	.0043	1.05396	-.0457		-.0124	1.04156	.0385
8	1.3681	.0624	.0350	.0086	1.4391	-.0319		-.00865	1.43045	.0822

C.G. #2 (CONT.)			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}
+.2769	-.0425	-.48535	-.1272	+.2709	-.0345	-.47785	-.1627	+.2769	-.0450	-.48835
	-.0287	-.09519	-.1026		-.0278	-.09429	-.1080		-.0299	-.09639
	-.01598	.30191	-.0840		-.02279	.29510	-.0586		-.01629	.30160
	-.00274	.68106	-.0667		-.01809	.66571	-.0110		-.005042	.68075
	.01066	1.06462	-.0509		-.01380	1.04016	.0335		+.00728	[illegible]
	.02280	1.46190	-.0416		-.01130	1.4278	.0735		[illegible]	1.45918

FA-806-23

RESTRICTED

Subject:— BASIC FLIGHT CRITERIA - PACK OFF

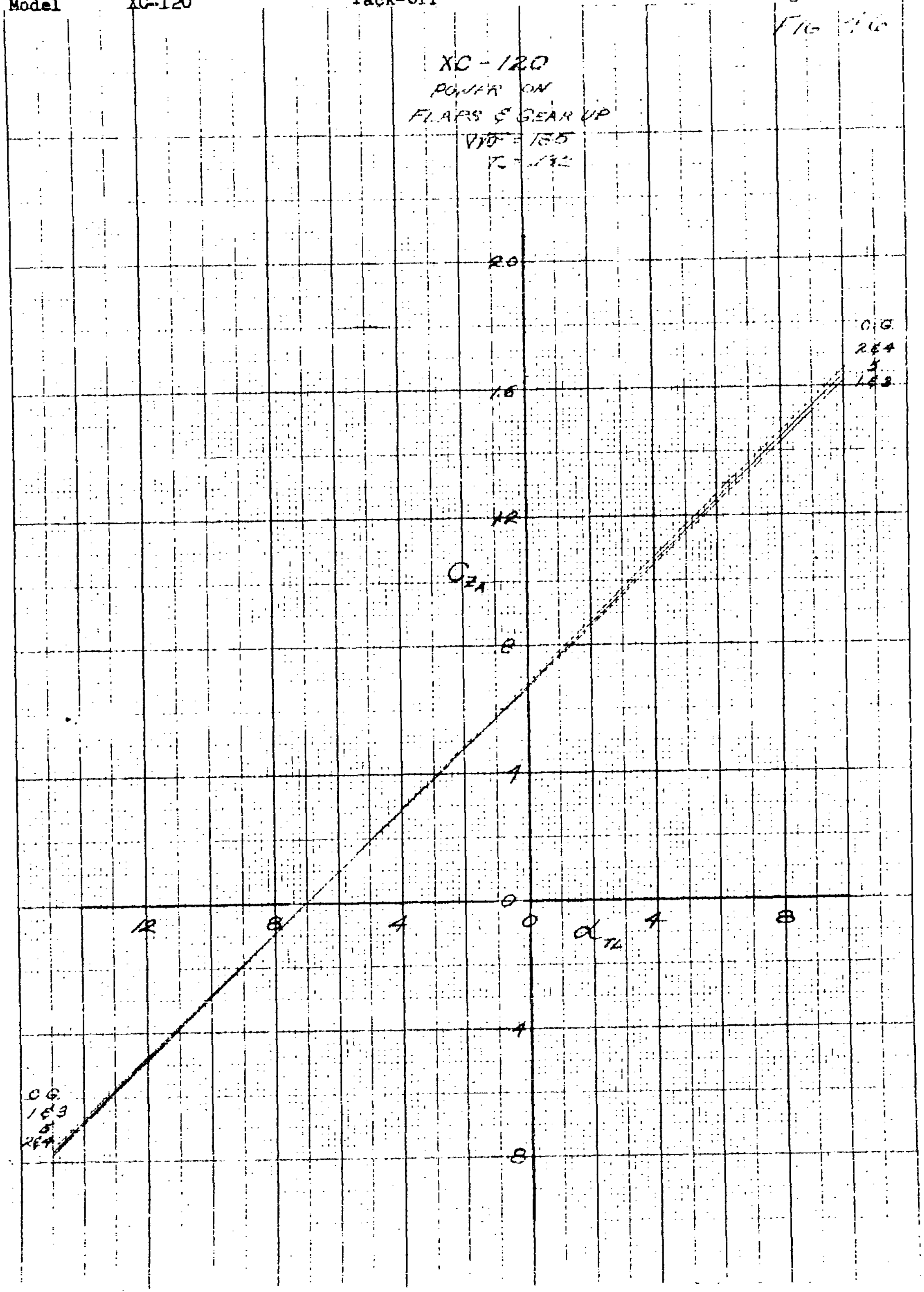
PART II-E-1-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR UP POWER ON

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

C.G. #5				C.G. #7			
$C_{M_{0-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.14.5	+.2742	-.0392	-.48255	-.1652	.2780	-.0459	-.48925
-.1029		-.02847	-.09496	-.1037		-.0381	-.10459
-.0689		-.01908	.29881	-.0523		-.0146	+.30329
-.0353		-.00977	.67403	.0074		+.00207	+.68587
-.0017		-.000470	1.05349	.0623		+.0174	+1.07136
+.0268		.00742	1.44652	.1130		+.0314	+1.4705


XC - 120
POWER ON
FLAPS & GEAR UP
V_{TO} = 155
20
C.G.
26.4
&
16.3
16
12
C_{Z_A}
8
4
0
12
8
4
0
α_{TL}
4
8
4
C.G.
16.3
&
26.4
8

REPORT NO. R107-014 **FAIRCHILD AIRCRAFT DIVISION** OF FAIRCHILD ENGINE & AIRPLANE CORPORATION — PAGES | PAGE 161

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

DATE 1 May 1949

REVISED

Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-E-1-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR UP POWER ON

$V\sqrt{\sigma}$ = 265 mph T_c = .047

						C.G. #1				C.G. #2
α_{TL}	$C_{Z_{A-T}}$ POWER OFF	$2\Delta C_{Z_{Wl}}$	C_{Z_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	-.42345	-.004	-.058	-.0143	-.44175	-.1254	+.2709	-.0340	-.47575	-.1608
-8	-.06389	+.001	-.039	-.0096	-.07249	-.1049		-.0284	-.10089	-.1107
-4	.30219	.006	-.019	-.00468	.303510	-.0891		-.02416	.27935	-.0638
0	.6508	.012	0		.6628	-.0731		-.01980	.64300	-.0203
4	1.00246	.018	.019	.00468	1.02514	-.0513		-.01389	1.01125	+.0305
8	1.3681	.025	.039	.0096	1.4027	-.0320		-.01001	1.39269	+.0739

C.G. #2 (CONT.)				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}
+.2769	-.0445	.48625	-.1312	+.2709	-.0356	-.47735	-.1664	+.2769	-.04602	-.48777
	-.03062	.10311	-.1057		-.02861	-.10110	-.1116		-.03087	-.10336
	-.01768	.28583	-.0865		-.0234	.28011	-.0625		-.01730	.28621
	-.00562	.65718	-.0701		-.0190	.64380	-.0161		-.00445	.65834
	.00845	1.03359	-.0529		-.0143	1.01084	.0290		.00803	1.03317
	.02043	1.42313	-.0432		-.01170	1.3910	.0680		.01882	1.42152

RESTRICTED

Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-E-1-b

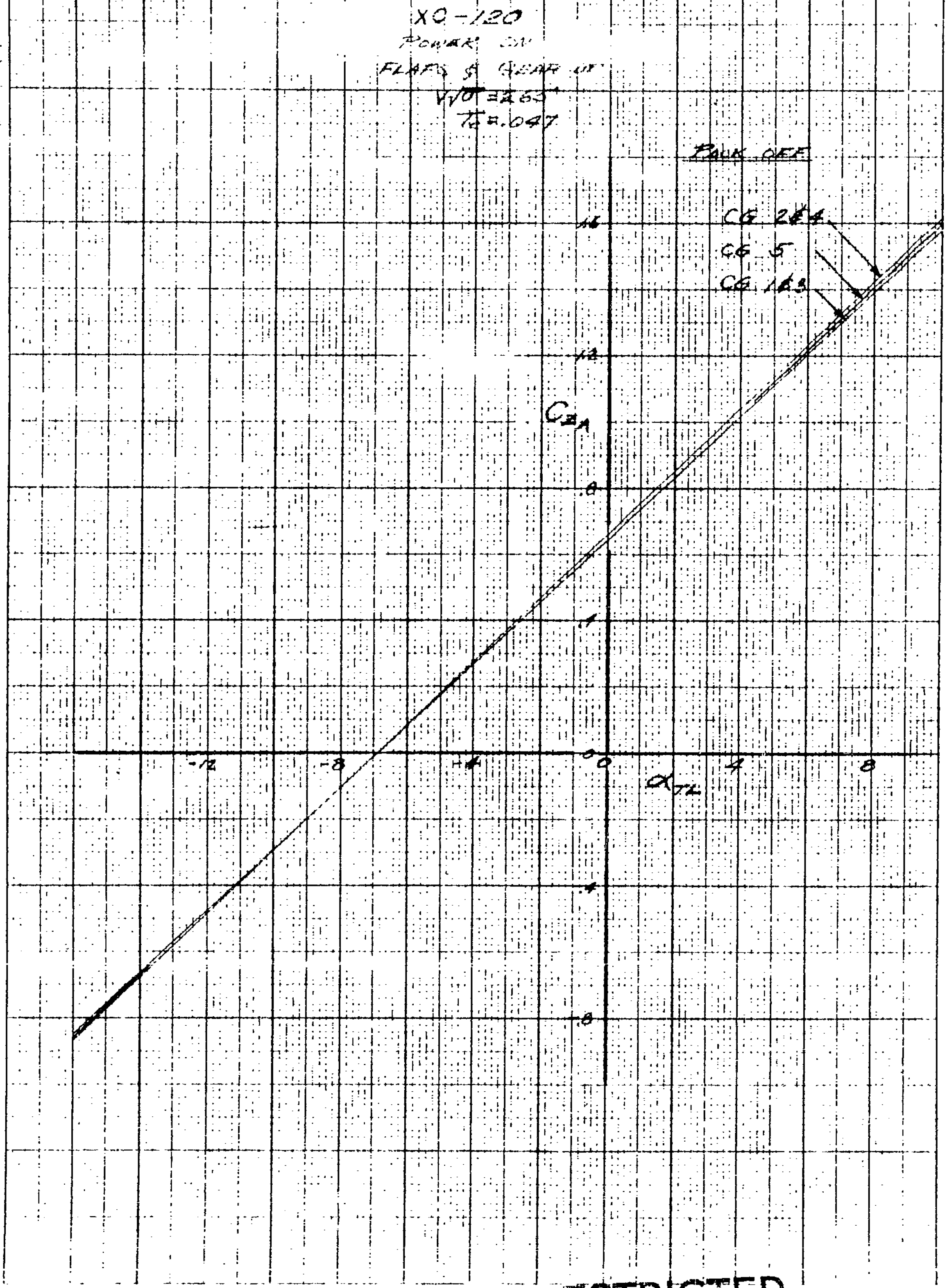
COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR UP   POWER ON

$V\sqrt{\sigma}$ = 265 mph T_c = .047

C.G. #5				C.G. #7			
$C_{M_{A-T}}$	$\frac{MAC_W}{\ell_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{\ell_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.1470	+.2742	-.04028	-.48205	-.1727	.2780	-.0480	-.48975
-.1082		-.0297	-.10219	-.1136		-.0315	-.10399
-.0739		-.02026	.28325	-.0581		-.062	+.28731
-.0426		-.01169	.65111	-.0035		-.00097	+.66183
-.0070		-.00192	1.02322	.0518		+.0144	+1.03954
+.0217		.00595	1.40865	.1040		+.0290	+1.4317

FIG. 47


XC-120
POWER ON
FLAPS & GEAR UP
PACK OFF
CG 2&4
CG 5
CG 1&3
.16
.12
.08
.04
0
-.04
-.08
-12
-8
-4
0
4
8

PART II-E-1-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR UP POWER ON

$V\sqrt{\sigma}$ = 331.3 mph T_c = .024

						C.G. #1				C.G. #2
α_{TL}	$C_{Z_{A-T}}$ POWER OFF	$2\Delta C_{Z_{W1}}$	C_{Z_P}	$\underline{2C_{Z_P}}$	$C_{Z_{A-T}}$ POWER ON	$C_{m_{A-T}}$	$\frac{MAC_W}{l_{ht}}$	$\underline{C_{Z_{HT}}}$	C_{Z_A}	$C_{m_{A-T}}$
-12	-.42345	-.002	-.066	-.0163	-.4418	-.1286	+.2709	-.0348	-.4766	-.1641
-8	-.06389	+.0005	-.044	-.0108	-.07419	-.1070		-.0290	-.1032	-.1130
-4	.30219	+.004	-.022	-.0054	.30079	-.0912		-.0247	.2761	-.0662
0	.6508	.006	0	0	.6568	-.0732		-.0198	.6370	-.0208
4	1.00246	+.010	.022	.0054	1.01786	-.0532		-.0141	1.0038	.0290
8	1.3681	+.014	.044	.0108	1.39290	-.0378		-.0102	1.3827	.0725

C.G. #2				C.G. #3				C.G. #4		
$\frac{MAC_W}{l_{ht}}$	$\underline{C_{Z_{HT}}}$	C_{Z_A}	$C_{m_{A-T}}$	$\frac{MAC_W}{l_{ht}}$	$\underline{C_{Z_{HT}}}$	C_{Z_A}	$C_{m_{A-T}}$	$\frac{MAC_W}{l_{ht}}$	$\underline{C_{Z_{HT}}}$	C_{Z_A}
+.2769	-.0454	-.4872	-.1335	+.2709	-.0362	-.4780	-.1688	+.2769	-.0467	-.4885
	-.0313	-.1055	-.1069		-.0290	-.1032	-.1130		-.0313	-.1053
	-.0183	.2825	-.0878		-.0238	.2770	-.0640		-.0177	.2831
	-.0058	.6510	-.0702		-.0190	.6378	-.0166		-.0046	.6522
	.0080	1.0259	-.0529		-.0143	1.0036	+.0285		.0079	1.0258
	.0201	1.4130	-.0432		-.0117	1.3812	.0672		.0186	1.4115

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

DATE 1 May 1949

REVISED

Subject:— BASIC FLIGHT CRITERIA - PACK OFF

PART II-E-1-b

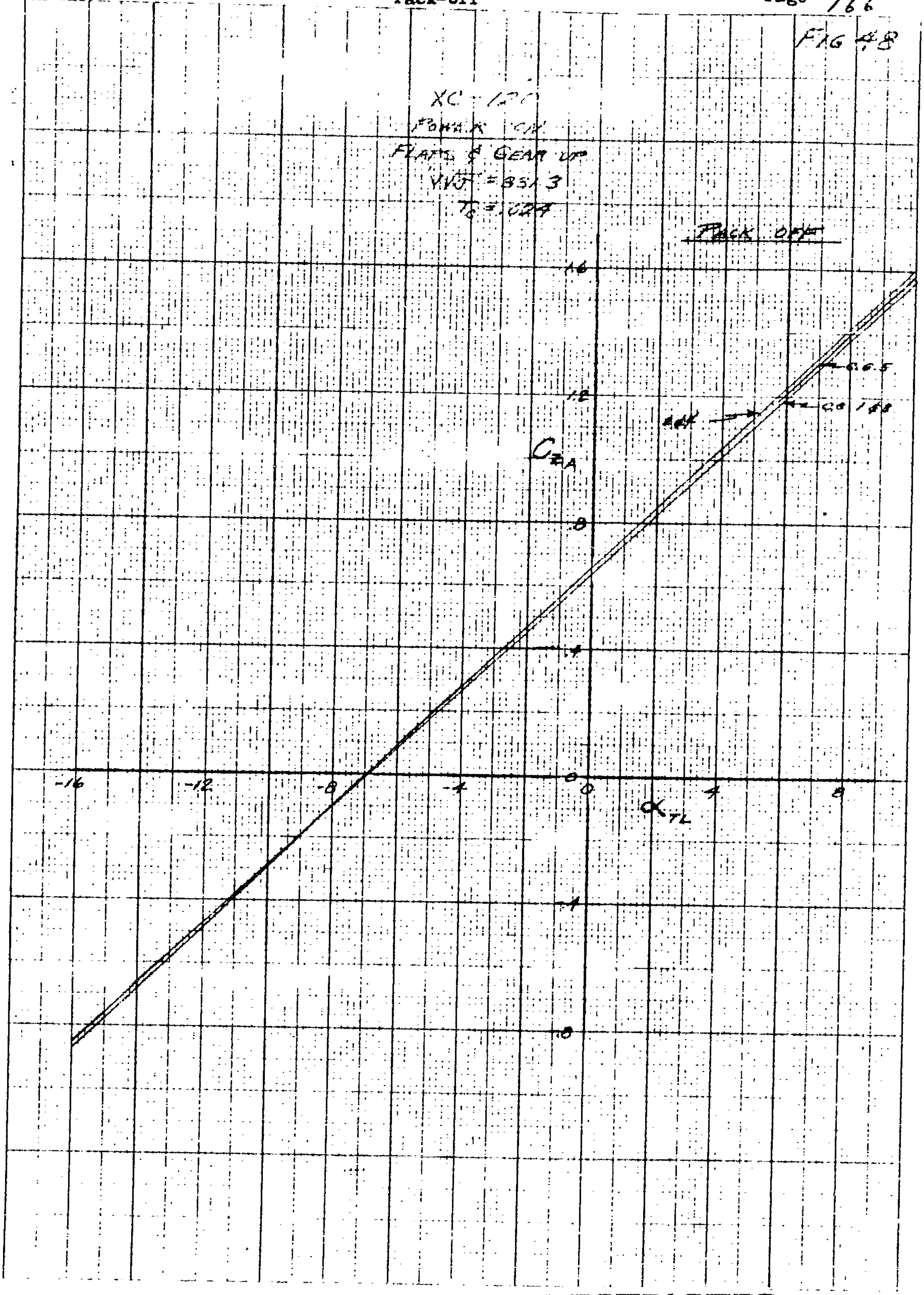
COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS
FLAPS & GEAR UP POWER ON .
$V\sqrt{\sigma}$ = 331.3 mph T_c = .024

C.G. #5				C.G. #7			
$C_{m_{A-T}}$	$\frac{MAC_w}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{m_{A-T}}$	$\frac{MAC_w}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.1499	+.2742	-.0411	-.4829	-.1760	.2780	-.0488	-.4906
-.1100		-.0302	-.1044	-.1159		-.0322	-.10639
-.0759		-.0208	.2800	-.0606		-.0168	+.28399
-.0428		-.0117	.6451	-.0028		-.00077	+.65603
-.0077		-.0021	1.0158	.0502		+.0144	+1.03226
+.0208		.0057	1.3986	.1021		+.0284	+1.4213

RESTRICTED

Pack-Off

FIG 48



PART II-E-2

2. **Flaps and Gear Down**

a. Power Off

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

$$C_{Z_A} = C_{Z_{WF}} + C_{Z_N} + C_{Z_B} + C_{Z_{VT}} + C_{Z_{L.G.}} + C_{Z_{HT}}$$

Where the aerodynamic force coefficients of the component parts are all based on the total 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 C_{Z_N}, C_{Z_B} and $C_{Z_{VT}}$ are the same as shown in Part II-E-1-a of this report.

Values of $C_{Z_{HT}}$ are obtained as outlined in Part II-E-1-a of this report.

Subject: – BASIC FLIGHT CRITERIA – PACK OFF

PART II-E-2-a

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR DOWN POWER OFF

							C.G. #1			
α_{TL}	$C_{Z_{WF}}$	C_{Z_N}	C_{Z_B}	$C_{Z_{VT}}$	$C_{Z_{LG}}$	$C_{Z_{A-T}}$	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{t}}$	$C_{Z_{HT}}$	C_{Z_A}
-12	.0046	-.083	-.01495	0	.0136	-.07975	-.2466	+.2709	-.0668	-.1466
-8	.3403	-.06076	-.00711	0	.0091	.28153	-.2323		-.0629	.2186
-4	.6806	-.0442	-.00181	0	.0046	.6392	-.2226		-.0603	.5789
0	1.0230	-.0342	0	0	0	.9888	-.2097		-.0568	.9326
4	1.3645	-.02205	.00181	0	.0046	1.3709	-.1989		-.0539	1.3170
8	1.7036	-.00081	.00711	0	.0091	1.7190	-.1920		-.0520	1.6670

C.G. #2				C.G. #2			
$C_{M_{A-T}}$	$\frac{MAC_W}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{t}}$	$C_{Z_{HT}}$	C_{Z_A}
-.2532	+.2769	-.0701	-.1499	-.2398	+.2709	-.06496	-.1447
-.2099		-.05812	.2234	-.2217		-.06006	.2215
-.1706		-.04723	.5920	-.2096		-.05678	.5824
-.1289		-.03565	.9531	-.1989		-.05388	.9349
-.0913		-.02528	1.3456	-.1920		-.0520	1.3189
-.0555		-.01536	1.7036	-.1913		-.05182	1.6672

PART II-E-2-a

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR DOWN POWER OFF

C.G. # 4				C.G. #6			
$C_{M_{A-T}}$	$\frac{MAC_w}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_w}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
-.2464	.2767	-.06822	-.14797	-.2443	+.2729	-.0666	-.1464
-.1993		-.0552	.1633	-.2158		-.0568	.2227
-.1587		-.0439	.5953	-.1942		-.0530	.5862
-.1189		-.0329	.9559	-.1742		-.0475	.9413
-.0842		-.0233	1.3476	-.1572		-.0429	1.3280
-.0548		-.0152	1.7038	-.1454		-.0397	1.6793

C.G #7			
$C_{M_{A-T}}$	$\frac{MAW_w}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
-.2578	.2780	-.0718	-.15055
-.2057		-.0530	+.22853
-.1560		-.0406	+.5986
-.1076		-.0277	+.9611
-.0633		-.0163	+1.3546
-.0193		-.0050	+1.7140

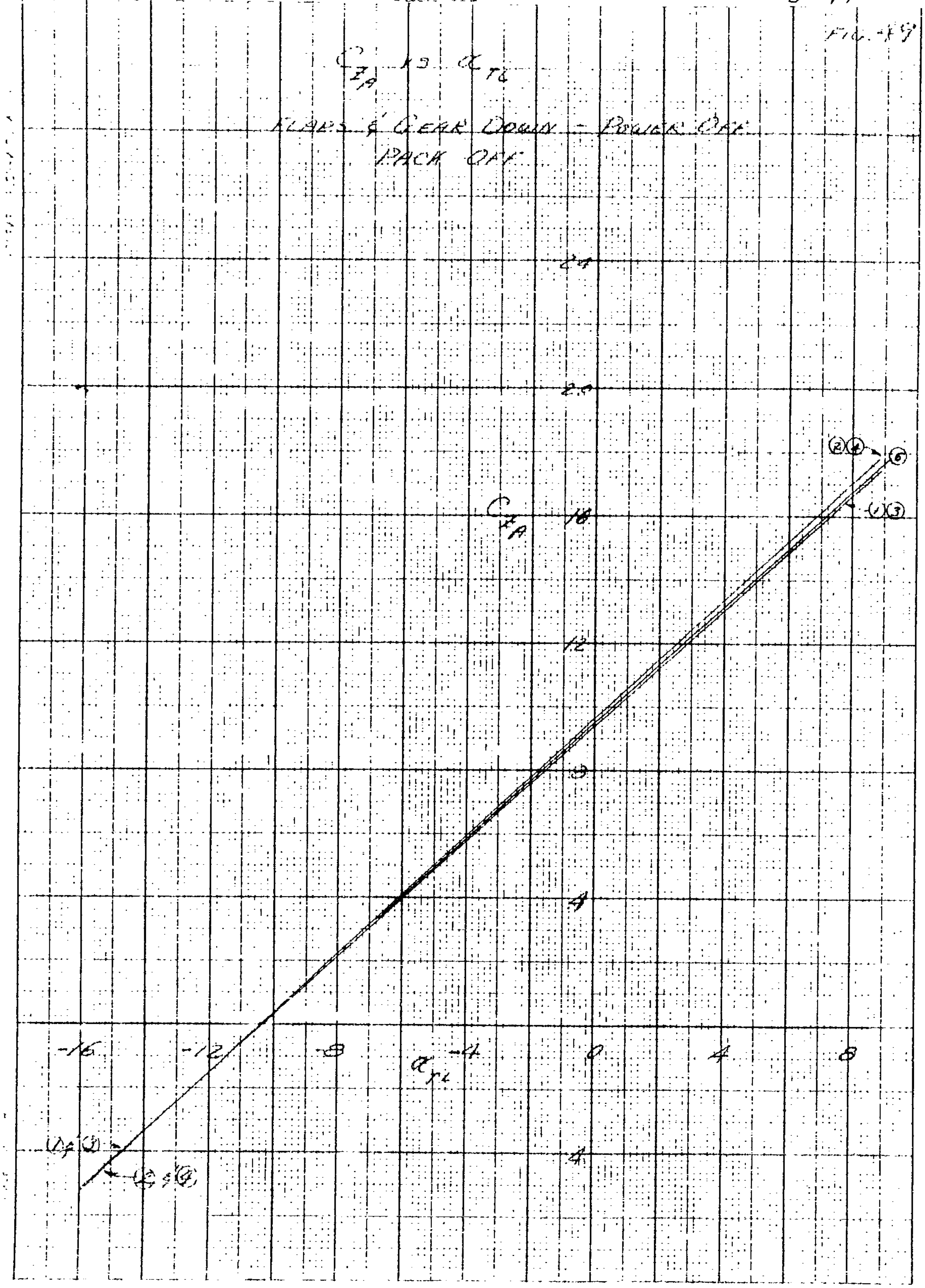


Fig. 49
C_{Z_A} vs α_{TL}
FLAPS & GEAR DOWN - POWER OFF
PACK OFF
C_{Z_A}
α_{TL}
24
20
16
12
8
4
-4
-16
-12
-8
-4
0
4
8
(2)(4)
(6)
(1)(3)
(1) & (3)
(2) & (4)

PART II-E-2-b

b. Power On

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

$$C_{Z_A} = C_{Z_{A-T\ \text{power off}}} + 2 \triangle C_{Z_{WFI}} + 2\ C_{Z_P} + C_{Z_{HT}}$$

where

$C_{Z_{A-T\ \text{power off}}}$ is the same as determined in Part II-E-1-a.

The values of $\triangle C_{Z_{WFI}}$ due to the propeller slipstream effects on the portion of the flapped wing immersed in the propeller slipstream as obtained from figure 56 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 corrections.

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

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

PART II-E-2-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS
FLAPS & GEAR DOWN POWER ON
$V\sqrt{\sigma}$ = 100 mph T_c = .639

α_{TL}	$C_{Z_{A-T}}$ POWER OFF	$2\Delta C_{Z_{HFI}}$	$2C_{Z_P}$	$C_{Z_{A-T}}$ POWER ON	C.G. 1 $C_{M_{A-T}}$	C.G. 1 $\frac{MAC_W}{\ell_{HT}}$	C.G. 1 $C_{Z_{HT}}$	C.G. 1 C_{Z_A}
-12	-.07975	.1020	.0134	.00885	-.3823	-.2709	-.10356	-.09471
-8	.28153	.1804	-.0092	.45273	-.3697		-.10015	.3526
-4	.6392	.2630	-.0045	.8983	-.3594		-.09736	.8009
0	.9888	.3480	0	1.3368	-.3435		-.09305	1.2438
4	1.3709	.4300	.0045	1.8054	-.3287		-.08904	1.7164
8	1.7196	.5150	.0092	2.2432	-.3162		-.08565	2.1576

C.G. 2 $C_{M_{A-T}}$	C.G. 2 $\frac{MAC_W}{\ell_{HT}}$	C.G. 2 $C_{Z_{HT}}$	C.G. 2 C_{Z_A}	C.G. 3 $C_{M_{A-T}}$	C.G. 3 $\frac{MAC_W}{\ell_{HT}}$	C.G. 3 $C_{Z_{HT}}$	C.G. 3 C_{Z_A}
-.3789	+.2769	-.10491	-.09606	-.4002	+.2709	-.10841	-.09956
-.3336		-.09237	.3604	-.3637		-.10394	.3488
-.2867		-.07938	.8189	-.3711		-.10053	.7978
-.2356		-.06507	1.2717	-.3579		-.09695	1.2399
-.1864		-.05161	1.7538	-.3471		-.09403	1.7114
-.1370		-.03793	2.2053	-.3411		-.09240	2.1508

RESTRICTED

Subject: BASIC FLIGHT CRITERIA - PACK OFF

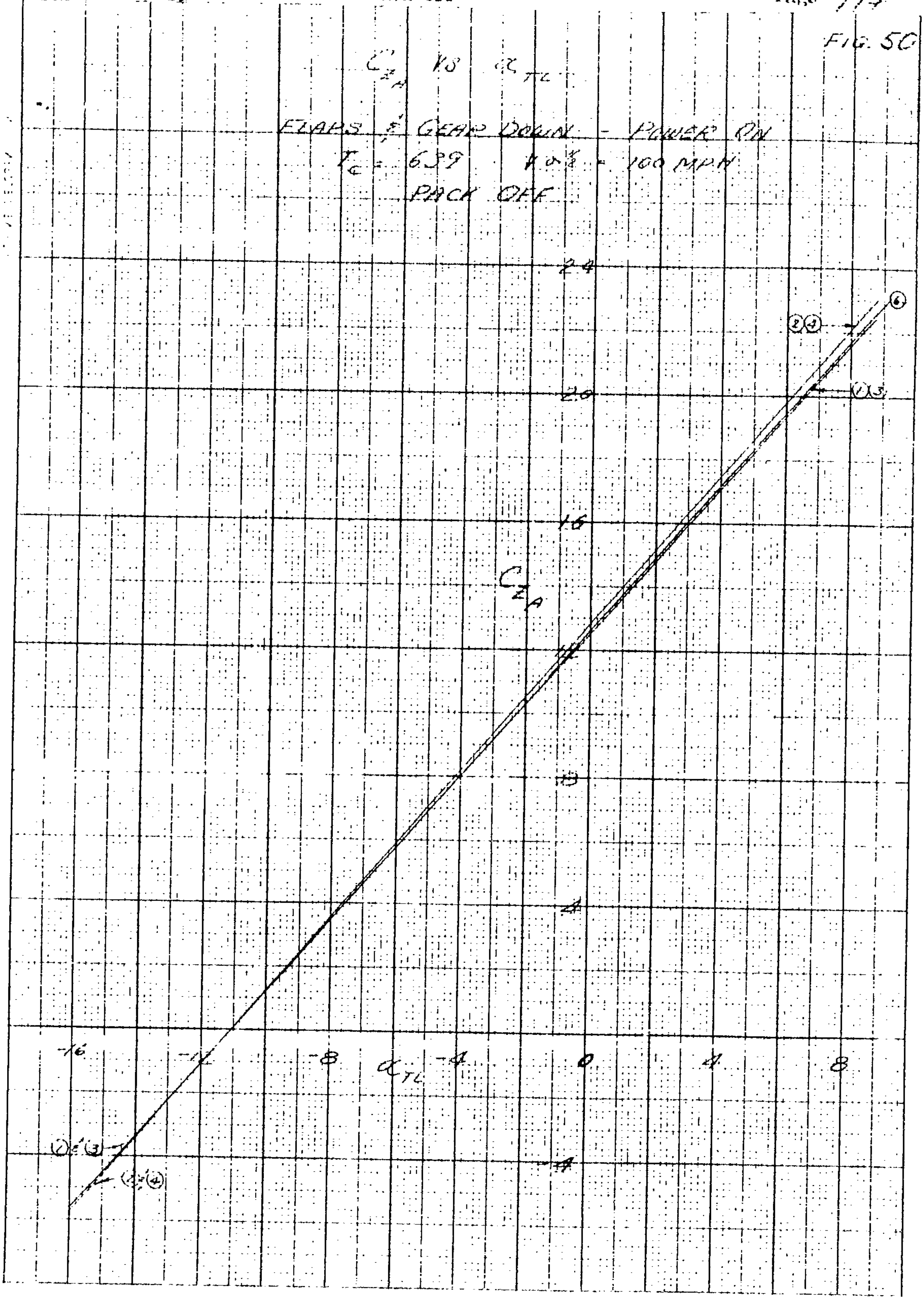
PART II-B-2-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS
FLAPS & GEAR DOWN POWER ON
$V\sqrt{\sigma} = 100$ mph $T_c = .639$

C.G. #4				C.G. #6			
$C_{M_{A-T}}$	$\frac{MAC_W}{l_{At}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{At}}$	$C_{Z_{HT}}$	C_{Z_A}
-.3967	.2769	-.10985	-.10100	-.3995	+.2729	-.1090	-.1001
-.3477		-.09627	.3565	-.3716		-.10135	.3514
-.2998		-.08301	.8153	-.3472		-.0948	.8035
-.2510		-.06950	1.267	-.3224		-.0881	1.2487
-.2046		-.05665	1.7488	-.2993		-.0817	1.7237
-.1627		-.04505	2.1982	-.2800		-.0764	2.1668

C.G. #7			
$C_{M_{A-T}}$	$\frac{MAC_W}{l_{At}}$	$C_{Z_{HT}}$	C_{Z_A}
-.3802	.2780	-.1057	-.09685
-.3250		-.0903	+.36243
-.2644		-.0733	+.8250
-.2051		-.0570	+1.2798
-.1478		-.0410	+1.7644
-.0893		-.0248	+2.2184

FIG. 50



RESTRICTED

Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-E-2-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR DOWN POWER ON

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

					C.G. #1			
α_{TL}	$C_{Z_{A-T}}$ POWER OFF	$24 C_{Z_{NF1}}$	$2 C_{Z_P}$	$C_{Z_{A-T}}$ POWER ON	$C_{M_{A-T}}$	$\frac{MAC_w}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
-12	-.07775	.0320	-.0122	-.05995	-.2989	+.2709	-.08097	-.1409
-8	.25153	.0650	-.0083	.33823	-.2825		-.07652	.2617
-4	.6342	.0990	-.0042	.7340	-.2696		-.07303	.6613
0	.9888	.1350	0	1.1238	-.2535		-.06867	1.055
4	1.3709	.1700	.0042	1.5451	-.2384		-.06458	1.4845
8	1.7190	.2030	.0083	1.9303	-.2276		-.06165	1.8687

C.G. #2				C.G. 3			
$C_{M_{A-T}}$	$\frac{MAC_w}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_w}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
-.3027	+.2769	-.08381	-.1438	-.3000	+.2709	-.08127	-.1412
-.2555		-.07074	.2625	-.2797		-.07577	.2625
-.2097		-.05806	.6759	-.2645		-.07165	.6624
-.1619		-.04483	1.0790	-.2506		-.06788	1.056
-.1168		-.03234	1.5126	-.2393		-.06482	1.4803
-.0743		-.02057	1.9097	-.2342		-.06358	1.8667

Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-E-2-b

COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

FLAPS & GEAR DOWN POWER ON

$V\sqrt{\sigma}$ = 160 mph T_c = .195

C.G. # 4				C.G. # 6			
$C_{M_{A-T}}$	$\frac{MAC_W}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
-.3037	+.2769	-.08408	-.1430	-.3029	+.2729	-.0827	-.1427
-.2528		-.07000	.2682	-.2718		-.0741	.2641
-.2060		-.05704	.6770	-.2461		-.0673	.6667
-.1598		-.04424	1.0796	-.2219		-.0605	1.0633
-.1176		-.03256	1.5125	-.1994		-.0545	1.4906
-.0814		-.02253	1.9078	-.1830		-.0499	1.8804

C.G. # 7			
$C_{M_{A-T}}$	$\frac{MAC_W}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
-.3065	.2780	-.0852	-.14515
-.2499		-.0693	+.26893
-.1952		-.0542	+.6798
-.1372		-.0382	+1.0856
-.0844		-.0235	+1.5216
-.0329		-.00915	+1.9212

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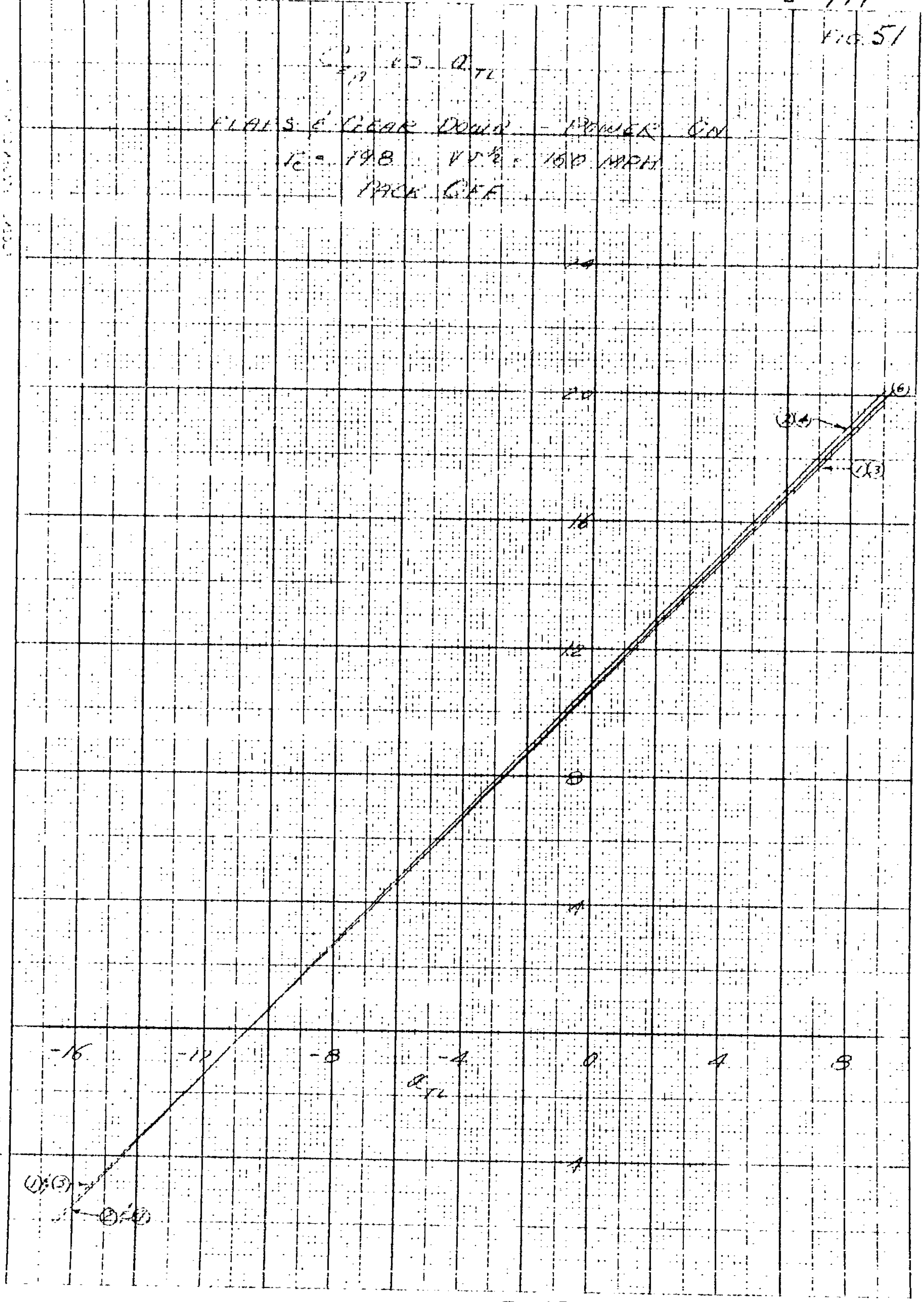
Pack-Off

Fig. 51



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

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$\underline{C_{LHT}} = \eta_{ht} C_{LHT} \frac{S_{HT}}{S_W} \frac{q_{ht}}{q} = \eta_{ht} C_{LHT}\ .2392 \times 1.00$

AIRPLANE LIFT COEFFICIENT

POWER OFF - FLAPS AND GEAR UP

								CG. 1		
$\alpha_{T.L.}$	C_{LW}	C_{LN}	$\underline{C_{LN}}$	C_{LB} (ONE)	$\underline{\Delta C_{LB}}$	C_{LVT}	C_{L-T}	$\eta_{ht} C_{LHT}$	C_{LHT}	C_{LA}
-12	-.3260	-.2690	-.0836	-.0290	-.0125	0	-.4221	-.1287	-.0308	-.4529
-8	.0110	-.1950	-.0606	-.0130	-.0056	0	-.0552	-.1109	-.0265	-.0817
-4	.3477	-.1330	-.0414	-.0025	-.0011	0	.3052	-.0988	-.0236	.2816
0	.6846	-.1090	-.0339	0	0	0	.6507	-.0831	-.0199	.6308
4	1.0210	-.0850	-.0264	.0025	.0011	0	.9957	-.0674	-.0161	.9796
8	1.3580	+.0010	.0003	.0130	.0056	0	1.3639	-.0574	-.0137	1.3502

CG 2			CG 3			CG 4			CG 5		
$\eta_{ht} C_{LHT}$	$\underline{C_{LHT}}$	C_{LA}	$\eta_{ht} C_{LHT}$	$\underline{C_{LHT}}$	C_{LA}	$\eta_{ht} C_{LHT}$	$\underline{C_{LHT}}$	C_{LA}	$\eta_{ht} C_{LHT}$	$\underline{C_{LHT}}$	C_{LA}
-.1710	-.0409	-.4630	-.1332	-.0319	-.4540	-.1756	-.0420	-.4641	-.1537	-.0376	-.4597
-.1194	-.0286	-.0838	-.1097	-.0262	-.0814	-.1102	-.0283	-.0835	-.1148	-.0275	-.0827
-.0720	-.0172	.2880	-.0939	-.0225	.2827	-.0683	-.0163	.2889	-.0819	-.0196	.2856
-.0248	-.0059	.6448	-.0798	-.0191	.6316	-.0199	-.0048	.6459	-.0496	-.0119	.6388
.0240	.0057	1.0014	-.0674	-.0161	.9796	+.0241	.0058	1.0015	-.0175	-.0042	.9915
.0673	.0161	1.3800	-.0626	-.0150	1.3489	.0616	.0147	1.3786	.0086	.0021	1.3660

PART II-F

AIRPLANE LIFT COEFFICIENT (CONT.)

POWER OFF – FLAPS AND GEAR UP

CG 7

$C_{M_{CENT}}$	$C_{L_{HT}}$	C_{L_A}
-.1853	-.0443	-.4664
-.1233	-.0295	-.0847
-.06577	-.0157	.2695
-.00567	-.0014	.6493
.04826	.0115	1.0072
.1014	.0243	1.3882

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AIRPLANE DRAG COEFFICIENT
POWER OFF – FLAPS AND GEAR UP

$\alpha_{F.L}$	α_{ht}	C_{D_W}	C_{D_N}	$\underline{C_{D_N}}$	C_{D_B} (ONE)	$2C_{D_B}$	$C_{D_{VT}}$	$\underline{2C_{D_{VT}}}$	$C_{D_{A-T}}$	CG 1 δ_e
-12	-9.40	.0120	.17790	.0072	.230	.0130	.0081	.000114	.0323	+12.0°
-8	-7.05	.0070	.14482	.0059	.205	.0116			.0246	8.39°
-4	-4.70	.0120	.12901	.0052	.191	.0108			.0281	4.58°
0	-2.35	.0270	.12286	.0050	.189	.0107			.0428	.92°
4	0	.0525	.12901	.0052	.191	.0108			.0686	-2.74°
8	2.35	.0990	.14482	.0059	.205	.0116			.1166	-6.63°

CG 1			CG 2				CG 3			
$C_{D_{HT}}$	$\underline{C_{D_{HT}}}$	C_{D_A}	δ_e	$C_{D_{HT}}$	$\underline{C_{D_{HT}}}$	C_{D_A}	δ_e	$C_{D_{HT}}$	$\underline{C_{D_{HT}}}$	C_{D_A}
.0142	.0034	.0357	+10.2°	.0137	.0033	.0356	11.88	.0141	.0034	.0357
.0117	.0028	.0274	8.04°	.0117	.0028	.0274	8.44°	.0117	.0028	.0274
.0098	.0023	.0304	5.67°	.0098	.0023	.0304	4.78°	.0098	.0023	.0304
.0091	.0022	.0450	3.29°	.0094	.0022	.0450	1.06°	.0092	.0022	.0450
.0095	.0023	.0709	.98°	.0085	.0020	.0706	-2.74°	.0095	.0023	.0709
.0111	.0027	.1193	-1.57°	.0091	.0022	.1188	-6.85°	.0114	.0027	.1193

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AIRPLANE DRAG COEFFICIENT

POWER OFF — FLAPS & GEAR UP

	CG 4				CG 5				CG 7	
$\alpha_{T.L.}$	δe	C_{DHT}	C_{DHT}	C_{DA}	δe	C_{DHT}	C_{DHT}	C_{DA}	δe	C_{DNT}
-12	+10.10°	.0137	.0033	.0356	+10.90°	.0140	.0033	.0356	9.66°	.0137
-8	8.09°	.0117	.0028	.0274	8.23°	.0117	.0028	.0274	7.88°	.0117
-4	5.82°	.0101	.0024	.0305	5.27°	.0098	.0023	.0304	5.92°	.0101
0	3.49°	.0094	.0022	.0450	2.28°	.0093	.0022	.0450	4.07°	.0094
4	.98°	.0085	.0020	.0706	-.71°	.0084	.0020	.0706	1.96°	.0092
8	-1.80°	.0092	.0022	.1188	-3.95°	.0094	.0022	.1188	-.18°	.0090

C_{DHT}	C_{DA}
.0033	.0356
.0028	.0274
.0024	.0305
.0022	.0450
.0022	.0708
.0022	.1188

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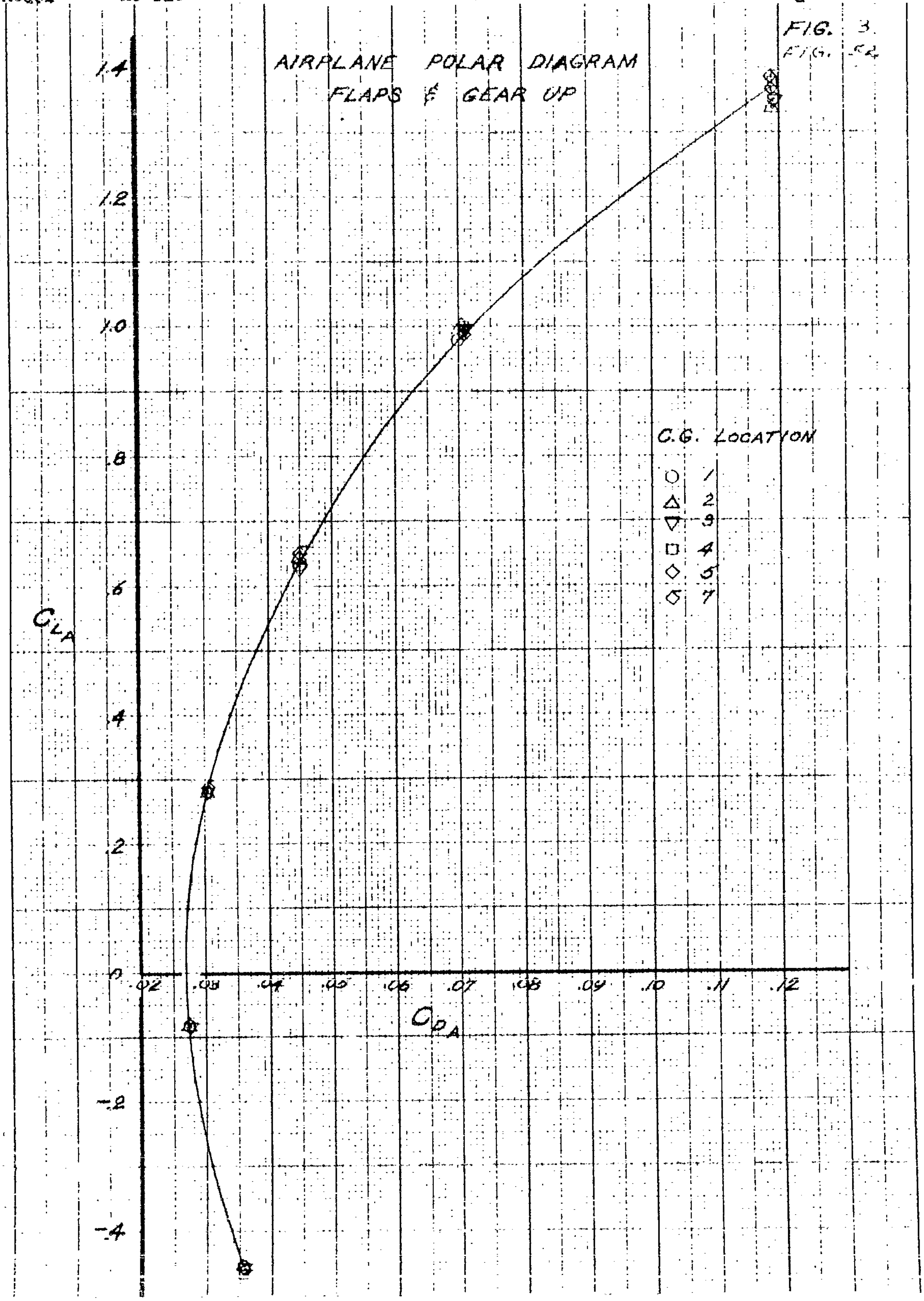
FIG. 3.


AIRPLANE POLAR DIAGRAM
FLAPS & GEAR UP
C.G. LOCATION
○ 1
△ 2
▽ 3
□ 4
◇ 5
7
C_{L_A}
C_{D_A}
1.4
1.2
1.0
.8
.6
.4
.2
0
-.2
-.4
.02
.03
.04
.05
.06
.07
.08
.09
.10
.11
.12

Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-G-1

G. V-n Diagrams

1. Limit Speeds & 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 DIAGRAMS

Type of Diagram	B		
Gross Weights Considered	Design Gross Weight Take-Off Gross Weight Landing Gross Weight	54,000 lbs.	
	Light Weight Alternate Condition	37,980 lbs.	
	Minimum Flying Weight	37,015 lbs.	
Load Factors Symmetrical	All weights +3.0 and -1.5		
Assymmetrical	All Weights - At $V_R = 0.8\ V_H$, +2 and -1 with $\frac{p\ B_W}{2\ V} = .08$ At $V_D = 1.25\ V_H$, +2 with $\frac{p\ B_W}{2\ V} = .015$ (Since flaps down is intended for landing only - no V-n diagram is required for flapped condition).		
Altitudes Considered	Sea Level	17,000 ft. (Highest MN)	
V_H True mph	265	285	
V_R True mph	212	228	
V_D True mph	331.3	356	
V_T True mph	495	511	

Converting the above speeds at 17,000 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.

PART II-G-1 (Cont.)

Terminal velocities for the preceding table were calculated by the method of reference (2).

$$V_{T_{mph}} = 0.6818\ M_T \times a$$

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 (4) for 2418 at $C_L = 0$ of .605.

C_{D_f} = equivalent airplane parasite drag coefficient at zero lift = .034 from performance calculations.

S_w = 1447.25 sq. ft.

$\frac{P}{P_o} = \frac{a^2 \sigma}{a_o^2}$ at sea level = 1.0
at 17,000 ft. = 0.5201

K = Constant for entering figure 3 of reference (2) -2B.

$$= \frac{W/S_w}{1481\, P/P_o\, C_{D_f}\ M^2_{CR}}$$

$$= \left[\frac{1}{(1447.25)(1481)(.034)(.366)}\right] \frac{W}{\left(\frac{P}{P_o}\right)}$$

$$= \frac{1}{26,650} \times \left(\frac{W}{\left(\frac{P}{P_o}\right)}\right)$$

Since maximum terminal velocity is realized with maximum gross weight, calculations are made for 54,000 lbs. only for simplification purposes.

$$V_{T_{SL}} = 0.6818 \times .65 \times 1116.19 = 495 \text{ mph}$$

$$V_{T\ 17000} = 0.6818 \times .714 \times 1050 = 511 \text{ mph}$$

PART II-G-2

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

$$\lambda = \frac{C_{Z_{A\,max}}\,q}{W/S_W}$$

Gross Weight	W/S_W	S_W/W
54,000 lbs.	37.31	.0268
37,980 lbs.	26.24	.0381
37,015 lbs.	25.58	.0391

$\alpha_{T.L.}$ for $+C_{Z_{A_{max}}} = 8.55°$ $\quad \alpha_{W_c} = 15.55°$

$\alpha_{T.L.}$ for $-C_{Z_{A_{max}}} = -14.88$ $\quad \alpha_{W_c} = -7.88°$

C. G. Location	$+C_{Z_{A_{max}}}$	$-C_{Z_{A_{max}}}$	$dC_{Z_A}/d\alpha_{T.L.}$
1	1.4041	-.7147	.09043
2	1.4351	-.7303	.09242
3	1.4049	-.7137	.09042
4	1.4362	-.7290	.09241
5	1.4214	-.7227	.09151
7	1.4446	-.7339	.0930

In order to simplify 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-C-2

Power Off

Maximum η's for C. G. Locations 1, 2, 3, 4, 5, & 7					
$V\sqrt{\sigma}$ mph	q	$C_{Z_{max}}$ q	54,000 lbs.	37,980 lbs.	37,015 lbs.
75	14.39	+20.8 -10.55	+.556 -.282	+.791 -.401	+.814 -.413
100	25.58	+37.0 -18.8	+.990 -.502	+1.41 -.717	+1.45 -.731
150	57.54	+83.1 -42.2	+2.225 -1.13	+3.17 -1.61	+3.26 -1.65
200	102.3	+148.0 -75.2	+3.96 -2.02	+5.65 -2.88	+5.79 -2.95
250	159.85	+231.0 -117.0	+6.18 -3.13	+8.80 -4.45	+9.02 -4.58
265	179.60	+259.0 -132.0	+6.94 -3.54	+9.86 -5.02	+10.15 -5.17

b. Flaps and Gear Up Power On

The power on maximum lift lines are complicated by the effects of power on the normal force coefficients and are considered at various speeds.

C. G. Location	$+C_{Z_{A_{max}}}$	$-C_{Z_{A_{max}}}$	$\frac{d\,C_{Z_A}}{d\,\alpha_{TL}}$
	$V\sqrt{\sigma}$ = 100 mph	T_c = .639	
1	1.630	-.762	.1026
2	1.678	-.781	.1050
3	1.652	-.771	.1027
4	1.672	-.787	.1050
5	1.657	-.775	.1039
7	1.6886	-.7866	.1056
	$V\sqrt{\sigma}$ = 160 mph	T_c = .198	
1	1.5082	-.7554	.09661
2	1.5415	-.7722	.09875
3	1.5068	-.7567	.09661
4	1.5381	-.7724	.09861
5	1.5258	-.7652	.09778
7	1.5516	-.7762	.09935

PART II-G-2b (Cont.)

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

1	1.4804	-.7502	.09520
2	1.5137	-.7681	.09739
3	1.4804	-.7521	.09528
4	1.5134	-.7683	.09738
5	1.4987	-.7612	.09645
7	1.5237	-.7722	.09799

C. G. Location	$+C_{Z_{A max}}$	$-C_{Z_{A max}}$	$\frac{d\,C_{Z_A}}{d\alpha_{TL}}$
$V\sqrt{\sigma}$ = 265 mph		T_c = .047	
1	1.4417	-.7471	.09342
2	1.4735	-.7634	.09547
3	1.4425	-.7462	.09342
4	1.4745	-.7621	.09546
5	1.4593	-.7555	.09453
7	1.4832	-.7677	.09607

In order to simplify construction of the V-n diagram the maximum positive and negative values of C_{Z_A} and maximum $\frac{d\,C_{Z_A}}{d\alpha_{TL}}$ 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. Locations 1, 2, 3, 4, 5, & 7

Power On

$V\sqrt{\sigma}$ mph	q	$C_{Z_{A\,max}}\,q$	54,000 lbs.	37,980 lbs.	37,015 lbs.
100	25.58	43.19	1.157	1.646	1.689
		-20.12	-.539	-.767	.787
160	65.5	101.63	2.724	3.872	3.974
		-50.84	-1.355	-1.937	1.980
185	87.5	133.32	3.573	5.079	5.213
		-67.57	1.811	-2.574	-2.642

PART II-G-3

3. <u>Gust Load Factors</u>

Gust load factors are determined in accordance with reference (2).

a. <u>Flaps and Gear Up - Power Off</u>

$$n = 1 \pm \frac{5.0\,K \,\frac{d\,C_{Z_A}}{d\alpha_{T.L.}}\, V\sqrt{\sigma}}{W/S_w}$$

Gross Weight	W/S_w	K	$\frac{5.0\,K}{W/S_w}$
54,000 lbs.	37.31	.575	.07706
37,980 lbs.	26.24	.550	.10480
37,015 lbs.	25.58	.549	.10731

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

So:

$$d\,C_{Z_A}/d\alpha_{T.L.} = .0930$$

Gross Weight	n
54,000 lbs.	$1 \pm .07706 \times .0930 \times V\sqrt{\sigma}$
37,980 lbs.	$1 \pm .10480 \times .0930 \times V\sqrt{\sigma}$
37,015 lbs.	$1 \pm .10731 \times .0930 \times V\sqrt{\sigma}$

n Gust Loads - Power Off

Speed	GROSS WEIGHT		
$V\sqrt{\sigma}$ mph	54,000 lbs.	37,980 lbs.	37,015 lbs.
100	+1.716	+1.975	+1.995
	+ .284	+ .025	- .005
200	+2.43	+2.95	+2.99
	- .43	- .95	- .99
265	+2.90	+3.58	+3.64
	- .90	-1.58	-1.64

PART II-G-3b

b. **Flaps and Gear Up - Power On**

$$n = 1 \pm \frac{5.0\,K\,\frac{d\,Cz_A}{d\alpha_{TL}}\,V\sqrt{\sigma}}{W/S_W}$$

Gross Weight	W/S_W	K	$5.0\,K/W/S_W$
54,000 lbs.	37.31	.575	.07706
37,980 lbs.	26.24	.550	.10480
37,015 lbs.	25.58	.549	.10731

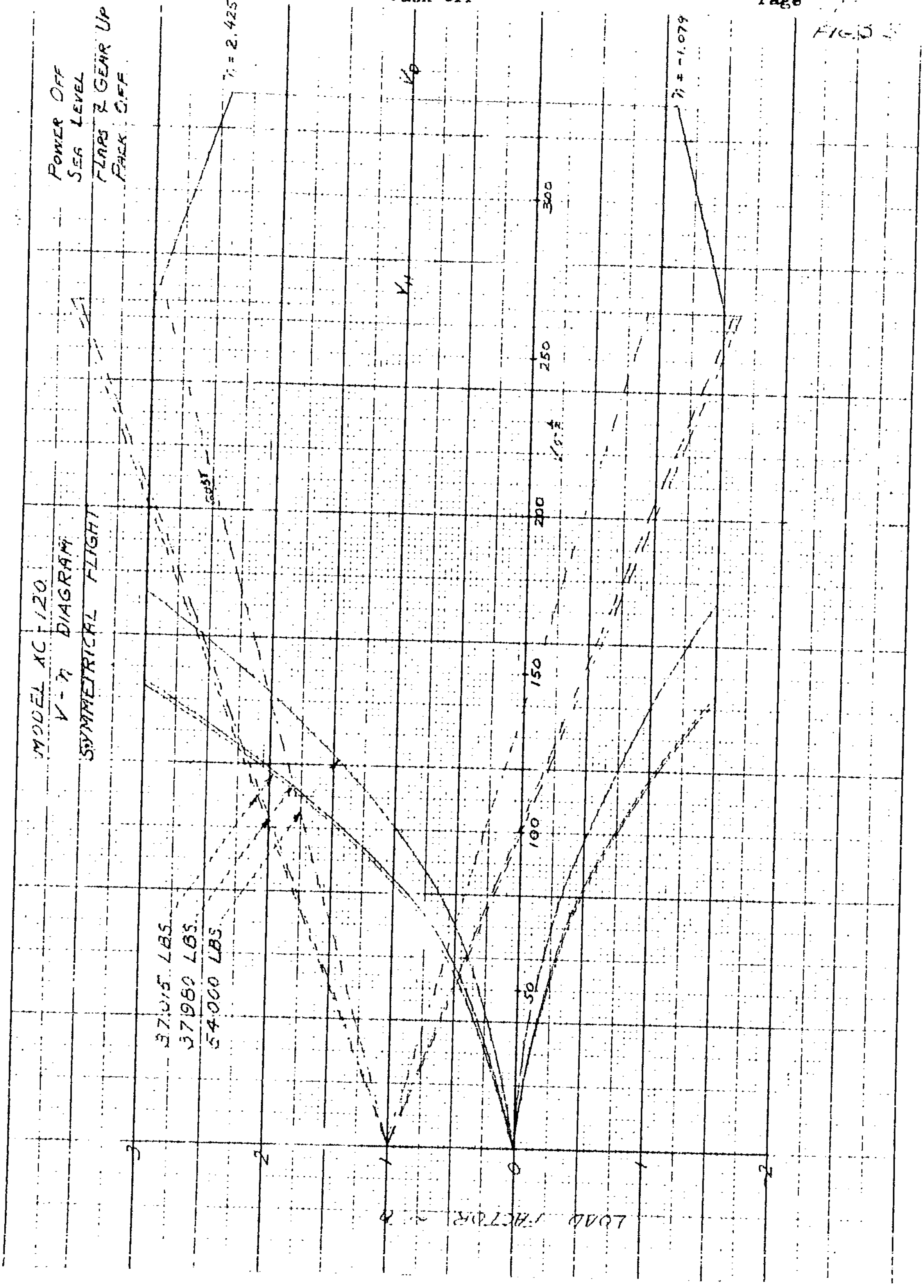
In order to simplify construction of the V-n diagram, the maximum value of $d\,Cz_A/d\alpha_{TL}$ at a given speed shall be considered as applicable to all c.g. locations.

$V\sqrt{\sigma}$ mph	$d\,Cz_A/d\alpha_{TL}$
100	.1056
160	.09935
185	.09799
265	.09607

n - Gust Loads - Power On

Speed $V\sqrt{\sigma}$ mph	Gross Weight 54,000 lbs.	37,980 lbs.	37,015 lbs.
100	1.814 .186	2.106 -.106	2.133 -.133
160	2.224 -.224	2.666 -.666	2.706 -.706
185	2.397 -.397	2.900 -.900	2.945 -.945
265	2.962 -.962	3.668 -1.668	3.732 -1.732

FIG 5



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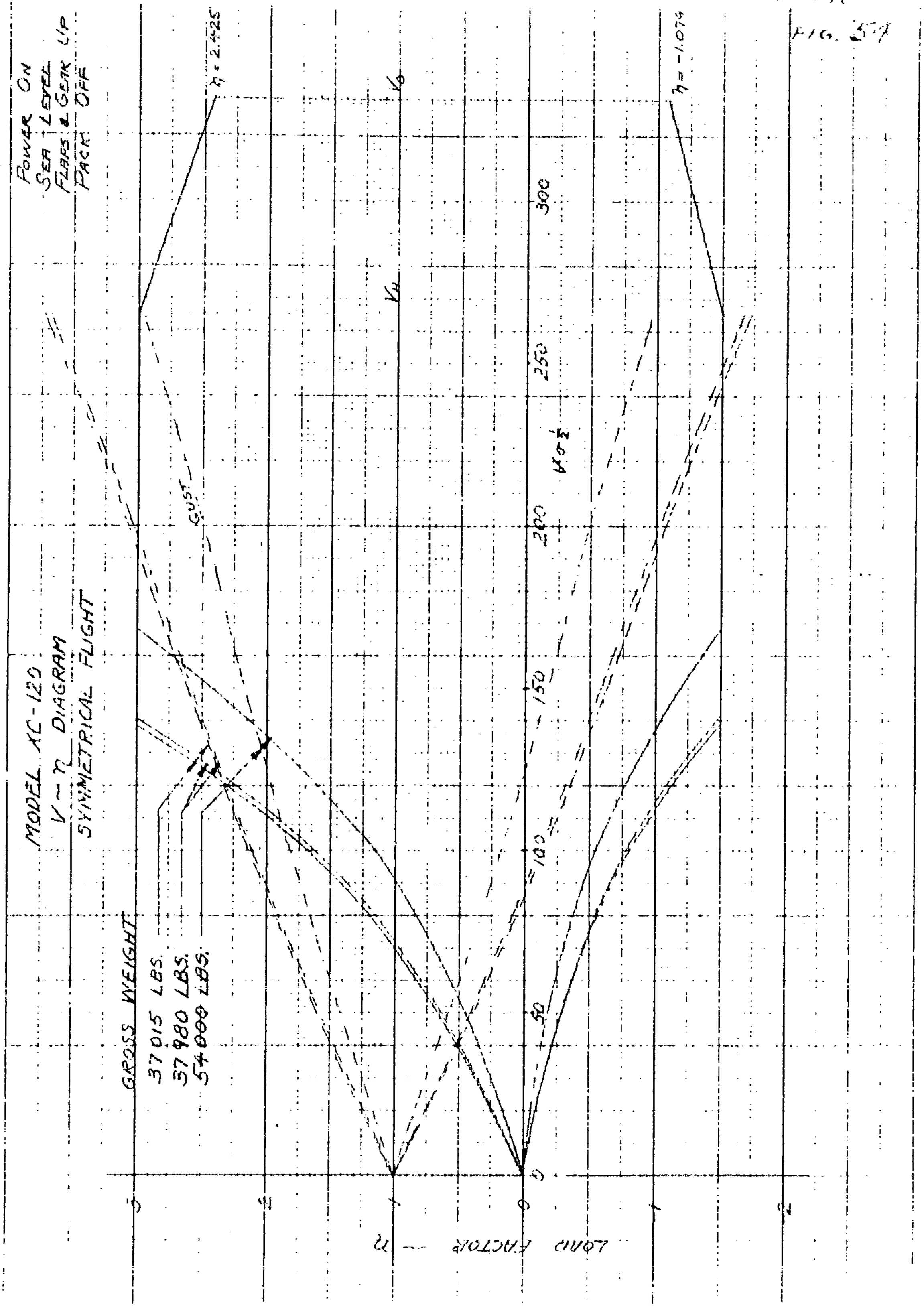


FIG. 57
POWER ON
SEA LEVEL
FLAPS & GEAR UP
PACK OFF
MODEL XC-120
V ~ n DIAGRAM
SYMMETRICAL FLIGHT
GROSS WEIGHT
37015 LBS.
37980 LBS.
54000 LBS.
n = 2.425
n = -1.074
GUST
LOAD FACTOR — n

PART II-G-3c

c. **Flaps & Gear Down - Power Off**

As the limiting speed for the flapped wing is 160 mph; V-n diagrams will show only gust loads for power off and power on up to that speed.

$$n = 1 \pm \frac{5.0\,K \dfrac{d\,C_{Z_A}}{d\alpha_{TL}} V\sqrt{\sigma}}{W/S_W}$$

Gross Weight	W/S_W	K	$\frac{5.0\,K}{W/S_W}$
54,000 lbs.	37.31	.575	.07706
37,980 lbs.	26.24	.550	.10480
37,015 lbs.	25.58	.549	.10731

C. G. Location	$d\,C_{Z_A}/d\alpha_{TL}$
1	.09068
2	.09268
3	.09060
4	.09259
6	.09130
7	.09323

The maximum value of $d\,C_{Z_A}/d\alpha_{TL}$ shall be considered as applicable to all C. G. Locations.

Gross Weight	n
54,000 lbs.	$1 \pm .07706 \times .09323 \times V\sqrt{\sigma}$
37,980 lbs.	$1 \pm .10480 \times .09323 \times V\sqrt{\sigma}$
37,015 lbs.	$1 \pm .10731 \times .09323 \times V\sqrt{\sigma}$

n - Gust Loads - Power Off

Speed $V\sqrt{\sigma}$ mph	Gross Weight 54,000 lbs	37,980 lbs	37,015 lbs
50	1.3592	1.4885	1.5002
	.6408	.5115	.4998
100	1.7184	1.9771	2.0005
	.2816	.0229	-.0005
160	2.1495	2.5633	2.6007
	-.1495	-.5633	-.6007

PART II-G-3d

d. **Flaps & Gear Down - Power On**

$$n = 1 \pm \frac{5.0\ K\ \frac{d\ C_{Z_A}}{d\alpha_{TL}}\ V\sqrt{\sigma}}{W/S_W}$$

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

C. G. Location	$\frac{d\ C_{Z_A}}{d\alpha_{TL}}$
1	.11261
2	.11507
3	.11252
4	.11496
6	.11335
7	.11576

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

C. G. Location	$\frac{d\ C_{Z_A}}{d\alpha_{TL}}$
1	.10048
2	.10267
3	.10239
4	.10254
6	.10115
7	.10332

Maximum values of $d\ C_{Z_A}/d\alpha_{TL}$ shall be considered as applicable to all c. g. locations for any given speed.

54,000 lbs. $1 \pm .07706 \times d\ C_{Z_A}/d\alpha_{TL} \times V\sqrt{\sigma}$

37,980 lbs. $1 \pm .10480 \times d\ C_{Z_A}/d\alpha_{TL} \times V\sqrt{\sigma}$

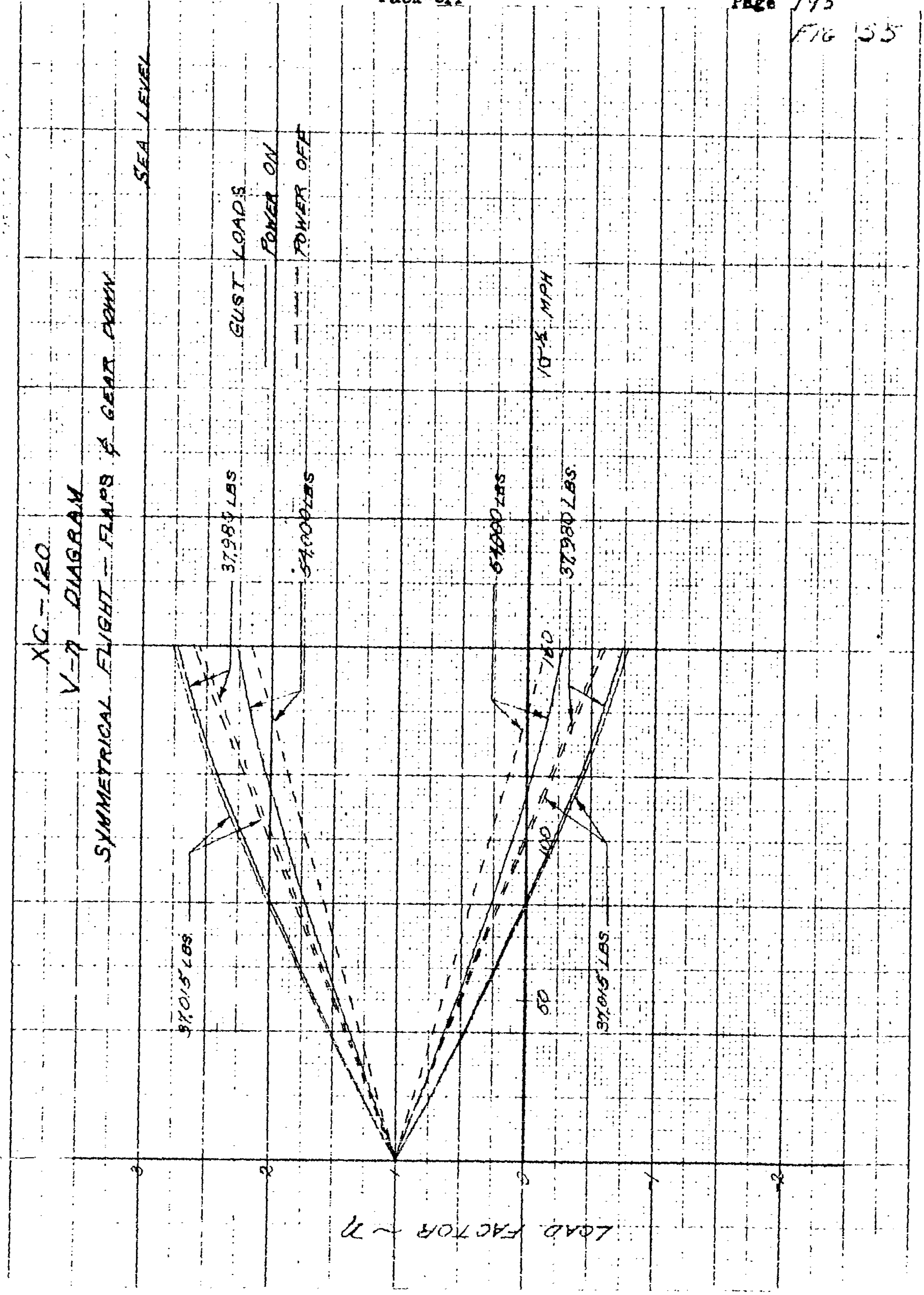
37,015 lbs. $1 \pm .10731 \times d\ C_{Z_A}/d\alpha_{TL} \times V\sqrt{\sigma}$

Gust Loads - Power On

$V\sqrt{\sigma}$ mph	Gross Weight 54,000	37,980	37,015
100	1.8920	2.2132	2.2422
	+.1080	-.2132	-.2422
160	2.2739	2.7308	2.7722
	-.2739	-.7308	-.7722

Pack-Off

FIG 55



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 $\triangle_{T.L.}$ 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 at C_{Z_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 light weight alternate and minimum flying weight gust conditions at $PLAA_{VH}$ are checked for possible 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.

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MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
			DATE 1 May 1949	
Subject: BASIC FLIGHT CRITERIA - PACK OFF			REVISED	

ANALYSIS OF AIRPLANE COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS
Condition Flaps & Gear Up - Power Off

V-n Diagram C. G. Location Gross Weight lbs.	PHAA 3 54000	NHAA 1 54000	$PLAA_{VH}$ 1 54000	$PLAA_{VD}$ 1 57000	$NLAA_{VH}$ 7 54000	$NLAA_{VD}$ 7 54000
AIRPLANE						
ηZ_A	3.0	-1.5	3.0	2.425	-1.5	-1.079
W/S_W, #/sq. ft.	37.312	37.312	37.312	37.312	37.312	37.312
$\eta A\ W/S_W$	+111.94	-55.968	+111.94	+90.482	-55.968	-40.260
C_{Z_A}	+1.39832	-.70406	+.62327	+.33234	-.31163	-.14343
$q = \eta A(W/S_W)/C_{Z_A}$, #/sq. ft.	+80.053	+79.493	179.6	280.7	179.6	280.7
$q/(W/S_W)$	2.1455	2.1305	4.8135	7.5230	4.8135	7.5230
$V\sigma^{1/2} = \sqrt{391\ q}$, mph	176.9	176.3	265	331.3	265	331.3
α T.L., degrees	8.55°	-14.88°	-0.10°	-3.47°	-10.38°	-8.59°
C_{X_A}	-.08251	-.12800	+.04652	+.05421	-.01756	+.01226
$\eta X_A = C_{X_A}\ q/(W/S_W)$	-.17703	-.27267	+.22392	+.40782	-.08453	+.09223
C_{L_A}	+1.39926	-.72172	+.62219	+.32913	-.30980	-.13983
C_{D_A}	+.12867	+.05008	+.04532	+.03291	+.03174	+.02876
WING						
C_{L_W}	1.4010	-.5670	+.6761	+.3924	-.1895	-.0387
C_{D_W}	.1073	.0215	.0264	.0134	.0087	.0068
C_{Z_W}	1.4014	.5421	.6761	.3909	-.1880	-.0393
C_{X_W}	-.1021	-.1248	.0276	.0371	-.0255	.0009
$\eta Z_W = C_{Z_W}\ q/(W/S_W)$	+3.0067	-1.1558	+3.2544	+2.9407	-.9049	-.2957
$\eta X_W = C_{X_W}\ q/(W/S_W)$	-.2191	-.2659	+.1329	+.2791	-.1227	.00677
$Z_W = \eta Z_W\ W$, lbs.	+162,361	-62,413	+175,736	+158,800	-48,866	-15,965
$X_W = \eta X_W\ W$, lbs.	-11,829	-14,358	+7,174	+15,072	-6,628	+366
C_{M_W} (a.c.)	-.0630	-.0630	-.0630	-.0630	-.0630	-.0630
$M_W = q\ S_W\ MAC_W\ C_{M_W}$, ft. lbs.	-102,353	-101,637	-229,631	-358,895	-229,631	-358,895
ΔC_{M_A} due to wing	-.15959	-.04664	-.09564	-.08072	-.07414	-.06463

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Subject:— BASIC FLIGHT CRITERIA – PACK OFF

FLAPS & GEAR UP – PACK OFF – POWER OFF

V-n Diagram	PHAA	NHAA	PLAA VH	PLAA VD	NLAA VH	NLAA VD
C. G. Location	3	1	1	1	7	7
Gross Weight lbs.	54000	54000	54000	54000	54000	54000
α TL degrees	8.55°	-14.88°	-0.10°	-3.47°	-10.38°	-8.59°
NACELLE						
C_{L_N}	.0068	-.1010	-.0839	-.0395	-.0738	-.0687
C_{D_N}	+.0060	+.0089	+.0050	+.0052	+.0066	+.0060
C_{Z_N}	+.0039	-.1045	-.0345	-.0423	-.0731	-.0634
C_{X_N}	+.0051	-.0173	+.0050	+.0030	-.0066	-.0034
ηZ_N $C_{Z_N}\, q/(W/S_W)$	+.00833	-.22264	-.16607	-.31822	-.35187	-.47696
ηX_N $C_{X_N}\, q/(W/S_W)$	+.01094	-.03686	+.02407	+.02257	-.03177	-.02257
$Z_N = \eta Z_N W$, lbs.	+452	-12,022	-8967	-17,184	-19,001	-25,756
$X_N = \eta X_N W$, lbs.	+591	-1990	+1300	+1219	-1,716	-1,381
C_{M_N} (.25 L_f)	+.0268	+.0062	+.0243	+.0174	+.0098	+.0115
$M_N = q\, S_W\, MAC_W\, C_{M_N}$, ft lbs.	+33792	+10,022	+88572	+99,123	+35721	+65,513
ΔC_{M_A} due to nacelle	+.02482	-.07001	+.00117	-.01156	-.04978	-.03974
BOOM (ONE)						
C_{L_B}	.00322	-.00913	.0000	-.0004	-.0047	-.0033
C_{D_B}	+.00574	+.00733	+.0053	+.0054	+.0061	+.0057
C_{Z_B}	+.0040	-.0108	.0000	-.0004	-.0058	-.0044
C_{X_B}	+.0053	+.0045	+.0053	+.0053	+.0052	+.0053
$\eta Z_B = C_{Z_B}\, q/(W/S_W)$	+.00858	-.01983	+.0000	-.00301	-.02792	-.03084
$\eta X_B = C_{X_B}\, q/(W/S_W)$	+.01137	+.00826	+.02551	+.03987	+.02503	+.03987
$Z_B = \eta Z_B W$, lbs.	+463	-1242	.0000	-162	-1508	-1666
$X_B = \eta X_B W$, lbs.	+614	+518	+1378	+2153	+1357	+2153
C_{M_B} (.25 L_B)	+.0407	-.0092	+.0118	+.0041	-.0050	-.0033
$M_B = q\, S_W\, MAC_W\, C_{M_B}$, ft. lbs.	+66124	-14842	+43010	+23357	-18225	-18799
ΔC_{M_A} due to one boom	+.03760	-.00719	+.01126	+.00365	-.004783	-.00331

Subject: BASIC FLIGHT CRITERIA - PACK OFF

FLAPS & GEAR UP - PACK OFF - POWER OFF

V-n Diagram	PHAA	NHAA	PLAA $_{VH}$	PLAA $_{VD}$	NLAA $_{VH}$	NLAA $_{VD}$
C. G. Location	3	1	1	1	7	7
Gross Weight lbs.	54000	54000	54000	54000	54000	54000
α_{TL} degrees	8.55°	-14.68°	-0.10°	-3.47	-[illegible]	-6.57°
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
$\eta X_{VT} = C_{X_{VT}}\, q/(W/S_W)$	.00122	.00121	.00274	.00429	.00274	.00429
$X_{VT} = \eta X_{VT}\, W$, lbs.	66	65	148	232	148	232
ΔC_{M_A} due to one tail	+.000175	.000138	.000138	.000138	.000138	.000138
HORIZONTAL TAIL						
α_{ht}	+2.[illegible]°	-11.15°	-2.45°	-4.41°	-8.49°	-7.44
q_{ht}/q	1.00	1.00	1.00	1.00	1.00	1.00
δ_e	-7.45°	+14.26°	+1.08°	+4.16°	9.22°	8.36°
$\eta_{ht}\, C_{L_{HT}}$	-.014976	-.035461	-.020010	-.022972	-.037101	-.03083
$C_{D_{HT}}$	+.00275	+.00388	+.00218	+.00237	.0031	.0030
$C_{Z_{HT}}$	-.014976	-.035461	-.020010	-.022972	-.037101	-.03083
$C_{X_{HT}}$	+.00275	+.00388	+.00218	+.00237	.0031	.0030
$\eta Z_{HT} = C_{Z_{HT}}\, q/(W/S_W)$	-.032131	-.075550	-.096270	-.172818	-.1786	-.23193
$\eta X_{HT} = C_{X_{HT}}\, q/(W/S_W)$	+.00590	+.00827	+.01049	+.01783	.01492	.02257
$Z_{HT} = \eta Z_{HT}\, W$, lbs.	-1732	-4076	-5196	-9323	-9644	-12,526
$X_{HT} = \eta X_{HT}\, W$, lbs.	+314	+447	+566	+963	806	1218[illegible]
$C_{M_{HT}}$ (a. c.)	+.009104	-.01743	-.001320	-.00508	-.010375	-.009097
$M_{HT} = q\, S_W\, MAC_W\, C_{M_{HT}}$, ft lbs	+14,791	-28,120	-4,811	-28,939	-37,816	-51,793
SUMMATION						
Z_A, lbs.	+162,006	-80,995	+161,573	+131,469	-80,527	-57,579
X_A, lbs.	-9,559	-14,735	+12,092	+22,024	-4,540	+4974
$C_{M_{A-T}}$	-.053220	-.130754	-.073784	-.084704	-.13321	-.11071

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Subject — BASIC FLIGHT CRITERIA - PACK OFF			DATE 1 May 1949	
			REVISED	

ANALYSIS OF AIRPLANE-COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS

Condition PACK OFF - POWER OFF FLAPS & GEAR UP

V-n Diagram C. G. Location Gross Weight lbs.		PLAA MW 5 37015		PLAA MW 1 37880		
AIRPLANE						
ηZ_A		3.64		3.58		
W/S_W, #/sq. ft.		25.576		26.243		
$\eta_A W/S_W$		93.0966		93.9499		
C_{Z_A}		.51836		.52311		
$q = \eta_A (W/S_W)/C_{Z_A}$, #/sq. ft.		179.6		179.6		
$q/(W/S_W)$		7.02234		6.84376		
$V \sigma^{1/2} = \sqrt{391\,q}$, mph		265		265		
α T.L., degrees.		-1.32		-1.18		
C_{X_A}		.052363		.0518215		
$\eta X_A = C_{X_A} q/(W/S_W)$		.36771		.354654		
C_{L_A}		.52337		.52896		
C_{D_A}		.039862		.040391		
WING						
C_{L_W}		.573402		.585190		
C_{D_W}		.0208		.0215		
C_{Z_W}		.57275		.5846303		
C_{X_W}		.0340038		.033545		
$\eta Z_W = C_{Z_W} q/(W/S_W)$		4.022045		4.001069		
$\eta X_W = C_{X_W} q/(W/S_W)$		.238786		.2295739		
$Z_W = \eta Z_W W$, lbs.		148,876		151,961		
$X_W = \eta X_W W$, lbs.		8839		8719		
C_{M_W} (a.c.)		-.0630		-.0630		
$\frac{M_W}{W} = q S_W MAC_W C_{M_W}$, ft. lbs.		-229.636		-229.636		
ΔC_{M_A} due to wing		-.0650013		-.09196096		

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ANALYSIS OF AIRPLANE COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS

Condition PACK OFF - POWER OFF FLAPS & GEAR UP

V-n Diagram		$PLAA_{VH}$		$PLAA_{VH}$		
C. G. Location		5		1		
Gross Weight lbs.		37015		37980		
AIRPLANE						
ηz_A		3.64		3.58		
W/S_W, #/sq. ft.		25.576		26.243		
$\eta_A W/S_W$		93.0966		93.9499		
C_{Z_A}		.51836		.52311		
$q = \eta_A(W/S_W)/C_{Z_A}$, #/sq. ft.		179.6		179.6		
$q/(W/S_W)$		7.02234		6.84376		
$V\sigma^{1/2} = \sqrt{391\ q}$, mph		265		265		
α T.L., degrees		-1.32		-1.18		
C_{X_A}		.052363		.0518215		
$\eta x_A = C_{X_A}q/(W/S_W)$		.36771		.354654		
C_{L_A}		.52337		.52896		
C_{D_A}		.039862		.040391		
WING						
C_{L_W}		.573402		.585190		
C_{D_W}		.0208		.0215		
C_{Z_W}		.57275		.5846303		
C_{X_W}		.0340038		.033545		
$\eta z_W = C_{Z_W}q/(W/S_W)$		4.022045		4.001069		
$\eta x_W = C_{X_W}q/(W/S_W)$		.238786		.2295739		
$Z_W = \eta z_W W$, lbs.		148,876		151,961		
$X_W = \eta x_W W$, lbs.		8839		8719		
C_{M_W} (a.c.)		-.0630		-.0630		
$M_W = q S_W MAC_W C_{M_W}$, ft. lbs.		-229.636		-229.636		
ΔC_{M_A} due to wing		-.0650013		-.09196096		

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POWER OFF - FLAPS & GEAR UP

V-n Diagram		PLAA VH		PLAA VH		
C. G. Location		5		1		
Gross Weight lbs.		37015		37980		
α TL degrees		-1.32		-1.18		
NACELLE						
C_{L_N}		-.035446		-.035135		
C_{D_N}		.0050562		.005046		
C_{Z_N}		-.036845		-.036565		
C_{X_N}		.0043269		.0043879		
$\eta Z_N = C_{Z_N}\, q/(W/S_w)$		-.25873		-.25024		
$\eta X_N = C_{X_N}\, q/(W/S_w)$		.030384		.030029		
$Z_N = \eta Z_N W$, lbs.		-9577		-9504		
$X_N = \eta X_N W$, lbs.		1125		1141		
C_{M_N} (.25 L_f)		.021487		.021710		
$M_N = q\, S_w\, MAC_w\, C_{M_N}$ ft. lbs.		78,321		79,133		
ΔC_{M_A} due to nacelle		-.0048021		-.0029824		
BOOM (ONE)						
C_{L_B}		-.0000432		-.000032		
C_{D_B}		.0005309		.00530		
C_{Z_B}		-.0001404		-.0001079		
C_{X_B}		.005322		.005322		
$\eta Z_B = C_{Z_B}\, q/(W/S_w)$		-.0009859		-.0007384		
$\eta X_B = C_{X_B}\, q/(W/S_w)$		.037372		.036422		
$Z_B = \eta Z_B W$, lbs.		-36.4		-28.0		
$X_B = \eta X_B W$, lbs.		1383.2		1383		
C_{M_B} (.25 L_B)		.008665		.009033		
$M_B = q\, S_w\, MAC_w\, C_{M_B}$, ft. lbs.		31,584		32,925		
ΔC_{M_A} due to one boom		.00833662		.00851996		

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Subject: BASIC FLIGHT CRITERIA - PACK OFF

POWER OFF FLAPS & GEAR UP

V-n Diagram	PLAA VH		PLAA VH		
C. G. Location	5		1		
Gross Weight lbs.	37015		37980		
α_{TL} degrees	-1.32		-1.18		
VERTICAL TAIL (ONE)					
$C_{L_{VT}}$	0		0		
$C_{D_{VT}}$	.00057		.00057		
$C_{Z_{VT}}$	0		0		
$C_{X_{VT}}$	.00057		.00057		
$\eta X_{VT} = C_{X_{VT}} q/(W/S_W)$	.0040026		.0039009		
$X_{VT} = \eta X_{VT} W$, lbs.	148		148		
ΔC_{M_A} due to one tail	.0001556		.00013769		
HORIZONTAL TAIL					
α_{ht}	-3.16		-3.08		
q_{ht}/q	1.00		1.00		
δ_e	3.314		2.056		
$\eta_{ht} C_{L_{HT}}$	-.0144997		-.021032		
$C_{D_{HT}}$	.002248		.0021046		
$C_{Z_{HT}}$	-.0144997		-.021032		
$C_{X_{HT}}$	.002248		.0021046		
$\eta Z_{HT} = C_{Z_{HT}} q/(W/S_W)$	-.1018218		-.1439374		
$\eta X_{HT} = C_{X_{HT}} q/(W/S_W)$	.015786		.0144034		
$Z_{HT} = \eta Z_{HT} W$, lbs.	-3769		-5467		
$X_{HT} = \eta X_{HT} W$, lbs.	584		547		
$C_{M_{HT}}$ (a. c.)	-.0040497		-.0025124		
$M_{HT} = q S_W MAC_W C_{M_{HT}}$, ft lbs	-14,761		-9158		
SUMMATION					
Z_A, lbs.	135457		136934		
X_A, lbs.	13611		13469		
$C_{M_{A-T}}$	-.052879		-.0776281		

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.

Subject: BASIC FLIGHT CRITERIA - PACK OFF

ANALYSIS OF AIRPLANE COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS

Condition POWER ON-PACK OFF FLAPS & GEAR UP

V-n Diagram	PHAA	NHAA	$PLAA_{VH}$	$PLAA_{VD}$	$NLAA_{VH}$	$NLAA_{VD}$
C. G. Location	3	1	1'	1	7	7
Gross Weight lbs.	54,000	54,000	54,000	54,000	54,000	54,000
AIRPLANE						
n_{Z_A}	3.0	-1.5	3.0	2.425	-1.5	-1.079
W/S_W, #/sq. ft.	37.312	37.312	37.312	37.312	37.312	37.312
$n_A W/S_W$	111.936	-55.968	111.936	90.481	-55.968	-40.260
C_{Z_A}	1.49177	-.75236	.62325	.32234	-.31163	-.14343
$q = n_A(W/S_W)/C_{Z_A}$, #/sq. ft.	75.036	74.39	179.6	280.7	179.6	280.7
$q/(W/S_W)$	2.01164	1.99373	4.81345	7.52302	4.81346	7.52315
$V\sigma^{1/2} = \sqrt{391\ q}$, mph	171	170	265	331.8	265	331.8
α T.L., degrees	8.55°	-14.85°	-.215°	-3.45°	-10.19°	-8.43°
C_{X_A}	-.18449	-.2466	.01865	.06072	-.04169	.00018
$n_{X_A} = C_{X_A} q/(W/S_W)$	-.37102	-.49165	.08978	.45683	-.20093	.00135
C_{L_A}	1.50735	-.78780	.61613	.32829	-.31331	-.14286
C_{D_A}	.03686	.05274	.01577	.03012	.01945	.02246
WING						
C_{L_W}	1.4010	-.567	.65862	.37426	-.17845	-.02694
C_{D_W}	.1078	.0215	.0251	.0135	.008	.0068
C_{Z_W}	1.4014	-.55353	.65793	.39252	-.17218	-.02765
C_{X_W}	-.10222	-.12483	.0276	.03716	-.02283	.00275
$n_{Z_W} = C_{Z_W} q/(W/S_W)$	2.81827	-1.10358	3.16689	2.95294	-.82854	-.20800
$n_{X_W} = C_{X_W} q/(W/S_W)$	-.20557	-.24887	.19285	.27954	-.10988	.02070
$Z_W = n_{Z_W} W$, lbs.	152,187	-59,593	171,012	159,451	-44,741	-11,232
$X_W = n_{X_W} W$, lbs.	-11,101	-13,439	7174	15,095	-5,933	1,118
C_{M_W} (a.c.)	-.0630	-.0630	-.0630	-.0630	-.0630	-.0630
$M_W' = q\ S_W\ MAC_W\ C_{M_W}$, ft. lbs.	-95,718	-95,343	-229,582	-358,818	-229,582	-358,818
ΔC_{M_A} due to wing	-.12229	-.04603	-.09673	-.08080	-.07314	-.06391

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POWER ON - PACK OFF - FLAPS & GEAR UP

V-n Diagram	PHAA	NHAA	$PLAA_{VH}$	$PLAA_{VD}$	$NLAA_{VH}$	$NLAA_{VD}$
C. G. Location	3	1	1	1	7	7
Gross Weight, lbs.	54,000	54,000	54,000	54,000	54,000	54,000
α TL degrees	8.55°	17.88°	-.215°	-3.45°	-10.19°	-8.43°
WING SECTION IMMERSED IN SLIPSTREAM (ONE)						
$\Delta C_{L_{WI}}$	.0393	-.0096	.006	.002	-.0006	.00012
$\Delta C_{D_{WI}}$	.0086	.0051	0	-.0007	.00072	.00013
$\Delta C_{Z_{WI}}$	.0402	-.1165	.00575	.0017	-.0003	.0002
$\Delta C_{X_{WI}}$	.00485	.0031	.00025	-.0002	.00066	.00016
$\lambda Z_{WI} = \Delta C_{Z_{WI}} q/(W/S_W)$	.08084	-.02093	.02768	-.01427	-.00144	.00150
$\lambda X_{WI} = \Delta C_{X_{WI}} q/(W/S_W)$	.00975	.00618	.00120	-.00150	.00318	.00120
$\Delta Z_{WI} = \lambda \Delta Z_{WI}$ W, lbs.	4365.6	-1,130	1,495	771.9	-78	81
$X_{WI} = \lambda \Delta X_{WI}$ W, lbs.	526.88	333.7	65	-81.2	171.6	65
$\Delta C_{M_{WI}}$ (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	.00108	-.00005	.00009	.00002	-.00001	.00003
PROPELLER (ONE)						
C_{L_P}	.01266	-.02300	-.00018	-.00288	-.00846	-.00665
C_{D_P}	-.05432	-.05657	-.01416	-.00698	-.01288	-.00624
C_{Z_P}	.00444	-.0077	-.00012	-.00246	-.00605	-.00566
C_{X_P}	-.0556	-.0606	-.01416	-.00715	-.01417	-.00715
$\lambda Z_P = C_{Z_P}\ q/(W/S_W)$	.00893	-.01535	-.00059	-.01851	-.02912	-.04258
$\lambda X_P = C_{X_P}\ q/(W/S_W)$	-.11181	-.12082	-.06816	-.05375	-.06821	-.05379
$Z_P = \lambda Z_P$ W, lbs.	482.17	-829	-32	-999.4	-1,573	-2,299
$X_P = \lambda X_P$ W, lbs.	-6038.0	-6,524	-3,681	-2,903	-3,683	-2,905
ΔC_{M_A} due to one propeller	.00659	-.00156	.00125	-.00183	-.00544	-.00571

POWER ON - PACK OFF - FLAPS & GEAR UP

V-n Diagram	PHAA	NHAA	$PLAA_{VH}$	$PLAA_{VD}$	$NLAA_{VH}$	$NLAA_{VD}$
C. G. Location	3	1	1	1	7	7
Gross Weight lbs.	54,000	54,000	54,000	54,000	54,000	54,000
α TL degrees	8.55°	-14.88°	.215°	-3.45°	-10.19°	-8.43°
NACELLE						
C_{L_N}	.0068	-.1010	-.03420	.03948	-.07306	-.06311
C_{D_N}	.0060	.0087	.00500	.00520	.00651	.0060
C_{Z_N}	.0039	-.1045	-.03457	-.04213	-.04182	-.06249
C_{X_N}	.0051	-.0173	.00488	.00289	-.00615	-.00318
$\eta Z_N = C_{Z_N} q/(W/S_W)$	.00784	-.20834	-.16640	-.31692	-.34567	-.47011
$\eta X_N = C_{X_N} q/(W/S_W)$	.01026	-.03449	.02351	.02174	-.02958	-.02389
$Z_N = \eta Z_N W$, lbs.	424	-11,250	-8,986	-17,114	-18,667	-25,386
$X_N = \eta X_N W$, lbs.	554	-1863	1267	1174	-1597	-1290
C_{M_N} (.25 L_F)	.0208	.0062	.02393	.01759	.01002	.01163
$M_N = q S_W MAC_W C_{M_N}$, ft. lbs.	31,668	9,383	87,205	100,178	36,511	66,256
ΔC_{M_A} due to nacelle	.02482	-.07001	.00073	-.01127	-.04845	-.01883
BOOM (ONE)						
C_{L_B}	.00322	-.00413	0	-.00039	-.00456	-.00316
C_{D_B}	.00574	.00733	.00532	.00536	.00609	.00582
C_{Z_B}	.0040	-.0108	-.00002	-.00071	-.00561	-.00399
C_{X_B}	.0053	.0045	.00532	.00534	.00508	.00530
$\eta Z_B = C_{Z_B} q/(W/S_W)$	.00804	-.02153	-.00010	-.00536	-.02702	-.03005
$\eta X_B = C_{X_B} q/(W/S_W)$	.01066	.00897	.02517	.04016	.02447	.03990
$Z_B = \eta Z_B W$, lbs.	434.38	-1162.7	-5.6	-289	-1,457	-1,623
$X_B = \eta X_B W$, lbs.	575.59	484.5	1,359	2,167	1,321	2,154
C_{M_B} (.25 L_B)	.0407	-.0092	.01134	.00420	-.00447	-.00305
$M_B = q S_W MAC_W C_{M_B}$, ft. lbs.	61,966	-13,923	41,337	23,930	-18,104	-17,399
ΔC_{M_A} due to one boom	.03960	-.00419	.01081	.01382	-.00476	-.00307

POWER ON - PACK OFF - FLAPS & GEAR UP

V-n Diagram	PHAA	NHAA	$PLAA_{VH}$	$PLAA_{VD}$	$NLAA_{VH}$	$NLAA_{VD}$
C. G. Location	3	1	1	1	7	7
Gross Weight lbs.	54,000	54,000	54,000	54,000	54,000	54,000
α_{TL} degrees	8.55°	-14.[illegible]	-.215°	-2.45°	-10.17°	-8.43°
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
$\eta X_{VT} = C_{X_{VT}} q/(W/S_W)$	.00115	.00114	.00274	.00429	.00274	.00429
$X_{VT} = \eta X_{VT}$ W, lbs.	61.884	61.3	148	231.6	148	231.6
ΔC_{M_A} due to one tail	.00018	.00014	.00014	.00014	.00138	.00138
HORIZONTAL TAIL						
α_{ht}	1.10°	-11.15°	-2.54°	-4.42°	-8.44°	-7.36°
q_{ht}/q	1.18	1.00	1.034	1.0047	1.00	1.00
δ_e	-3.57°	14.18°	1.468°	4.059°	8.42°	7.78°
$\eta_{ht} C_{L_{HT}}$	-.01081	-.03633	-.01934	-.02380	-.03752	-.03345
$C_{D_{HT}}$	.00239	.00385	.00220	.00236	.00306	.00265
$C_{Z_{HT}}$	-.01081	-.03633	-.01734	-.02380	-.03952	-.03345
$C_{X_{HT}}$	.00239	.00385	.00220	.00236	.00306	.00285
$\eta Z_{HT} = C_{Z_{HT}} q/(W/S_W)$	.00725	-.07247	-.09307	-.17905	-.19023	-.25163
$\eta X_{HT} = C_{X_{HT}} q/(W/S_W)$	.00481	.00768	.01060	.01772	.01474	.02141
$Z_{HT} = \eta Z_{HT}$ W, lbs.	-1174	-3,913	-5,021	-9,667	-10,261	-13,574
$X_{HT} = \eta X_{HT}$ W, lbs.	260	414	572	9,567	796	1,156
$C_{M_{HT}}$ (a. c.)	.00515	-.01733	-.00132	-.00498	-.01066	-.00951
$M_{HT} = q\, S_W\, MAC_W\, C_{M_{HT}}$, ft lbs	7837.6	-26,217	-4825.6	-28,380	-38,840	-54,157
SUMMATION						
Z_A, lbs.	162,001	-80,999	159,920	131,655	-78,889	-57,874
X_A, lbs.	20,034	-26,591	4,797	15,102	-10,818	+ 760
$C_{M_{A-T}}$	-.03990	-.13377	-.07131	-.08777	-.14176	-.11997

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

DATE 1 May 1949

Subject BASIC FLIGHT CRITERIA - PACK OFF

REVISED

ANALYSIS OF AIRPLANE COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS
Condition FLAPS & GEAR UP - POWER ON GUST

V-n Diagram		PLAA V_H	PLAA V_H			
C. G. Location		5	1			
Gross Weight lbs.		37 015	37 980			
AIRPLANE						
$n Z_A$		3.732	3.668			
W/S_W , #/sq. ft.		25.576	26.2428			
$n A W/S_W$		95.4496	96.25859			
C_{Z_A}		.531456	.53596			
$q = nA(W/S_W)/C_{Z_A}$, #/sq. ft.		179.6	179.6			
$q/(W/S_W)$		7.0222	6.8437			
$V\sigma^{1/2} = \sqrt{391 q}$, mph		265	265			
α T.L., degrees		-1.265	-1.145			
C_{X_A}		.03137	.0310016			
$n X_A = C_{X_A} q/(W/S_W)$		.220286	.212165			
C_{L_A}		.5372	.54173			
C_{D_A}		.01916	.01976			
WING						
C_{L_W}		.57845	.58855			
C_{D_W}		.0210	.0215			
C_{Z_W}		.57787	.5880			
C_{X_W}		.033766	.033254			
$n Z_W = C_{Z_W} q/(W/S_W)$		4.05791	4.02409			
$n X_W = C_{X_W} q/(W/S_W)$		.237111	.22755			
$Z_W = n Z_W W$, lbs.		150,204	152,835			
$X_W = n X_W W$, lbs.		8,777	8,642			
C_{M_W} (a.c.)		-.0630	-.0630			
$M_W = q S_W MAC_W C_{M_W}$, ft. lbs.		-229,631	-229,631			
ΔC_{M_A} due to wing		-.065096	-.092182			

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DATE 1 May 1949

Subject: BASIC FLIGHT CRITERIA - POWER OFF

REVISED

V-n Diagram		PLAA V_H	PLAA V_H			
C. G. Location		5	1			
Gross Weight lbs.		37 015	37,980			
α TL degrees		-1.265	-1.145			
NACELLE						
C_{L_N}		-.03529	-.03515			
C_{D_N}		.005042	.0056379			
C_{Z_N}		-.03669	-.03656			
C_{X_N}		.0043675	.0043879			
$\eta Z_N = C_{Z_N}\, q/(W/S_W)$		-.25764	-.25021			
$\eta X_N : C_{X_N}\, q/(W/S_W)$		.030669	.030029			
$Z_N = \eta Z_N W$, lbs.		-9,537	-9,503			
$X_N = \eta X_N W$, lbs.		1,135	1,141			
C_{M_N} (.25 L_F)		.021596	.021874			
$M_N = q\, S_W\, MAC_W\, C_{M_N}$, ft. lbs.		78,716	79,729			
ΔC_{M_A} due to nacelle		-.004629	-.002815			
BOOM (ONE)						
C_{L_B}		-.000043	-.000032			
C_{D_B}		.00532	.00532			
C_{Z_B}		-.000014	-.000129			
C_{X_B}		.00532	.00532			
$\eta Z_B = C_{Z_B}\, q/(W/S_W)$		-.000983	-.0008828			
$\eta X_B = C_{X_B}\, q/(W/S_W)$		.037358	.036468			
$Z_B = \eta Z_B W$, lbs.		-36	-34			
$X_B = \eta X_B W$, lbs.		1383	1383			
C_{M_B} (.25 L_B)		.00881	.00902			
$M_B = q\, S_W\, MAC_W\, C_{M_B}$, ft. lbs.		32,112	32,877			
ΔC_{M_A} due to one boom		.00848168	.008511977			

Subject: BASIC FLIGHT CRITERIA - POWER OFF

V-n Diagram		PLAA V_H	PLAA V_H			
C. G. Location		5	1			
Gross Weight, lbs.		37,015	37,980			
α TL degrees		-1.265	-1.145			
WING SECTION IMMERSED IN SLIPSTREAM (ONE)						
$\Delta C_{L_{WI}}$		.0053	.0054			
$\Delta C_{D_{WI}}$		0	0			
$\Delta C_{Z_{WI}}$		.0051	.0053			
$\Delta C_{X_{WI}}$		.00008	.00009			
$h \Delta Z_{WI} = \Delta C_{Z_{WI}} q/(W/S_W)$		.035111	.034218			
$h \Delta X_{WI} = \Delta C_{X_{WI}} q/(W/S_W)$		.000562	.0006159			
$\Delta Z_{WI} = h \Delta Z_{WI}$ W, lbs.		1300	1300			
$\Delta X_{WI} = h \Delta X_{WI}$ W, lbs.		21	23			
~~$\Delta M_{WI\,(a.c.)}$~~						
~~$\Delta M_{WI} = q\, S_W\, MAC_W\, C_{M_{WI}}$,~~ ft. lbs.						
ΔC_{M_A} due to one wing section immersed in slipstream		.000298835	.000077123			
PROPELLER (ONE)						
C_{L_P}		-.00116	-.00101			
C_{D_P}		-.01413	-.01413			
C_{Z_P}		-.000849	-.00072629			
C_{X_P}		-.01415	-.01415			
$h Z_P = C_{Z_P}\, q/(W/S_W)$		-.005962	-.0049705			
$h X_P = C_{X_P}\, q/(W/S_W)$		-.099364	-.096838			
$Z_P = h Z_P$ W, lbs.		-221	-189			
$X_P = h X_P$ W, lbs.		-3,678	-3,678			
ΔC_{M_A} due to one propeller		.00006865	.00076325			

MODEL XC-120 PREPARED BY CHECKED BY APPROVED BY

DATE 1 May 1949

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Subject: BASIC FLIGHT CRITERIA - PACK OFF

V-n Diagram		PLAA V_H	PLAA V_H				
C. G. Location		5	1				
Gross Weight lbs.		37015	37980				
α_{TL} degrees		-1.265	-1.145				
VERTICAL TAIL (ONE)							
$C_{L_{VT}}$		0	0				
$C_{D_{VT}}$		.00057	.00057				
$C_{Z_{VT}}$		0	0				
$C_{X_{VT}}$		.00057	.00057				
$\eta X_{VT} = C_{X_{VT}} q/(W/S_W)$		.004002	.0039001				
$X_{VT} = \eta X_{VT} W$, lbs.		148	148				
ΔC_{M_A} due to one tail		.0001556	.0001377				
HORIZONTAL TAIL							
α_{ht}		-3.10	-3.05				
q_{ht}/q		1.029	1.030				
δ_e		3.326	2.17				
$\eta_{ht} C_{L_{HT}}$		-.014182	-.020643				
$C_{D_{HT}}$		.0096	.0097				
$C_{Z_{HT}}$		-.014182	-.020643				
$C_{X_{HT}}$		.0096	.0097				
$\eta Z_{HT} = C_{Z_{HT}} q/(W/S_W)$		-.09959	-.14127				
$\eta X_{HT} = C_{X_{HT}} q/(W/S_W)$		.0674	.06638				
$Z_{HT} = \eta Z_{HT} W$, lbs.		-3686	-5366				
$X_{HT} = \eta X_{HT} W$, lbs.		2495	2521				
$C_{M_{HT}}$ (a. c.)		-.004165	-.002780				
$M_{HT} = q S_W MAC_W C_{M_{HT}}$, ft lbs		-15,247	-10,133				
SUMMATION							
Z_A, lbs.		139,066	140,120				
X_A, lbs.		8155	8056				
$C_{M_{A-T}}$		-.05172	-.076191				

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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 conditions 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 Condition

The wing loads as determined in Part II-H-I 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-a.

Condition		Possibly Critical For
PHAA		
Speed	176.9	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.
Power	Off	
C. G. Location	3	
Gross Weight	54,000	
$PLAA_{VH}$		
Speed	265	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.
Power	Off	
C. G. Location	1	
Gross Weight	54,000	
$PLAA_{VD}$		
Speed	331.3	Rear spar, outer panel
Power	Off	
C. G. Location	1	
Gross Weight	54,000	
NHAA		
Speed	176.3	Lower surface in compression. Upper surface outer panel to center section splice.
Power	Off	
C. G. Location	1	
Gross Weight	54,000	

Subject: BASIC FLIGHT CRITERIA - PACK OFF

PART II-I-1 (Cont.)

Condition		Possibly Critical For
NLAA VD		Outer panel skin splices
Speed	331.3	
Power	Off	
C. G. Location	7	
Gross Weight	54,000	
PLAA VH gust		Local attachment of weight items
Speed	265	
Power	On	
C. G. Location	5	
Gross Weight	37,015	
PHAA		Special leading edge conditions
Speed		
Power	Off	
C. G. Location	1 & 3	
Gross Weight	54,000	
NHAA		Special leading edge conditions
Speed		
Power	Off	
C. G. Location	1 & 3	
Gross Weight	54,000	

Subject: BASIC FLIGHT CRITERIA - PACK OFF

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 PLAA 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 Conditions

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.

ANALYSIS OF AIRPLANE COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS
Condition SPECIAL LEADING EDGE - FLAPS & GEAR UP - POWER OFF

V-n Diagram	PHAA	NHAA					
C. G. Location	32%	19%					
Gross Weight lbs.	54000	54000					
AIRPLANE							
ηZ_A	3.0	-1.5					
W/S_W, #/sq. ft.	37.31	37.31					
$\eta_A W/S_W$	111.93	-55.96					
C_{Z_A}	1.789	-.860	Extrapolated				
$q = \eta A(W/S_W)/C_{Z_A}$, #/sq. ft.	62.56	65.06					
$q/(W/S_W)$	1.6767	1.7437					
$V\sigma^{1/2} = \sqrt{391\,q}$, mph	156.4	159.5					
α T.L., degrees	12.69	-16.55	$\alpha_{TL} = \frac{C_{L_W}}{.0842} - 8.13°$				
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	$C_{L_{W\,MAX}} \times 1.25$				
C_{D_W}	.210	.0298	Extrapolated				
C_{Z_W}	1.7561	-.6885	$C_{L_W} \cos \alpha_{T.L.} + C_{D_W} \sin \alpha_{T.L.}$				
C_{X_W}	-.1790	-.1736	$C_{D_W} \cos \alpha_{T.L.} - C_{L_W} \sin \alpha_{T.L.}$				
$\eta Z_W = C_{Z_W} q/(W/S_W)$	2.9444	-1.2005					
$\eta X_W = C_{X_W} q/(W/S_W)$	-.3001	-.3027					
$Z_W = \eta Z_W W$, lbs.	158.997	-64,827					
$X_W = \eta X_W W$, lbs.	-16,205	-16,345					
C_{M_W} (a.c.)	-.0630	-.0630					
$M_W = q S_W MAC_W C_{M_W}$, ft. lbs.	-79,987	-83,183					
ΔC_{M_A} due to wing							

PART II-I-2

2. **Flaps and Gear Down**

As noted in Part II-H-2 flapped condition 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 aileron 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, Asymmetrical Flight - Roll.

PART II-L

L. Flap Loads

It is required by reference (2)-6B that flap limit loading be determined from the formula:

$$\frac{W\#}{sq.\ ft.} = .002558 \left(\Delta C_L \underset{only}{flap}\right) \times V^2 \times \sigma$$

where (ΔC_L 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 is the prime determinant of the maximum flap load. 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. = 54,000 lbs. $\eta = 2.2739$ $T_c = .198$

$$C_{Z_A} = \frac{\eta \times W/S_w}{q} = \frac{(2.2739)(37.312)}{65.5} = 1.2953$$

$\alpha_{TL} = 2.21°$ C. G. 1 & 3

$C_{L_{WF}} = 1.212$ from figure 6 reference (1)

$C_{L_{WFI}} = .0775$ from figure 126 reference (1)

PART II-L (Cont.)

Inasmuch as the previous stated values ($C_{L_{WF}} = 1.212$ and $\Delta C_{L_{WFI}} = .0775$) for this pack-off condition are appreciably smaller than thosefor the pack-on condition shown on page 235, Vol. I, of this report, the flap loads will also be smaller. Therefore, the flap loads as determined in Part II-L, Vol. I of this report for the Pack-On condition, are considered applicable to the pack off condition.

<u>Flap loads outside of the slipstream</u>

107.39 lbs./sq. ft.

<u>Flap load in the slipstream</u>

159.6 lbs./sq. ft.

<u>Weighted Average Load on Flaps</u>

140.72 lbs./sq. ft.

PART II-M

M. Horizontal Tail Loads

1. Balance Conditions

Balancing tail loads were determined for all six points of the V-n diagrams for all C. G. locations and gross weights, flaps up and flaps down.

The maximum balance load was determined to be -13,574 lbs. for the following condition:

Gross Weight 54,000 lbs. $\bar{z}$ = -1.079

Flaps and gear up, power off $NLAA_{VD}$ 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

Gust loads on the horizontal tail are determined as specified in reference (2)-6B for minimum flying weight (37,015 lbs.) light weight alternate condition (37,980 lbs.) and design gross weight (54,000 lbs.).

The increments of horizontal tail load from a 50 ft. per second gust (50 K 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 stabilizer and elevators acting as a single unit. The effect of 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 as end plates neither of which is entirely applicable in the case of the XC-120, which has twin vertical tails with tips. 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-7b as the last term in the above equations. This will result in the following equation which will be used.

$$\Delta L_{HT} = \Delta F_{HT} = \frac{57.3}{V_H} U S_H \; q \; \frac{(1-52 \times a_w \times K_F \times K_T \times K_S)}{A} \left(C_{L\alpha_{ht}}\right)$$

$$\left(\frac{\pi A_H \quad a_{oH}}{\pi A_H + (57.3 \; a_{oH} \quad r)}\right) \text{ being equal to } C_{L\alpha_{ht}}$$

where:

$\Delta F_H = \Delta L_T$ = Increment of the horizontal tail load due to gust lbs.

V_H = Ture level flight high speed 265 mph = 389 fps

$U = \pm 50 \, K/\sigma^{\frac{1}{2}}$ K = Gust alleviation factor Fig. 5 reference (2)-2b.

PART II-M-2 (Cont.)

S_H = horizontal tail surface area = 346.2 sq. ft.

q = $V^2 \sigma / 391$ = dynamic pressure = 179.6 lbs./sq. ft.

a_w = .0842 = wing lift curve slope

K_P = downwash correction factor for horizontal tail position = .561 (from fig. 1, reference (2)-6B where l_{ht} = 47.5 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-7-b effective lift curve slope of horizontal tail.

Weight lbs.	S_w Sq.ft.	W/S_w	K	U	$\frac{57.3\ S_H\ q}{V_H}$	$\frac{1-52\ a_w K_P K_T K_S}{A}$	$\eta_{ht}\, C_{L_{\alpha ht}}$	ΔL_T
37015	1447.25	25.58	549	27.45	9150	.569	.0450	± 6431
37980	1447.25	26.24	550	27.50	9150	.569	.0450	± 6443
54000	1447.25	37.31	575	28.75	9150	.569	.0450	± 6736

The balancing tail loads for unaccelerated flight are obtained from Part II-D for 265 mph power off and power on, presented in figures 56 and 57 respectively.

For Unaccelerated Level Flight at 265 mph

$$C_L = \frac{391 \times G.W.}{\sigma \times S_w \times V^2}$$

PART II-M-2 (Cont.)

		C_L
For	37,015 lbs.	.1425
	37,980 lbs.	.146
	54,000 lbs.	.208

$\alpha_{T.L.}$ is obtained from figure 51 which is a plot of the balanced airplane lift coefficient versus angle of attack of the thrust line for 265 mph power off, and figure 53 for power on at 265 mph.

$L_{HT} = \eta_{ht} C_{LHT}\; q \frac{q_{ht}}{q} S_{ht}$

PACK OFF

FLAPS & GEAR UP - POWER OFF V_{TO} = 265 MPH

	CG 1		CG 2		CG 3		CG 4		CG 5	
δ_{STL}	$\eta_{ht} C_{LHT}$	L_{HT}	$\eta_{ht} C_{LHT}$	L_{HT}	$\eta_{ht} C_{LHT}$	L_{HT}	$\eta_{ht} C_{LHT}$	L_{HT}	$\eta_{ht} C_{LHT}$	L_{HT}
-12	-.1287	-8002.2	-.1710	-10632.3	-.1332	-8282.0	-.1756	-10918.4	-.1537	-9556.7
-8	-.1109	-6895.5	-.1194	-7424.0	-.1097	-6820.9	-.1182	-7349.4	-.1148	-7138.0
-4	-.0988	-6143.1	-.0720	-4476.8	-.0937	-5828.5	-.0683	-4246.7	-.0819	-5092.3
0	-.0831	-5167.0	-.0248	-1542.0	-.0798	-4961.8	-.0199	-1237.3	-.0496	-3084.0
4	-.0674	-4190.8	+.0240	+1492.3	-.0674	-4190.8	.0241	1498.5	-.0175	-1088.1
8	-.0572	-3556.6	.0673	4184.5	-.0626	-3892.3	.0616	3830.1	.0086	534.7

FLAPS & GEAR UP - POWER ON V_{TO} = 265 MPH

	CG 1		CG 2		CG 3		CG 4		CG 5	
q_{ht}/q	$\eta_{ht} C_{LHT}$	L_{HT}	$\eta_{ht} C_{LHT}$	L_{HT}	$\eta_{ht} C_{LHT}$	L_{HT}	$\eta_{ht} C_{LHT}$	L_{HT}	$\eta_{ht} C_{LHT}$	L_{HT}
+1.000	-.1416	-8804.3	-.1866	-11602.3	-.1483	-9220.9	-.1930	-12000.3	-.1685	-10476.9
+1.000	-.1187	-7380.5	-.1288	-8008.5	-.1190	-7399.1	-.1290	-8020.9	-.1240	-7710.0
1.000	-.1010	-6279.9	-.0739	-4594.9	-.0968	-6018.8	-.0725	-4507.9	-.0846	-5260.2
1.035	-.0800	-5150.	-.0224	-1441.0	-.0768	-4950	-.0180	-1157	-.0430	-2761.0
1.046	-.0555	-3600	.0338	2200	-.0571	-3710	.0321	2082	-.0078	-507.0
1.050	-.0398	-2598	.0815	5320	-.0466	-3040	.0750	4900	.0240	1569

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DATE 1 May 1949

Subject:- BASIC FLIGHT CRITERIA - PACK OFF

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PACK OFF

FLAPS AND GEAR UP POWER OFF

$V\sqrt{\sigma} = 265$ mph

C.G. 7

α_{TL}	$\eta_{ht} C_{L_{HT}}$	L_{HT}
-12	-.1853	-11510
-8	-.1233	-7670
-4	-.06577	-4090
0	-.00567	-352
4	.04826	30[illegible]
8	.1014	6310

FLAPS AND GEAR UP POWER ON $V\sqrt{\sigma} = 265$ mph

C.G. 7

α_{TL}	$\eta_{ht} C_{L_{HT}}$	L_{HT}
-12	-.2012	-12,560
-8	-.1324	-8230
-4	-.0678	-4210
0	-.00392	-252
4	.0527	3752
8	.1155	7540

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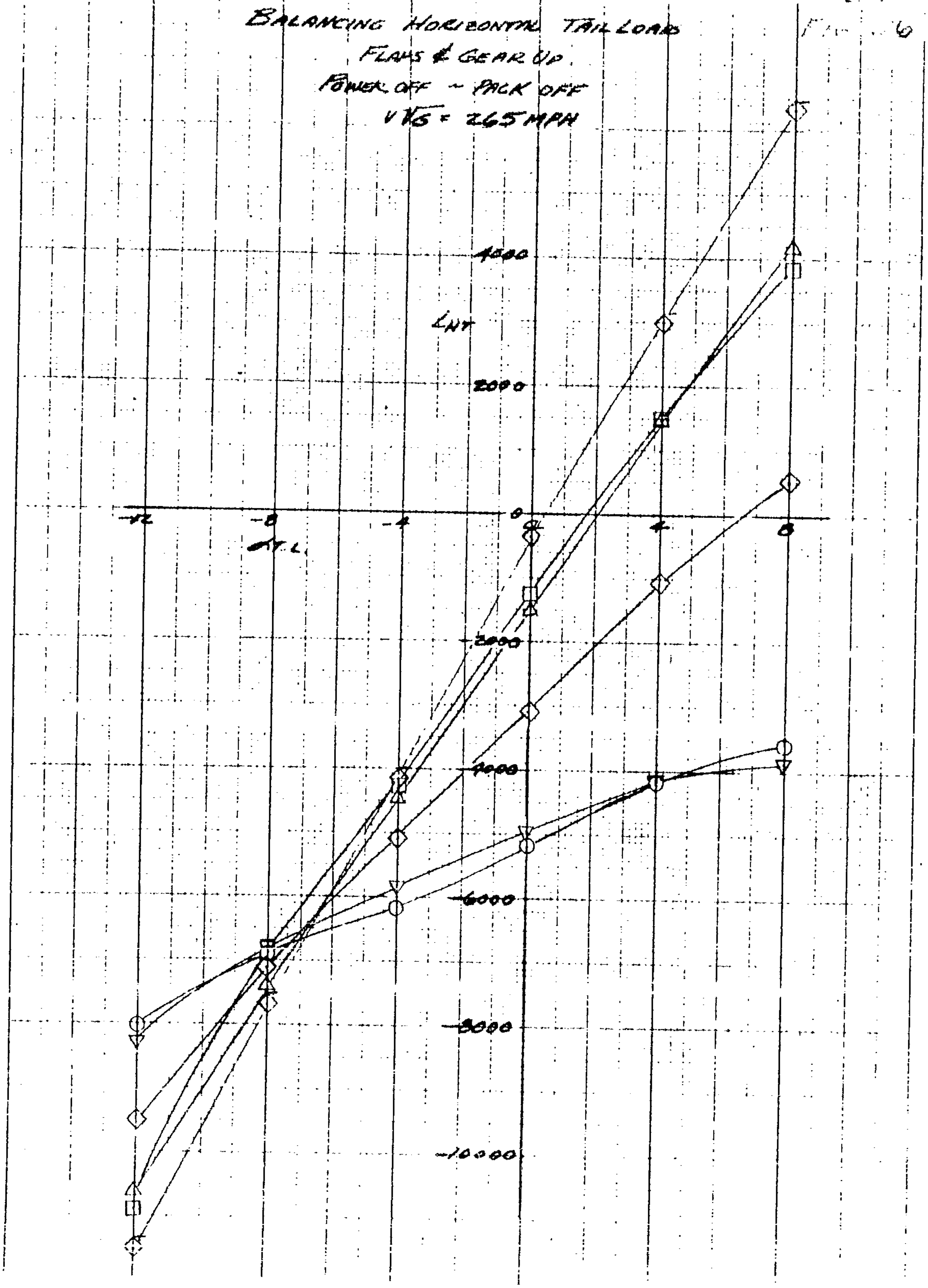
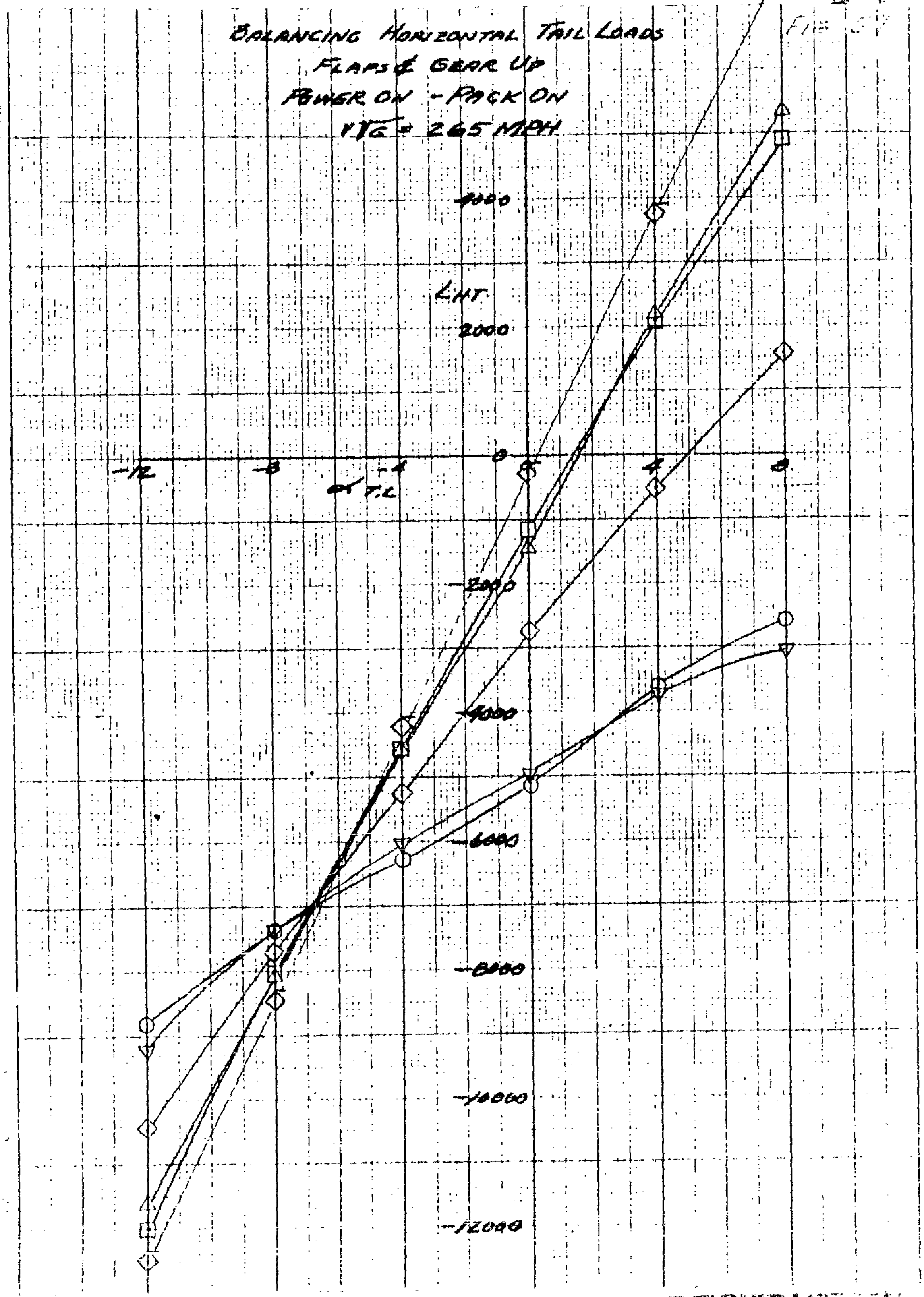

BALANCING HORIZONTAL TAIL LOAD
FLAPS & GEAR UP.
POWER OFF ~ PACK OFF
$V\sqrt{\sigma}$ = 265 MPH
L_{HT}
4000
2000
0
-2000
-4000
-6000
-8000
-10000
-12
-8
-4
4
8
$\alpha_{T.L.}$


BALANCING HORIZONTAL TAIL LOADS
FLAPS & GEAR UP
POWER ON - PACK ON
V√G = 265 MPH
LHT
4000
2000
0
-2000
-4000
-6000
-8000
-10000
-12000
-12
-8
-4
4
8
α T.L

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PART II-M-2b

The balanced airplane airplane lift coefficient for flaps and gear up, power off was determined in Part II-F and is shown in figure 57 versus TL.

Similar calculations for the power on condition at 265 mph are presented on the following pages and shown in figure 58 versus TL.

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PROPELLER

$\sqrt{\sigma}V = 265$ MPH $T_C = .047$

$\alpha_{T.L}$	$\cos\alpha$	$\sin\alpha$	$C_{Z(Fp)}$	$C_{X(Fp)}$	$C_Z\cos\alpha$	$C_X\sin\alpha$	C_L (1 EA.)	$\frac{C_{Lp}}{(S_w)}$	$\underline{2C_{Lp}}$
-12	.9781	-.2079	-.0580	-.1170	-.0567	.0243	-.0810	-.00997	-.01994
-8	.9903	-.1392	-.0396		-.0392	.0163	-.0555	-.00683	-.01366
-4	.9976	-.0698	-.0192		-.0192	.0082	-.0274	-.00337	-.00674
0	1.0000	0	0		0	0	0	0	0
4	.9976	.0698	.0192		.0192	-.0082	.0274	.00337	.00674
8	.9903	.1392	.0396		.0392	-.0163	.0555	.00683	.01366

$\frac{F_p}{S_w} = .12313 \times C_L$

FORMULA:

$C_L = C_Z \cos\alpha_{T.L} - C_X \sin\alpha_{T.L}$

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Subject: BASIC FLIGHT CRITERIA - PACK OFF

$$C_{LHT} = \eta_{ht} C_{LNT} \frac{S_{HT}}{S_W} \cdot \frac{q_{ht}}{q} = \eta_{ht} C_{LNT} \cdot .2392 \frac{q_{ht}}{q}$$

AIRPLANE LIFT COEFFICIENT ~ POWER ON
FLAPS & GEAR UP - 265 MPH $T_c = .047$

						CG ①			CG ②	
$\alpha_{T.L.}$	C_{LA-T} Power Off	ΔC_{LWT} (2)	C_{Lp} (2)	C_{LA-T} Power On	$\frac{q_{ht}}{q}$	$\eta_{ht} C_{LNT}$	C_{LHT}	C_{LA}	$\eta_{ht} C_{LHT}$	C_{LHT}
-12	-.4221	-.0028	-.0199	-.4448	1.000	-.1416	-.0339	-.4787	-.1866	-.0446
-8	-.0552	.0012	-.0137	-.0677	1.000	-.1187	-.0284	-.0961	-.1288	-.0308
-4	.3052	.0066	-.0067	.3051	1.000	-.1010	-.0242	.2809	-.0739	-.0177
0	.6507	.0122	0	.6629	1.035	-.0800	-.0198	.6431	-.0224	-.0056
4	.9957	.0182	.0067	1.0206	1.046	-.0555	-.0139	1.0067	.0338	.0085
8	1.3639	.0238	.0137	1.4014	1.050	-.0398	-.0100	1.3914	.0815	.0205

	CG ③			CG ④			CG 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}
-.5331	-.1493	-.0355	-.4803	-.1930	-.0462	-.4910	-.1685	-.0403	-.4851
-.0985	-.1190	-.0285	-.0962	-.1290	-.0309	-.0986	-.1240	-.0297	-.0974
.2874	-.0968	-.0232	.2819	-.0725	-.0173	.2878	-.0846	-.0202	.2849
.6573	-.0768	-.0190	.6439	-.0180	-.0045	.6584	-.0430	-.0107	.6522
1.0271	-.0571	-.0143	1.0063	.0321	.0081	1.0287	-.0078	-.0020	1.0186
1.4219	-.0466	-.0117	1.3837	.0750	.0188	1.4202	.0240	.0060	1.4074

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DATE 1 May 1949

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AIRPLANE LIFT COEFFICIENT - POWER ON

FLAPS & GEAR UP : 265 MPH $T_C = .047$

	CG 7		
$\delta_{T.C}$	$\Delta C_{L_{HT}}$	$C_{L_{HT}}$	C_{L_A}
-12	-.2012	-.0480	-.4928
-8	-.1324	-.0315	-.0992
-4	-.0678	-.0162	.2889
0	-.00392	-.00017	.66193
4	.0577	.0144	1.0350
8	1155	.0290	1.4304

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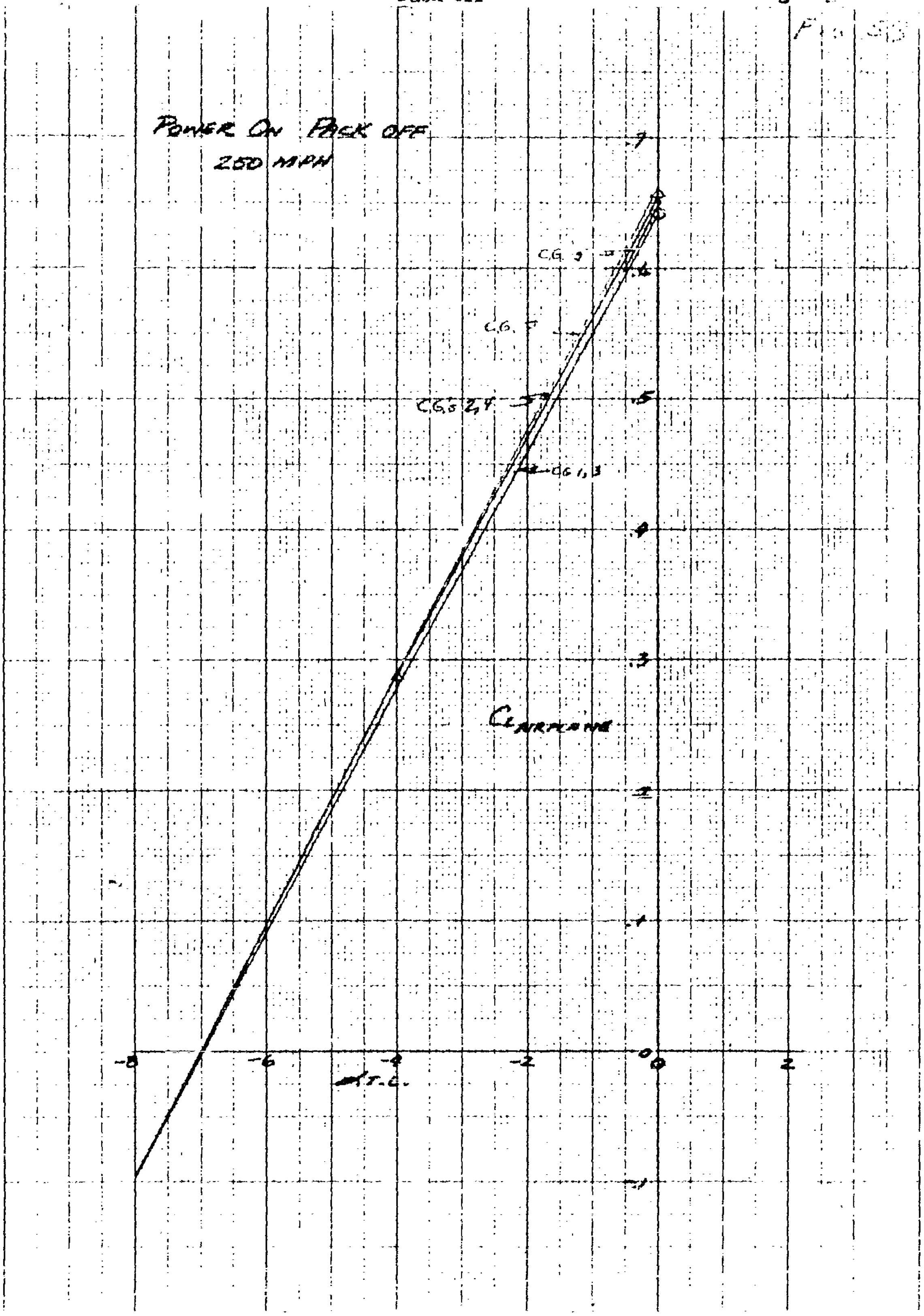
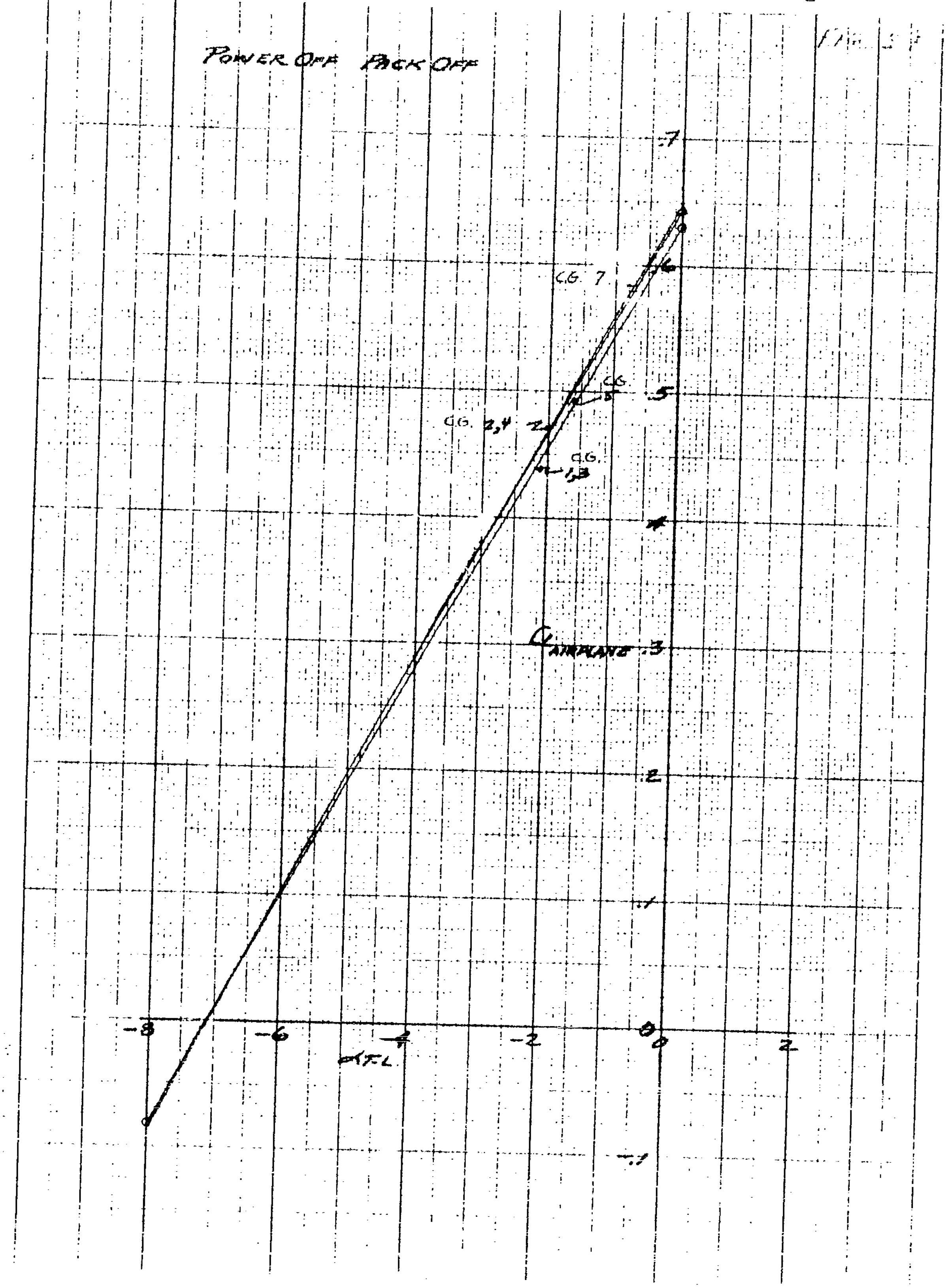

POWER ON PACK OFF
250 MPH
C.G. 5
C.G. 7
C.G.s 2,4
C.G. 1,3
$C_{L_{AIRPLANE}}$
.7
.6
.5
.4
.3
.2
.1
0
-.1
-8
-6
-4
-2
0
2


POWER OFF PACK OFF
C.G. 7
C.G. 5
C.G. 2,4
C.G. 1,3
$C_{L\,AIRPLANE}$
.7
.5
.4
.3
.2
.1
-.1
-8
-6
-4
-2
0
2
α_{FL}

PART II-M-2c

2c. Total Horizontal Tail Loads with Gust

Power Off

Gross Weight lbs.	C_{L_A}	C. G.	α T. L.	Balancing Load	Gust Load	Total Load
37,015	.1425	1	-5.54	-6410	± 6431	-12841
		2	-5.58	-5590		-12021
		3	-5.54	-6210		-12641
		4	-5.58	-5230		-11661
		5	-5.56	-5820		-12251
		7	-5.58	-5500		-11931
37,980	.146	1	-5.5	-6410	± 6443	-12853
		2	-5.55	-5600		-12043
		3	-5.50	-6190		-12633
		4	-5.55	-5190		-11633
		5	-5.53	-5810		-12253
		7	-5.55	-5480		-11923
54,000	.208	1	-4.79	-6300	± 6736	-13036
		2	-4.86	-5070		-11806
		3	-4.79	-6020		-12756
		4	-4.86	-4750		-11486
		5	-4.83	-5470		-12206
		7	-4.86	-4870		-11606

Power On

Gross Weight lbs.	C_{L_A}	C. G.	α T. L.	Balancing Load	Gust Load	Total Load
37,015	.1425	1	-5.46	-6650	± 6431	-13081
		2	-5.54	-5960		-12391
		3	-5.46	-6500		-12931
		4	-5.54	-5880		-12311
		5	-5.51	-6150		-12581
		7	-5.54	-5880		-12311
37,980	.146	1	-5.44	-6660	± 6443	-13103
		2	-5.50	-5930		-12373
		3	-5.44	-6500		-12943
		4	-5.50	-5840		-12283
		5	-5.47	-6120		-12563
		7	-5.54	-5840		-12283
54,000	.208	1	-4.74	-6480	± 6736	-13216
		2	-4.84	-5350		-12086
		3	-4.74	-6280		-13016
		4	-4.84	-5280		-12016
		5	-4.79	-5680		-12416
		7	-4.84	-5200		-11936

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 $\alpha_{TL} = -4.74$

balancing $\eta_{ht} \, C_{L_{HT}} = -.104$ figure 38

balancing $\alpha_{ht} = -5.18$ figure 65 reference (1)

$$\delta_e = \frac{-.104 + .2332}{.0246} = 5.25$$

$$\eta_{ht} \, C_{L_{HT}} = L_{HT}/q \times q_{ht}/q \times S_{ht}$$

$$-.21255 = -13216/179.6 \times 346.2$$

$$\therefore \alpha_{ht} = \frac{(-\delta_e)(+.0246) + \eta_{ht} \, C_{L_{HT}}}{.0450}$$

$$\alpha_{ht} = \frac{-5.25 \times .0246 - .21255}{.0450}$$

$$\alpha_{ht} = -7.593$$

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PART II-M-3

3. Maneuvering Loads

The maximum maneuvering tail load for design gross weight of 54,000 lbs. at the maneuvering speed ($V\sqrt{\sigma}$ = 167 mph) and at the maneuvering load factor (3.0 n) was determined in reference (13) and is 10272 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 the addition of the incremental angle of attack of the horizontal tail and the incremental elevator deflection obtained in maneuvering, to the horizontal tail angle of attack and elevator deflection for the initial balance condition.

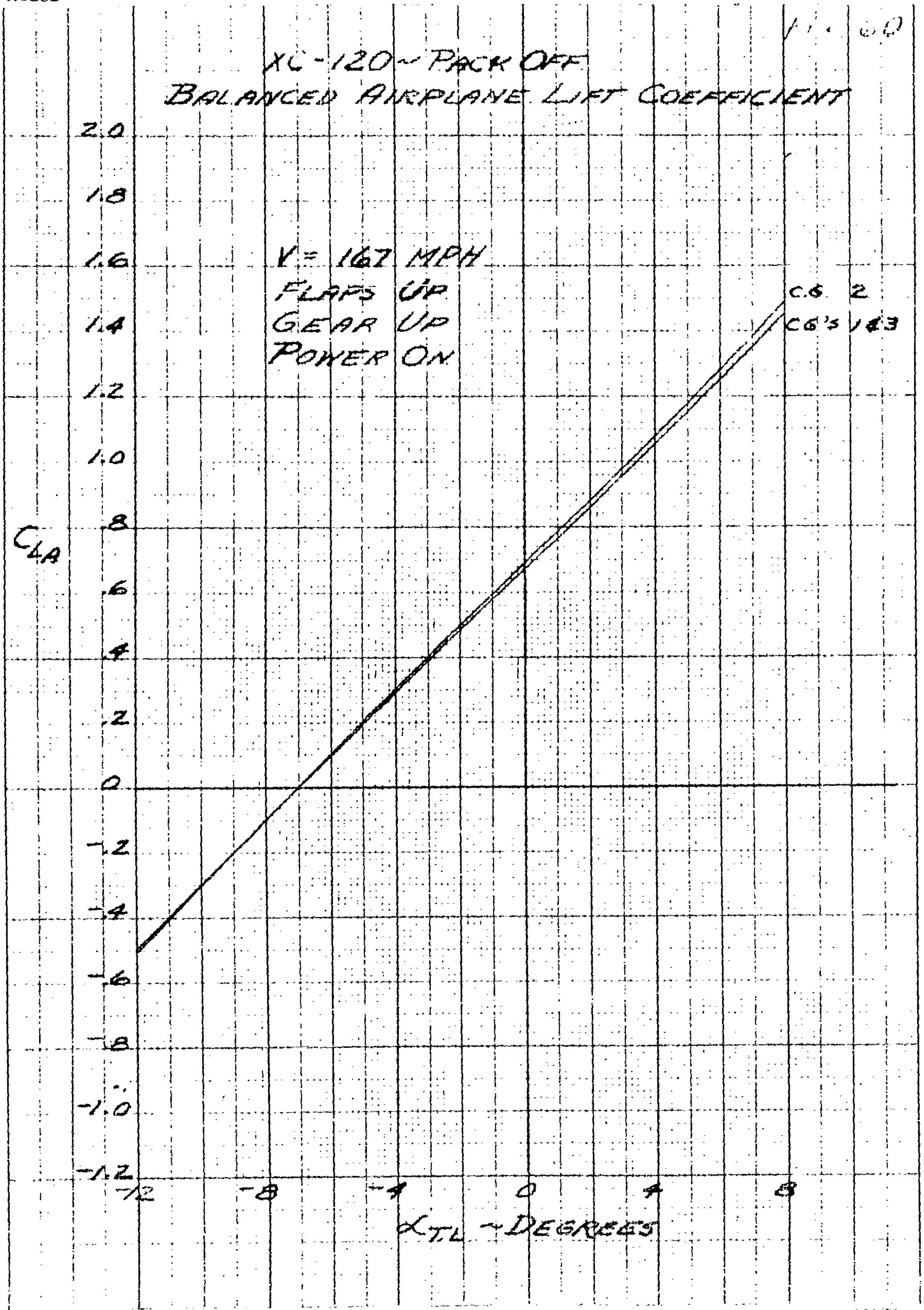
The variation of balanced airplane lift coefficient with angle of attack of the thrust line for $V\sqrt{\sigma}$ = 167 mph, flaps and gear up, power on is determined in the following table for various center of gravity locations.

BALANCED AIRPLANE LIFT COEFFICIENT

FLAPS & GEAR UP $V_{TG} = 167$ MPH POWER ON

20% UPPER CG ①						28% UPPER CG ②		
$\alpha_{T.L.}$	$\frac{S_{ht}}{S_w}\frac{q_{ht}}{q}$	$\eta_{ht} C_{LHT}$	C_{LHT}	C_{LA-T} Power On	C_{LA}	$\eta_{ht} C_{LHT}$	C_{LHT}	C_{LA}
-12	.2392	-.131	-.0313	-.4649	-.4962	-.175	-.0419	-.5068
-8	.2392	-.108	-.0258	-.0710	-.0968	-.115	-.0275	-.0985
-4	.2490	-.086	-.0214	.3186	.2972	-.059	-.0147	.3039
0	.2722	-.062	-.0169	.6943	.6774	-.006	-.0016	.6927
4	.2780	-.042	-.0117	1.0643	1.0526	+.043	.0120	1.0763
8	.2808	-.028	-.0079	1.4623	1.4544	.085	.0239	1.4862

20% LOWER CG ③					
$\alpha_{T.L.}$	$\frac{S_{ht}}{S_w}\frac{q_{ht}}{q}$	$\eta_{ht} C_{LHT}$	C_{LHT}	C_{LA-T} Power On	C_{LA}
-12	.2392	-.143	-.0342	-.4649	-.4991
-8	.2392	-.115	-.0275	-.0710	-.0985
-4	.2490	-.090	-.0224	.3186	.2962
0	.2722	-.066	-.0180	.6943	.6763
4	.2700	-.049	-.0136	1.0643	1.0507
8	.2808	-.039	-.0110	1.4623	1.4513


XC-120 ~ PACK OFF
BALANCED AIRPLANE LIFT COEFFICIENT
V = 167 MPH
FLAPS UP
GEAR UP
POWER ON
C.G. 2
CG'S 1 & 3
C_{L_A}
$\alpha_{T.L}$ ~ DEGREES

PART II-M-3

<u>For the initial balance condition (power on)</u>

$V\sqrt{\sigma}$ = 167 mph Gross Weight = 54,000 lbs.

C_{L_A} = .5231

C. G. Location	α_{TL}	α_{ht}	$\eta_{ht} C_{L_{HT}}$	δ_e
1	-1.6	-3.30	-.0717	3.13°
2	-1.7	-3.35	-.0284	4.98°
35% MAC_w Upper	-1.7	-3.35	-.0095	6.52
3	-1.6	-3.30	-.0757	2.98

α_{ht} is from figure 63, reference (1)

$\eta_{ht} C_{L_{HT}}$ is interpolated from figures 3 and 31

$$\delta_e = \frac{\eta_{ht} C_{L_{HT}} - .0450\,\alpha_{ht}}{.0246}$$

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	$\Delta\alpha_{ht}$	$\Delta\delta_e$
35% MAC_w Upper	4.08	7.276
3	4.70	-16.88

The total values for the critical condition are then

C. G. Location	α_{ht}	δ_e
35% MAC_w Upper	.73	13.796
3	1.40	-13.90

PART II-M-4

4. Summary of Design Horizontal Tail Loads

Condition	NLAA V_D n = -1.079 Power Off	NLAA V_D n = -1.079 Power On	Gust 50 fps Power On	Maneuvering n = 3.0 Power On	
Gross Wt.	54,000	54,000	54,000	54,000	54,000
$V\sqrt{\sigma}$ = mph	331.3	331.3	265	167	167
C. G. Loc.	7	7	1	35% MAC_w Upper	3
Load, lbs.	-12,526	-13,574	-13,216	10,272	-7,687
α_{ht}, deg.	-7.44	-7.36	-7.593	.73	1.40
δ_e, deg.	8.36	7.78	5.25	13.796	-13.90
q_{ht}/q,	1.00	1.00	1.00	1.117	1.117

PART II-N

1. Elevator Trim Tab

As the maximum tab load will occur at maximum surface deflection at V_D (331.3 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:

$$H_{ett} = \frac{d\,C_{H_{ett}}}{d\,\delta_{ett}} \; \delta_{ett} \times q \times \frac{q_{ht}}{q} \times \delta_{ett} \times c_{ett}$$

$S_{ett} = 5.44$ sq. ft. $\qquad c_{ett} = .5$ ft.

δ_{ett} maximum $= 25^{\circ}$ $\qquad$ Page 15 reference (1)

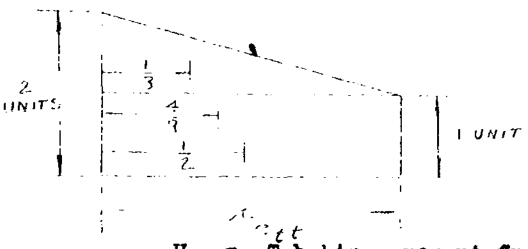
$q = 280.7$ $\qquad \frac{q_{ht}}{q} = 1.00$

$H_{ett} = -.006 \times 25 \times 280.7 \times 1.00 \times 5.44 \times .5$

$H_{ett} = -114$ ft. lbs.

Using the following load distribution as specified in reference (2) the tab load is determined.

Centroid of Loading:



$-y\,N_t = H_t$

$N_t = \frac{H_t}{\frac{4}{9}\,.5'} = \frac{H_t}{-.222} = -4.5\,H_t$

H_t = Tab hinge moment fr./lbs.

N_t = Normal force on Tab lbs.

y = Force Arm

A = Area

PART II-N

Therefore,

$$N_{e_{tt}} = -4.5 \times H_{e_{tt}} = -4.5 \times -114$$

$$= 513 \text{ lbs.}$$

$$\frac{513}{5.44 \text{ sq. ft.}} = 94.30 \text{ lbs./sq. ft. of tab area}$$

PART II-N-2

2. Elevator Spring Tab

Since the elevator trim tab investigation showed that the tab angles and loads were relatively small it was decided to determine the elevator spring tab load at maximum deflection and at V_D (331.3 mph) for the design load. This is a conservative procedure.

The elevator spring tab hinge moment is determined as follows:

$$H_{e_{st}} = \frac{d\,C_{H_{e_{st}}}}{\delta_{e_{st}}} \; \delta_{e_{st}} \times q \times \frac{q_{ht}}{q} \times S_{e_{st}} \times c_{e_{st}}$$

$S_{e_{st}}$ = 3.83 sq. ft. $\qquad c_{e_{st}}$ = .5 ft.

$\delta_{e_{st_{maximum}}} = 22^{\circ}$ page 15 reference (1)

q = 280.7 $\qquad \frac{q_{ht}}{q}$ = 1.00

$H_{e_{st}} = -.006 \times 22 \times 280.7 \times 1.00 \times 3.83 \times .5$

$H_{e_{st}} = -70.85$ ft. lbs.

Using the same load distribution as for the elevator trim tab

$N_{e_{st}} = -4.5 \times H_{e_{st}}$

$N_{e_{st}} = -4.5 \times -70.85$

$N_{e_{st}} = 318.8$ lbs.

$\frac{318.8}{3.83} = 83.24$ lbs./sq. ft. of tab area

PART III

III. ASYMMETRICAL FLIGHT - YAW

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 yaw. The method of analysis and method of combination is the same as presented in Part III-A of Volume I of this report, with the crew nacelle being substituted for the fuselage. Crew nacelle is designated by subscripts N or n as C_{L_N} - crew nacelle lift coefficient.

1. Angular Relationships

The angular relationship of each component part of the airplane is the same as presented in Volume I. For convenience they are again presented here with the addition of the angular relationship of the crew nacelle.

$$\psi_w = \psi_n = \psi_b = \psi_p = \psi_{ht}$$

$$\psi_{vt_{right}} = \psi_w - \sigma_{p_{ave\ right}} - \frac{d\sigma}{d\psi}\psi$$

$$\psi_{vt_{left}} = \psi_w - \sigma_{p_{ave\ left}} - \frac{d\sigma}{d\psi}\psi$$

PART III-B

B. Yawing Moments of Component Parts

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

The contribution of each part is found by the equations developed in Part III-A of Volume I of this report.

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

The final yawing moment coefficients of the component parts are based on wing area, wing span and free stream dynamic pressure.

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 ($V\sqrt{\sigma}$ = 331.3 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 249	Page 249
Wing Flapped	251	251
Crew Nacelle	252	252
Booms	253	253
Horizontal Tail	255	255
Landing Gear	255	255
Propellers	Figure 61	256
Slipstream Effects	62	260
Vertical Tail	63 - 66	Figure 67

PART III-B-1

1. Wing Unflapped

Zero Yaw Single Engine Operation

For $\psi = 0°$, $C_{N_W} = 0$, Slipstream effects are covered in Part III-B-8. The foregoing holds for all C. G. positions.

5° Yaw V_D $V\sigma^{\frac{1}{2}} = 331.3$ mph

$$C_{N_W} = C_{N_{W_{AC}}} + C_{Y_W}\left(\frac{x_W}{B_W} - \frac{A.C._W}{B_W}\right)$$

$$\frac{A.C._W}{B_W} = \frac{.256 \times 168.28}{109.27 \times 12} = .0329$$

C. G. Position	x_W	x_W/B_W	$\left(\frac{x_W}{B_W} - \frac{A.C._W}{B_W}\right)$
1 & 3	33.66	.0257	-.0072
2 & 4	47.10	.0360	+.0031
5	41.02	.0313	-.0016
7	50.48	.0385	+.0056

For design Gross Weight of 54,000 lbs.

$$C_{L_W} = \frac{54,000}{.00256 \times 1447.24(331.3)^2} = .133$$

At this speed the airplane is flying at an

$\alpha_{TL} = -6.25°$ figure 2, reference (1)

Assuming that C_{D_W} in yaw = C_{D_W} in pitch at

$\alpha_{TL} = -6.47°$

then C_{D_W} in yaw = .0078 figure 3, reference (1)

PART III-B-1

Wing Unflapped (Cont.)

$$C_{Y_W} = \left(C_{D_W} \text{ in yaw}\right) \cdot \sin \psi$$

$$= .0078(.0872)$$

$$= .000699$$

$C_{N_{W_{AO}}}$ = .00006 figure 5, reference (1)

C. G. position	C_{Y_W} x (3)	C_{N_W}
1 & 3	-.00001	.00005
2 & 4	negligible	.00006
5	"	.00006
7	"	.000064

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. Crew Nacelle

Zero Yaw Single Engine Operation

For $\psi = 0$; $C_{N_N} = 0$

5° Yaw V_D $V\sigma^{\frac{1}{2}} = 331.3$ mph

$$C_{N_N} = C_{N_{N.25L_n}} \frac{S_n}{S_w} \frac{L_n}{B_w} + C_{Y_N} \frac{S_n}{S_w} \left(\frac{X_n}{B_w} - \frac{.25\ L_n}{B_w} \right)$$

$C_{N_{N.25L_n}} = .009$ figure 24 reference (1)

$\frac{S_n}{S_w} = \frac{271}{1447.24} = .187$ $\frac{L_n}{B_w} = \frac{50.2}{109.27} = .461$

$$C_{Y_N} = C_{C_N} \cos\psi + C_{D_{N_{yaw}}} \frac{F_n}{S_n} \sin\psi$$

$C_{C_N} = .0238$ figure 25 reference (1)

$C_{D_N} = .1313$ figure 80 reference (1)

For small values of ψ C_{D_N} yaw is assumed equal to C_{D_N} at the corresponding α_{TL}.

Then

$$C_{Y_N} = (.0238)(.9962) + (.1313)\left(\frac{58.8}{271}\right)(.0872)$$

$$= .0262$$

$$C_{N_N} = (.009)(.187)(.461) + (.0262)(.187)\left(\frac{x_n}{B_w} - .115\right)$$

$$= .00078 + .0049\left(\frac{x_n}{B_w} - .115\right)$$

C. G. position	x_n/B_w	C_{N_N}
1 & 3	.2045	.00044
2 & 4	.2145	.00049
5	.211	.00047
7	.2173	.000501

PART III-B-4

4. Booms

Zero Yaw Single Engine Operation

For $\psi = 0^o$, the equation:

$$C_{N_B} = 2\left[C_{N_{B.25L_b}} \frac{S_b}{S_w}\frac{L_b}{B_w} + C_{Y_B}\frac{S_b}{S_w}\left(\frac{x_b}{B_w} - \frac{.25\,L_b}{B_w}\right)\right]$$

equals zero, however with single engine operation a yawing moment is caused by the cowl flaps of the operating engine being open while those of the dead engine are closed.

So that for zero yaw, single engine operation

$$C_{N_B} = \Delta C_{D_{cowl\ flaps}} \frac{y_b}{B_w}$$

where $\Delta C_{D_{cowl\ flaps}}$ is the drag coefficient of the open cowl flaps based on wing area.

$$\Delta C_{D_{cowl\ flaps}} = .0037 \quad \text{reference (10)}$$

For all C. G. locations:

$$C_{N_B} = .0037 \times \frac{14.59}{109.27} = .00048$$

5^o Yaw $\quad V_D \quad V\sigma^{\frac{1}{2}} = 331.3$ mph

From figure 30 reference (1)

$$C_{N_{B.25L_b}} = .0096$$

$$C_{Y_B} = C_{C_B}\cos\psi + C_{D_{B\ yaw}}\frac{F_b}{S_b}\sin\psi$$

$$C_{D_{B\ yaw}} = C_{D_{B\ pitch}} \quad \text{for corresponding } \alpha_{TL}$$

From figures 26 and 27 reference (1)

$$C_{D_B} = .1938 \quad \text{figure 80 reference (1)}$$

$$C_{D_{B\ yaw}} = .1938 \times \frac{40.76}{363.2} = .0228$$

PART III-B-4

Booms (Cont.)

From figure 31, reference (1)

$$C_{D_B} = .0055$$

$$C_{Y_B} = (.0055)(.9962)+(.0228)(.0872)$$

$$= .00747$$

$$C_{N_B} = 2\left[C_{N_{B.25L_b}} \frac{S_b\ L_b}{S_w\ B_w} + C_{Y_B} \frac{S_b}{S_w}\left(\frac{x_b}{B_w} - \frac{.25\ L_b}{B_w}\right)\right]$$

$$= 2\left[(.0096)\frac{363.2 \times 68.2}{1447.24 \times 109.27} + (.0074)\left(\frac{363.2}{1447.24}\right)\left(\frac{x_b}{109.27} - \frac{.25 \times 68.2}{109.27}\right)\right]$$

$$= .0030 + .0038\left(\frac{x_b}{109.27} - .157\right)$$

1	2	3	4	5	6
C. G. Location	x_b	$\frac{x_b}{109.27 \times 12}$	(3) - .157	(4) x .0037	(5) + .0030 = C_{N_B}
1 & 3	166.1	.127	-.030	-.000114	+.00289
2 & 4	179.6	.137	-.020	-.000076	+.00293
5	173.6	.132	-.025	-.000095	+.00291
7	182.9	.139	-.0175	-.0000647	+.00293

PART III-B-5

5. Horizontal Tail

It is assumed that the yawing moment of the horizontal tail due to yaw is negligible.

6. Landing Gear

The yawing moments of the landing gear down are not required by flight criteria requirements.

PART III-B-7

7. Propellers

For single engine operation, zero yaw, pack off, the yawing moments of the propellers will be identical with those for pack on. Complete calculations may be found in reference (5) pages 395 and 396. A curve of propeller yawing moment versus airspeed follows in figure 61

5° Yaw $\quad V_P \; (V\sigma^{-\frac{1}{2}} = 331.3$ mph)

$$C_{N_P} = -2 \frac{l_P}{B_W} \frac{F_P}{S_W} C_{Y_P}$$

For two engine operation the only difference between the yawing moments of the propellers pack on and pack off will be that due to change in arm (l_p) length with difference longitudinal C. G. locations.

$$C_{Y_P} = .0275$$

$$C_{N_P} = -2 \times \frac{178.2}{109.27 \times 12 \times 1447.25} \times .0275 \; l_P$$

$$= -.00000516 \; l_P$$

C. G. Position	l_P	C_{N_P}
1 & 3	-174.5	.000903
2 & 4	-187.9	.000965
5	-181.9	.000934
7	-191.3	.000987

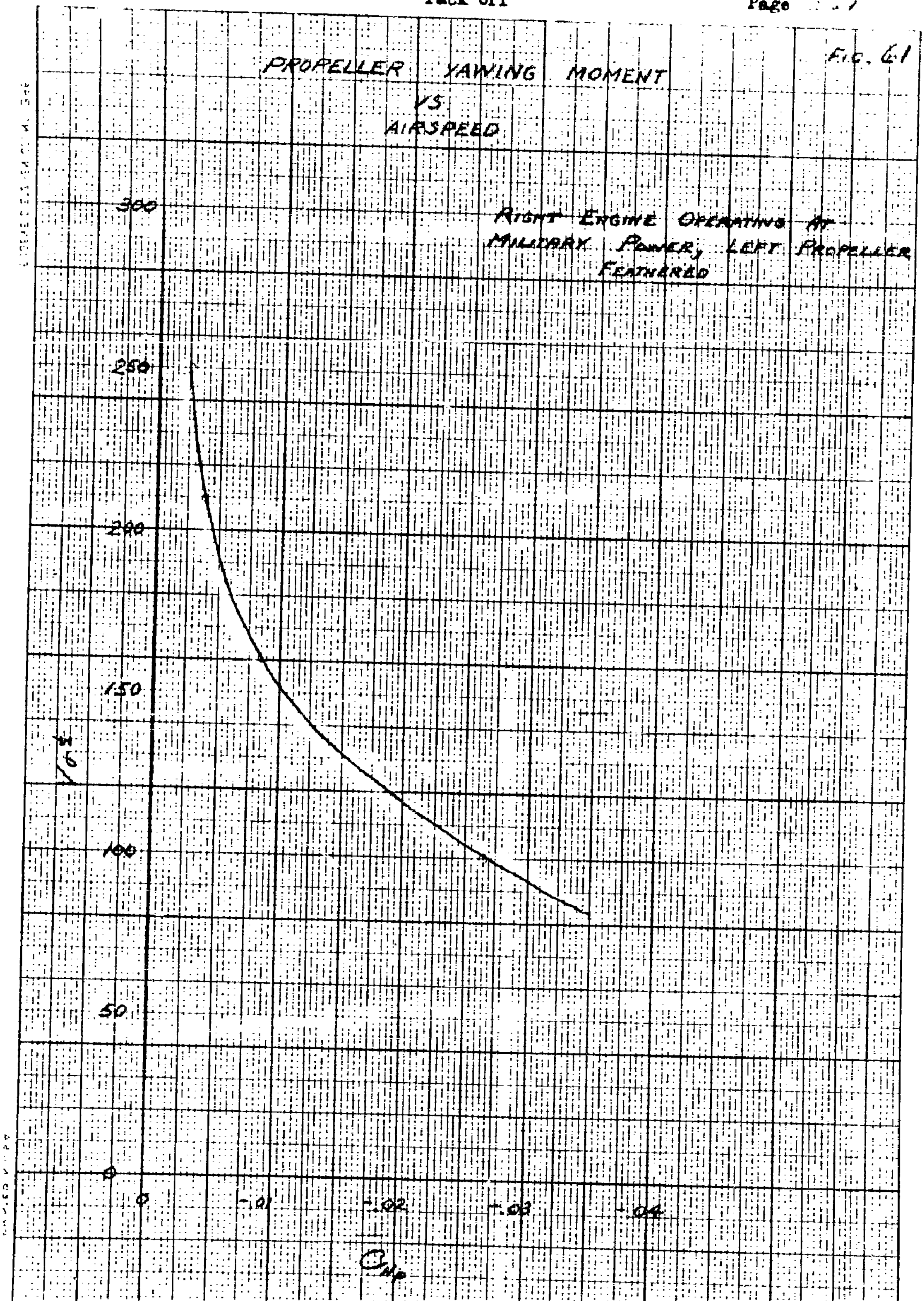


FIG. 61
PROPELLER YAWING MOMENT
VS
AIRSPEED
RIGHT ENGINE OPERATING AT MILITARY POWER, LEFT PROPELLER FEATHERED
300
250
200
150
100
50
0
0
-.01
-.02
-.03
-.04
C_{N_p}

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PART III-B-8

8. Slipstream Effects

The yawing moments due to slipstream effects are negligible for two engine operation, but for single engine operation may be appreciable. The moments will vary with speed, airplane angle of attack and thrust.

Single Engine Operation

$$C_{N_S} = \triangle X_S \frac{Yb}{q\, S_w\, B_w}$$

a. Wing Unflapped

For the single engine condition, unflapped wing, zero yaw, yawing moments due to slipstream effects are determined at various angles of attack for a range of speed as required by design criteria.

For zero yaw $\triangle X_S = \triangle D_S$

$$\triangle C_{D_{WI}} = \frac{\triangle D_S}{q\, S_w}$$

So that for all C. G. locations

$$C_{N_S} = \triangle C_{D_{WI}} \frac{Yb}{B_w} \qquad \frac{Yb}{B_w} = \frac{14.55}{109.27} = .1332$$

at $\psi = 0°$, the yawing moments will be identical with those for pack on; calculations of yawing moments due to slipstream with varying angle of attack and over an enveloping speed range may be found in Volume I of this report. Curves of yawing moment versus α for different speeds follow in figure 62 .

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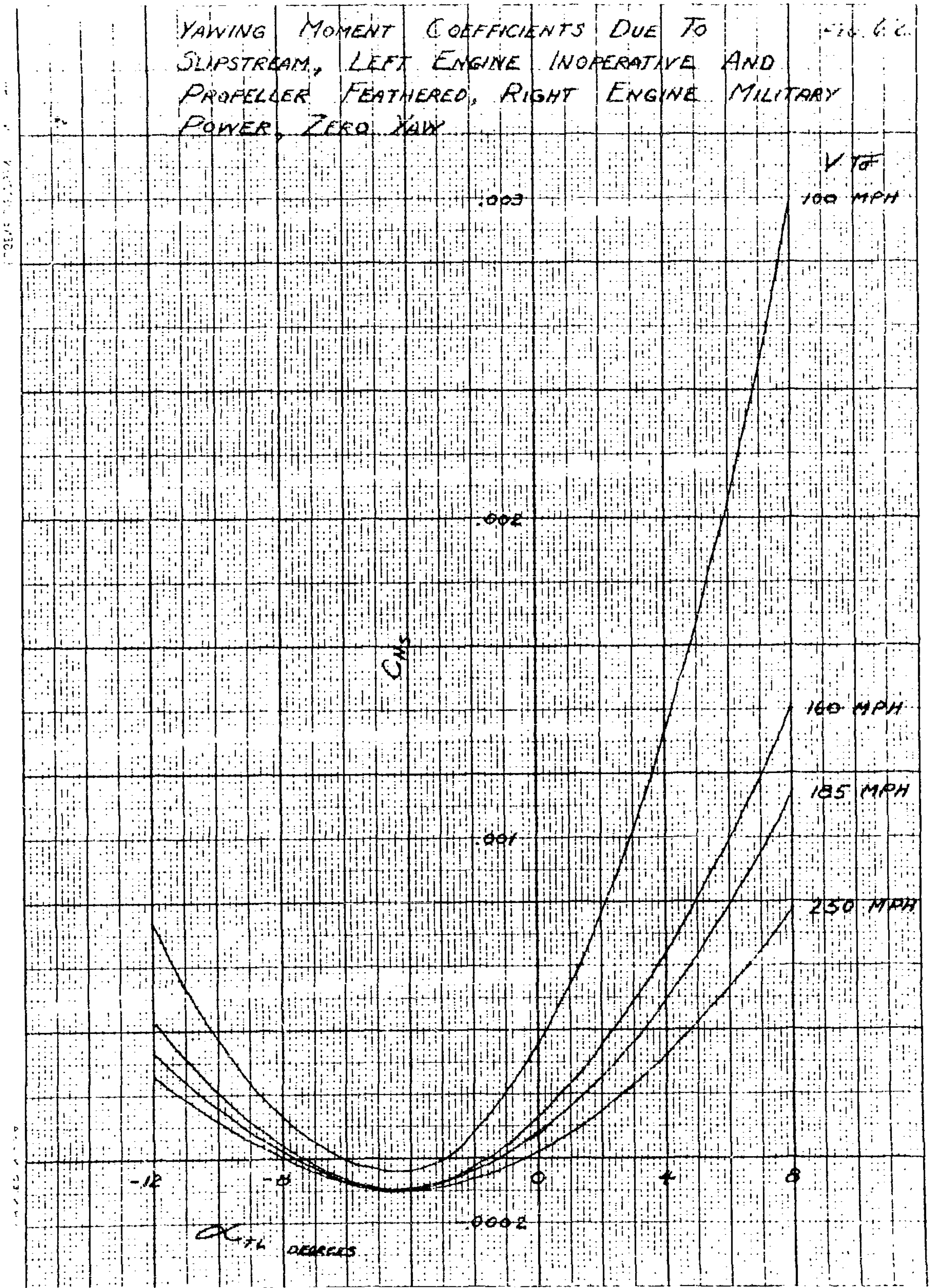
YAWING MOMENT COEFFICIENTS DUE TO SLIPSTREAM, LEFT ENGINE INOPERATIVE AND PROPELLER FEATHERED, RIGHT ENGINE MILITARY POWER, ZERO YAW
V TO
100 MPH
160 MPH
185 MPH
250 MPH
.003
.002
.001
-.0002
C_{NS}
-12
-8
0
4
8
α_{TL} DEGREES

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PART III-B-8a

8a. <u>Two Engine Operation</u>

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 Tails

a. Single Engine Operation

For single engine operation the dynamic pressure at the vertical tail in the slopstream of the operating engine will be different 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 deflections.

$$C_{N_{VT}} = \frac{S_{vt}}{S_w}\left\{\left[C_{X_{VT}}\ \frac{q_{vt}}{q_L}\ \frac{y_b}{B_w} + C_{X_{VT}}\ \frac{q_{vt}}{q_R}\ \frac{y_b}{B_w}\right] + \left[C_{Y_{VT}}\left(\frac{q_{vt}}{q}\right)_L \frac{l_{vt}}{B_w} + C_{Y_{VT}}\left(\frac{q_{vt}}{q}\right)_R \frac{l_{vt}}{B_w}\right]\right.$$

where (L) and (R) denote left and right vertical surfaces.

Calculations of $C_{N_{VT}}$ for a range of speeds from $V\sqrt{\sigma}$ = 100 to 250 mph, and rudder deflections $\delta_r = 0^\circ$ to 15° follow. Since the minimum gross weight at a particular speed requires maximum rudder deflection to maintain zero yaw the q ratios will be chosen at G. W. = 35,904 lbs. The foregoing is for the single engine condition, viz.: $\psi = 0^\circ$, right engine operating, and left propeller feathered.

$$\frac{S_{vt}}{S_w} = \frac{99.6}{1447.24} = .0687$$

l_{vt} = 621.1 for C. G. 1 & 3
607.6 for C. G. 2 & 4
613.8 for C. G. 5
604.2 for C. G. 7

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PART III-B-9a

YAWING MOMENT OF VERTICAL TAIL - SINGLE ENGINE OPERATION

$\delta_R = -15°$

1	2	3	4	5	6	7	8	9	10	11
$V\sqrt{\sigma}$	$\frac{S_{vt}}{S_w}$	$C_{X_{VT}}$	$\frac{y_b}{B_w}$ +R -L	$\frac{q_{vt}}{q}$ L	$\frac{q_{vt}}{q}$ R	3×4×5 L	3×4×6 R	7+8	$C_{Y_{VT}}$	$\frac{l_{vt}}{B_w}$
100	.0687	.0456	.133	1.00	1.42	-.00608	+.00861	.00253	.3000	.475
160					1.1		.00668	.00060		
185					1.058		.00640	.00032		
250					1.022		.00623	.00015		

12	13	14	15	16						
C.G. 1 & 3					C.G. 2 & 4		C.G. 5		C.G. 7	
10×11×5	10×11×6	12+13	9+14	$C_{N_{VT}}$ 2×15	$\frac{l_{vt}}{B_w}$	$C_{N_{VT}}$	$\frac{l_{vt}}{B_w}$	$C_{N_{VT}}$	$\frac{l_{vt}}{B_w}$	$C_{N_{VT}}$
.1425	.2050	.3475	.3500	.0240	.463	.0234	.469	.0237	.4608	.02316
	.1570	.2995	.3001	.0206		.0201		.0203		.01998
	.151	.2935	.2935	.0201		.0196		.0198		.01950
	.146	.2885	.2887	.0197		.0192		.0194		.01910

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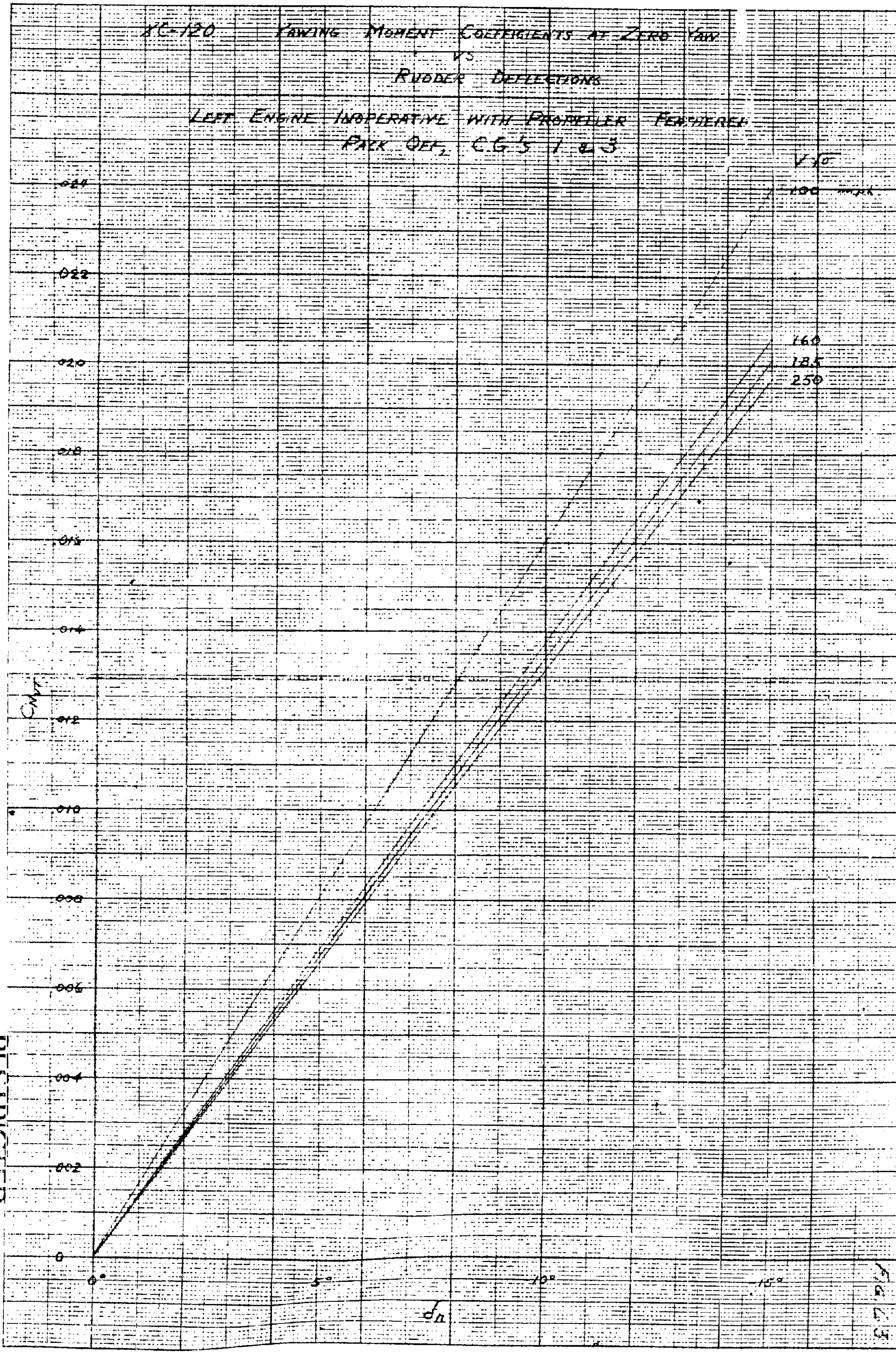
XC-120 YAWING MOMENT COEFFICIENTS AT ZERO YAW
VS
RUDDER DEFLECTIONS
LEFT ENGINE INOPERATIVE WITH PROPELLER FEATHERED
PACK OFF, CG'S 1 & 3
$V\sqrt{\sigma}$
100 MPH
160
185
250
.024
.022
.020
.018
.016
.014
.012
.010
.008
.006
.004
.002
0
C_{N_T}
0°
5°
10°
15°
δ_R

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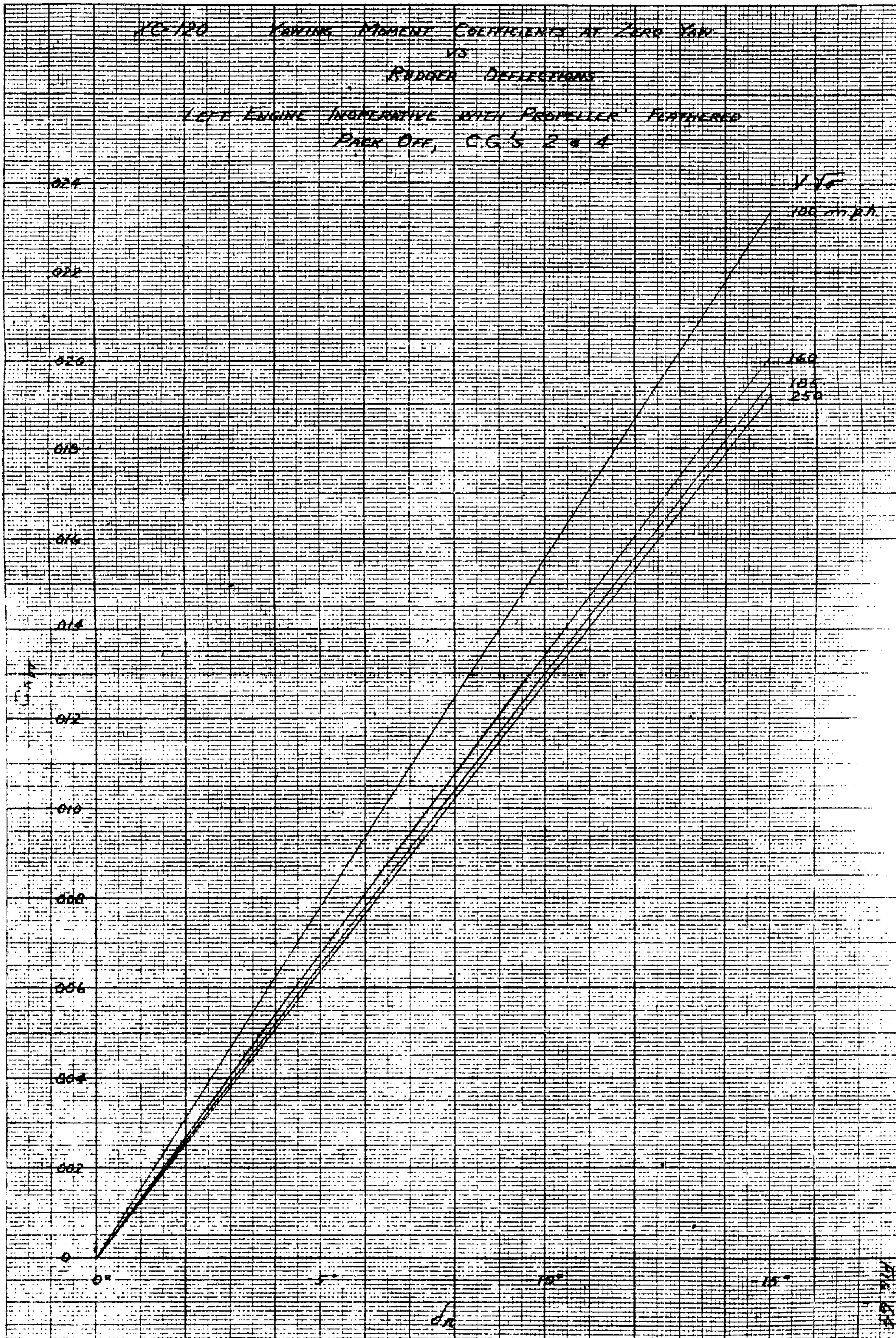
XC-120 YAWING MOMENT COEFFICIENTS AT ZERO YAW
VS
RUDDER DEFLECTIONS
LEFT ENGINE INOPERATIVE WITH PROPELLER FEATHERED
PACK OFF, C.G.'S 2 & 4
$V\sqrt{\sigma}$
100 m.p.h.
160
185
250
.024
.022
.020
.018
.016
.014
.012
.010
.008
.006
.004
.002
0
C_n
0°
5°
10°
15°
δ_R
FIG. 64

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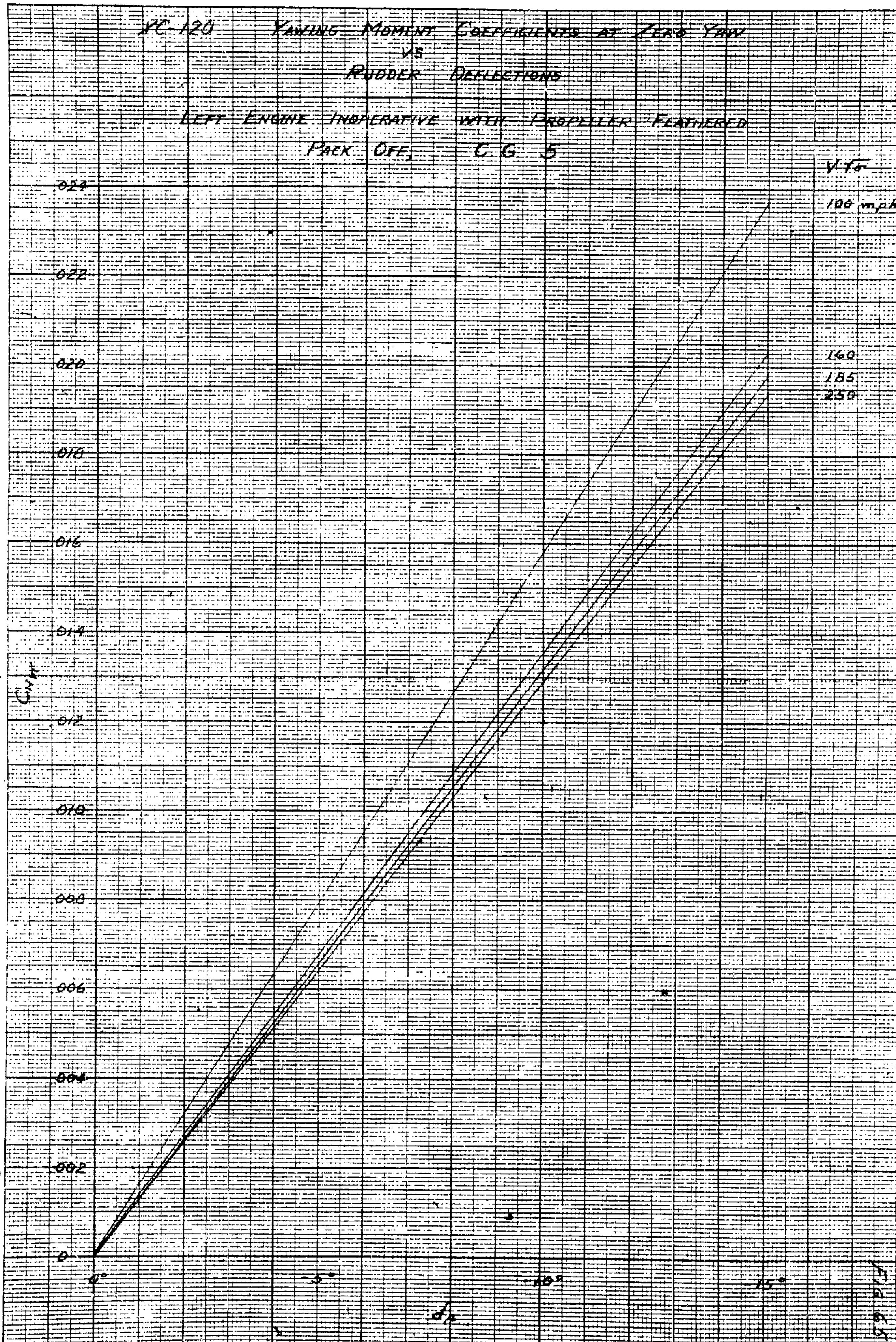

Fig 65

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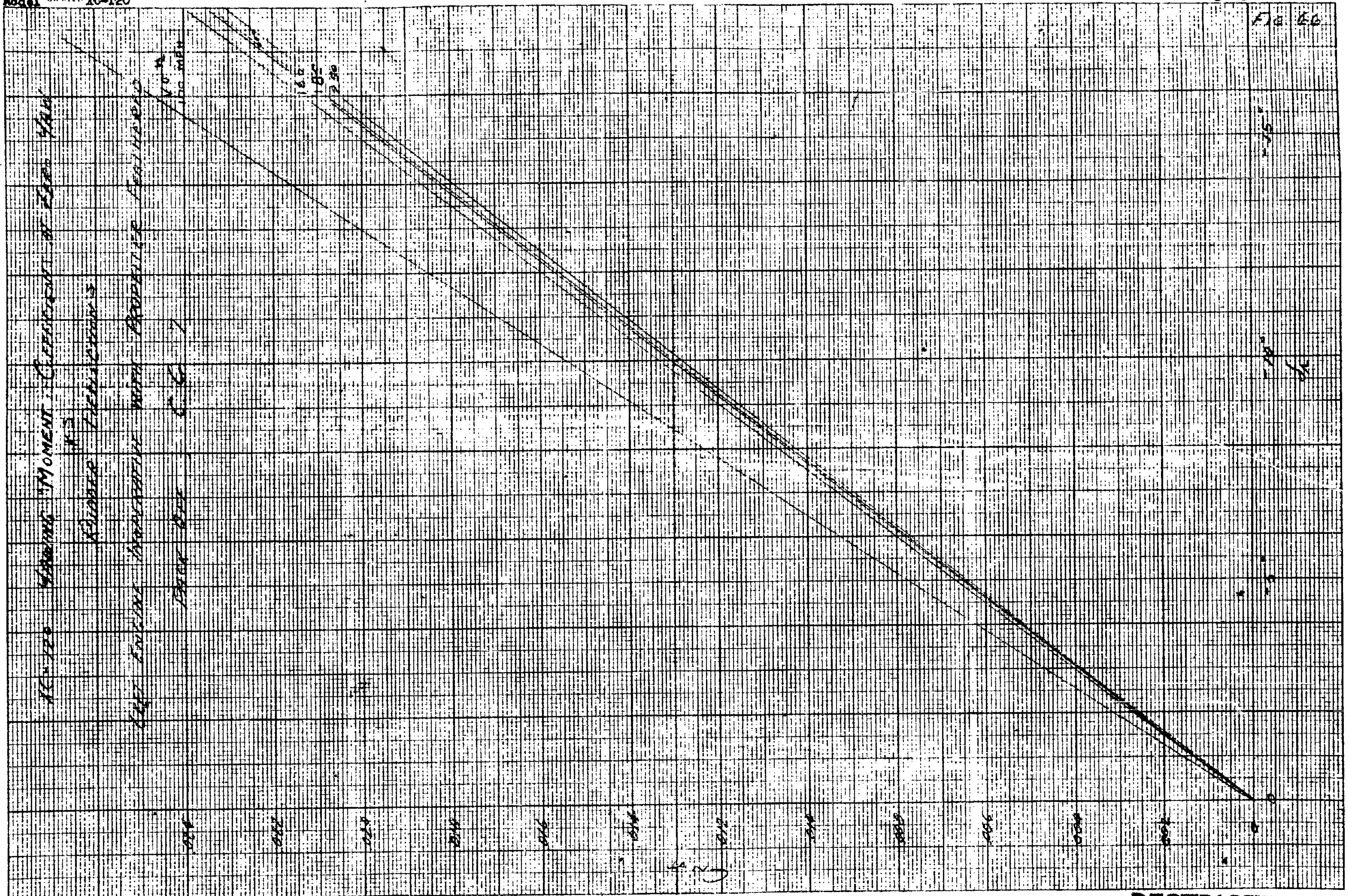

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PART III-B-9b

9b. Vertical Tails

Two Engine Operation

$$C_{N_{VT}} = 2 \; C_{Y_{VT}} \; \frac{q_{vt}}{q} \; \frac{S_{vt}}{S_w} \left(- \frac{l_{vt}}{b_w}\right)$$

for $\psi = 5°$ and V_D $V\sigma^{\frac{1}{2}}$ = 331.3 mph

$$\alpha_{vt} = \psi - \sigma_{p_{ave}} - \psi \frac{d\sigma}{d\psi}$$

$\frac{d\sigma}{d\psi}$ = .107 page 241 reference 1

$\sigma_{p_{ave}}$ = .1376 page 336 reference 1

$$\alpha_{vt} = 5 - .1376 - 5(.107)$$
$$= 4.327°$$

$$C_{Y_{VT}} = \eta_{vt} \; C_{C_{VT}} = .75(.056\,\alpha_{vt} - .0269\,\delta_r)$$

δ_r	0°	-5°	-10°	-15°
$C_{Y_{VT}}$	.1820	.0814	-.0190	-.1200

$\frac{q_{vt}}{q}$ = 1.0055 page 336 reference 1

$$C_{N_{VT}} = 2\,C_{Y_{VT}}\,(1.0055)\left(\frac{99.6}{1447.24}\right)\left(-\frac{l_{vt}}{1310}\right)$$

$$= -.0001053 \; C_{Y_{VT}} \; l_{vt}$$

For C. G. locations 1 and 3

l_{vt} = 621.1 $C_{N_{VT}} = -.0652\,C_{Y_{VT}}$

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DATE 1 May 1949

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PART III-B-9b

Vertical Tails (Cont.)

For C. G. Locations 2 and 4

$$l_{vt} = 607.6 \qquad C_{N_{VT}} = -.0636\ C_{Y_{VT}}$$

For C. G. Location 5

$$l_{vt} = 613.8 \qquad C_{N_{VT}} = -.0646\ C_{Y_{VT}}$$

For C. G. Location 7

$$l_{vt} = 604.2 \qquad C_{N_{VT}} = -.06362\ C_{Y_{VT}}$$

δ_r	$C_{Y_{VT}}$	C. G. 1 & 3 $C_{N_{VT}}$	C. G. 2 & 4 $C_{N_{VT}}$	C. G. 5 $C_{N_{VT}}$	C. G. 7
0°	.1820	-.01190	-.01156	-.01173	-.011578
-5°	.0814	-.00532	-.00518	-.00525	-.005178
-10°	.0190	.00124	-.00121	-.00122	-.001208
-15°	.1200	.00782	-.00762	-.00775	-.00763

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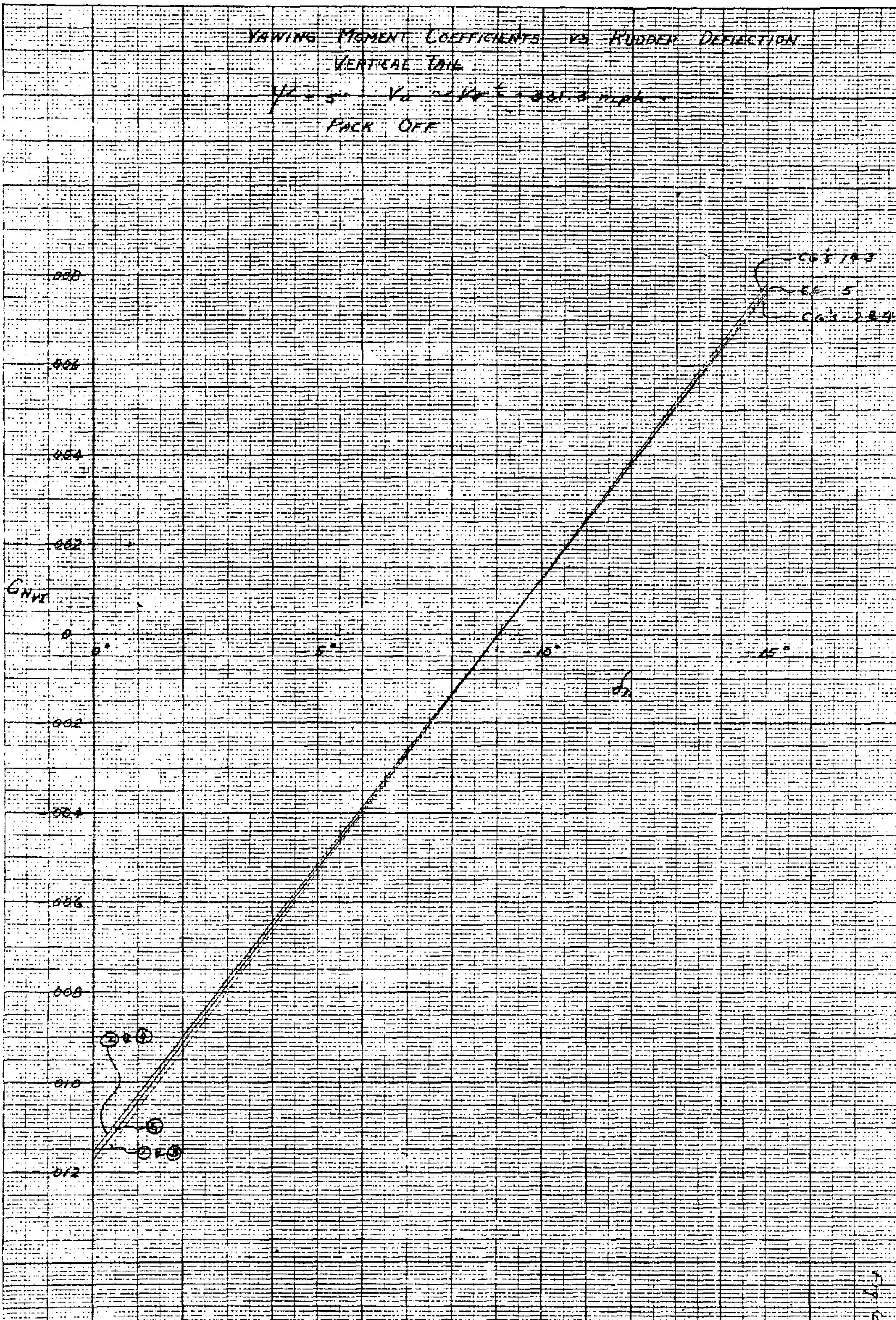

Fig 67

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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 obtain the yawing moment of the tailless airplane.

The characteristics of the tailless airplane are determined for the two conditions, viz. single engine operation, zero yaw and two engine operation $\psi = 5^\circ$, V_D $V\sigma^{\frac{1}{2}} = 331.3$ mph.

1. Zero Yaw -Single Engine Operation

$\Sigma\, C_{N_{A-T}}$ for single engine operation, $= 0^\circ$

$$C_{N_{A-T}} = C_{N_W} + C_{N_N} + C_{N_B} + C_{N_S} + C_{N_P}$$

Using minimum gross weight which is critical, the yawing moment of the tailless airplane is determined for a range of speeds.

$$C_L = \frac{391 \times 35904}{\sigma \times 1447.24V^2}$$ ($\sigma = 1.0$ at sea level)

α_{TL} is from figure 2 reference (1)

$V\sqrt{\sigma}$	100	160	185	250
C_L	.97	.378	.285	.155
α_{TL}	3.5°	-3.6°	-4.7°	-6.2°
C_{N_W}	0	0	0	0
C_{N_N}	0	0	0	0
C_{N_B}	.00048	.00048	.00048	.00048
C_{N_S}	.00117	-.00009	-.0001	-.00007
C_{N_P}	-.02652	-.00842	-.00568	-.00244
$C_{N_{A-T}}$	-.02487	-.00803	-.00530	-.00203

PART III-C-2

2. $\sum C_N$ for $\psi = 5^o$ and V_D = 331.3 mph

$$C_{N_{A-T}} = C_{N_W} + C_{N_N} + C_{N_B} + C_{N_P}$$

C. G. Position	C_{N_W}	C_{N_N}	C_{N_B}	C_{N_P}	$C_{N_{A-T}}$
1 & 3	.00005	.00044	.00289	.00090	.00428
2 & 4	.00006	.00049	.00292	.00097	.00444
5	.00006	.00047	.00291	.00093	.00437
7	.000064	.000501	.00293	.00099	.00450

From reference (5) pages 440 and 441 we shall use an overall interference factor of .65

Considering interference effects: F_I = .65

C. G. Position	$(C_{N_{A-T}} / F_I)$	$C_{N_{A-T}}$
1 & 3	.00428	.002782
2 & 4	.00444	.002886
5	.00437	.00284
7	.00450	.002925

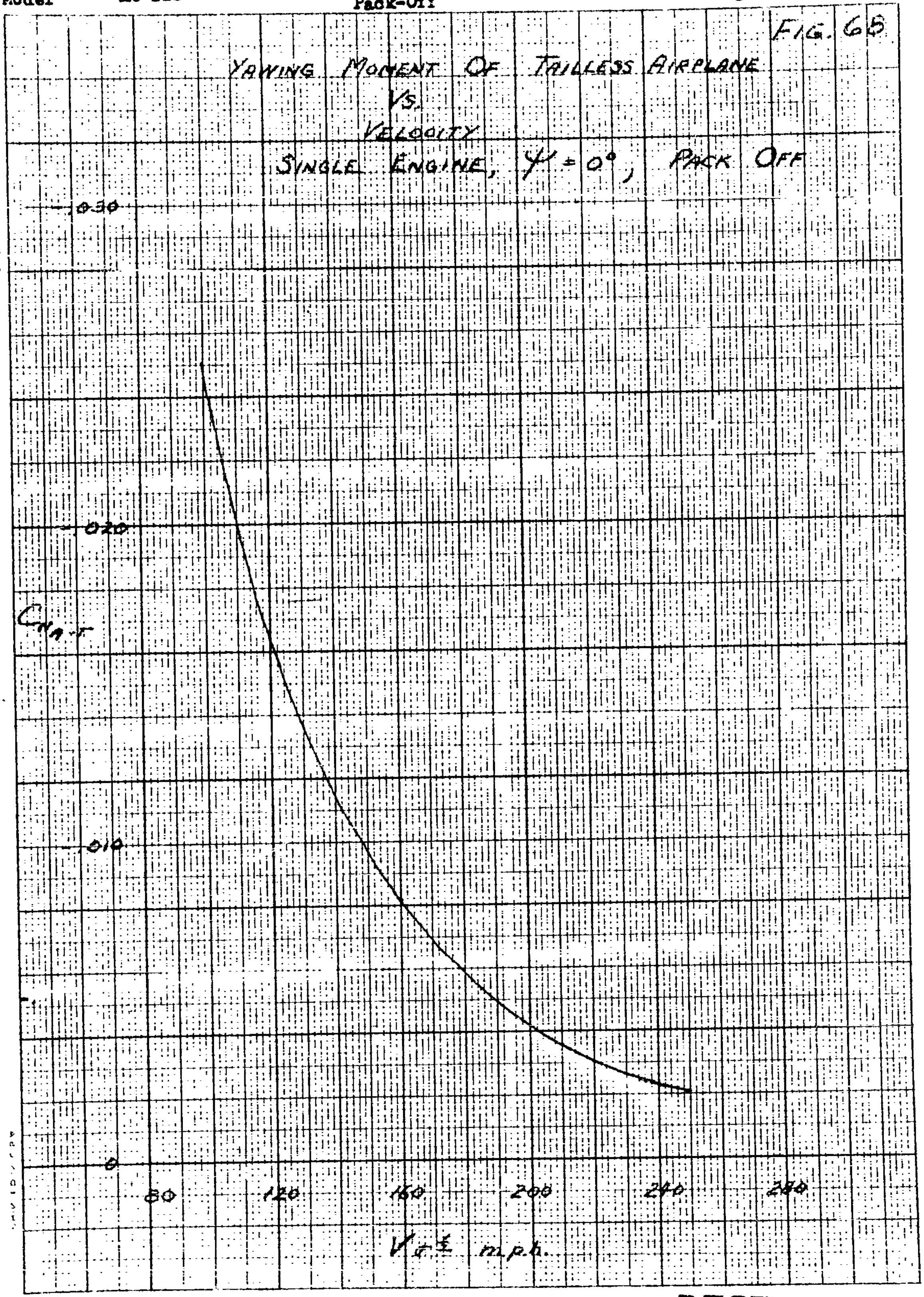
FIG. 65
YAWING MOMENT OF TAILLESS AIRPLANE
VS.
VELOCITY
SINGLE ENGINE, $\psi = 0°$, PACK OFF
.030
.020
.010
0
$C_{n_{A-T}}$
80
120
160
200
240
280
$V_{c.s.}$ m.p.h.

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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) Side gust of 50 fps at $V\sigma^{\frac{1}{2}} = 265$ 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 $V\sigma^{\frac{1}{2}} = 250$ mph.

(c) Zero yaw with one engine dead and the other engine delivering military power at V_{max} ($V\sigma^{\frac{1}{2}} = 200$ mph) for this condition, plus a side gust of 30 fps.

(d) Five degrees yaw with resulting rudder deflection at V_D ($V\sigma^{\frac{1}{2}} = 331.3$ mph).

(e) Five degrees yaw with zero rudder deflection at V_D ($V\sigma^{\frac{1}{2}} = 331.3$ mph).

A summary of all loads is presented on page 281. All loads given herein are for one tail only.

PART III-D (Cont.)

Vertical tail load for a side gust of 50 fps at $V\sigma^{\frac{1}{2}} = 265$ with initial $\alpha_{vt} = 0$.

$$q = 179.6 \qquad \frac{q_{vt}}{q} = 1.020 \text{ figure 65 reference (1)}$$

$$\Delta\alpha_{vt} = U/V = \frac{50}{265 \times 1.467} = .1286 = 7.3^{\circ}$$

$$\alpha_{vt} = 0 + 7.3^{\circ} = 7.3^{\circ}$$

$$\eta_{vt} C_{L_{VT}} = .75(.056\alpha_{vt} + .0269\,\delta_r) = .3066 \text{ for } \delta_r = 0$$

$$L_{VT} = \eta_{vt} C_{L_{VT}} \times q \times \frac{q_{vt}}{q} \times S_{vt}$$

$$= .3066 \times 179.6 \times 1.020 \times 99.6$$

$$= 5594 \text{ lbs. (each vertical tail)}$$

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Subject: BASIC FLIGHT CRITERIA - PACK OFF

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_σ mph	100			160			185			250		
q #/□'	25.6			65.5			87.5			160		
$C_{N_{R-T}}$	-.02487			-.00803			-.00530			-.00203		
$C_{N_{VT}}$	.02487			.00803			.00530			.00203		
δ_r °	-15.53	-15.95	-15.77	-5.83	-5.95	-5.92	-3.95	-4.03	-4.01	-1.53	-1.61	-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
$\eta_{vt} C_{C_{VT}}$ *	-.313	-.322	-.317	-.118	-.120	-.1195	-.0796	-.0813	-.0807	-.0309	-.0325	-.0319
$\eta_{vt} C_{C_{VT}}$	-.313	-.322	-.317	-.118	-.120	-.1195	-.0796	-.0813	-.0807	-.0309	-.0325	-.0319
q_{vt}/q *	1.42	1.42	1.42	1.098	1.098	1.098	1.057	1.057	1.057	1.022	1.022	1.022
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} *	-1130	-1163	-1144	-842	-857	-850	-730	-749	-740	-501	-529	-517
L_{VT}	-795	-818	-805	-765	-777	-773	-692	-709	-701	-490	-518	-503

* Indicates tail in slipstream of operating engine

VERTICAL TAIL LOADS FOR ZERO YAW WITH ONE ENGINE DEAD AND THE OTHER ENGINE DELIVERING TAKE-OFF (MILITARY) POWER FOR VARIOUS SPEEDS

CG	7	7	7	7
$V^{1/2}$	100	160	185	250
$q\frac{\rho}{\rho_0}$	25.6	65.5	87.5	160
$C_{N_{AT}}$	-.02487	-.00803	-.00530	-.00203
$C_{N_{VT}}$	.02487	.00803	.00530	.00203
δ_R	-16.1	-6.0	-4.05	-1.62
α_{VT} *	0	0	0	0
$\eta_{VT} C_{L_{VT}}$ *	-.3248	-.12105	-.08171	-.03268
$\eta_{VT} C_{L_{VT}}$	-.3248	-.12105	-.08171	-.03268
$\frac{q_{VT}}{q}$ *	1.42	1.098	1.057	1.022
$\frac{q_{VT}}{q}$	1.00	1.00	1.00	1.00
L_{VT} *	-1176	-867	-753	-532
L_{VT}	-828	-790	-712	-521

* INDICATES TAIL IN SLIPSTREAM OF OPERATING ENGINE

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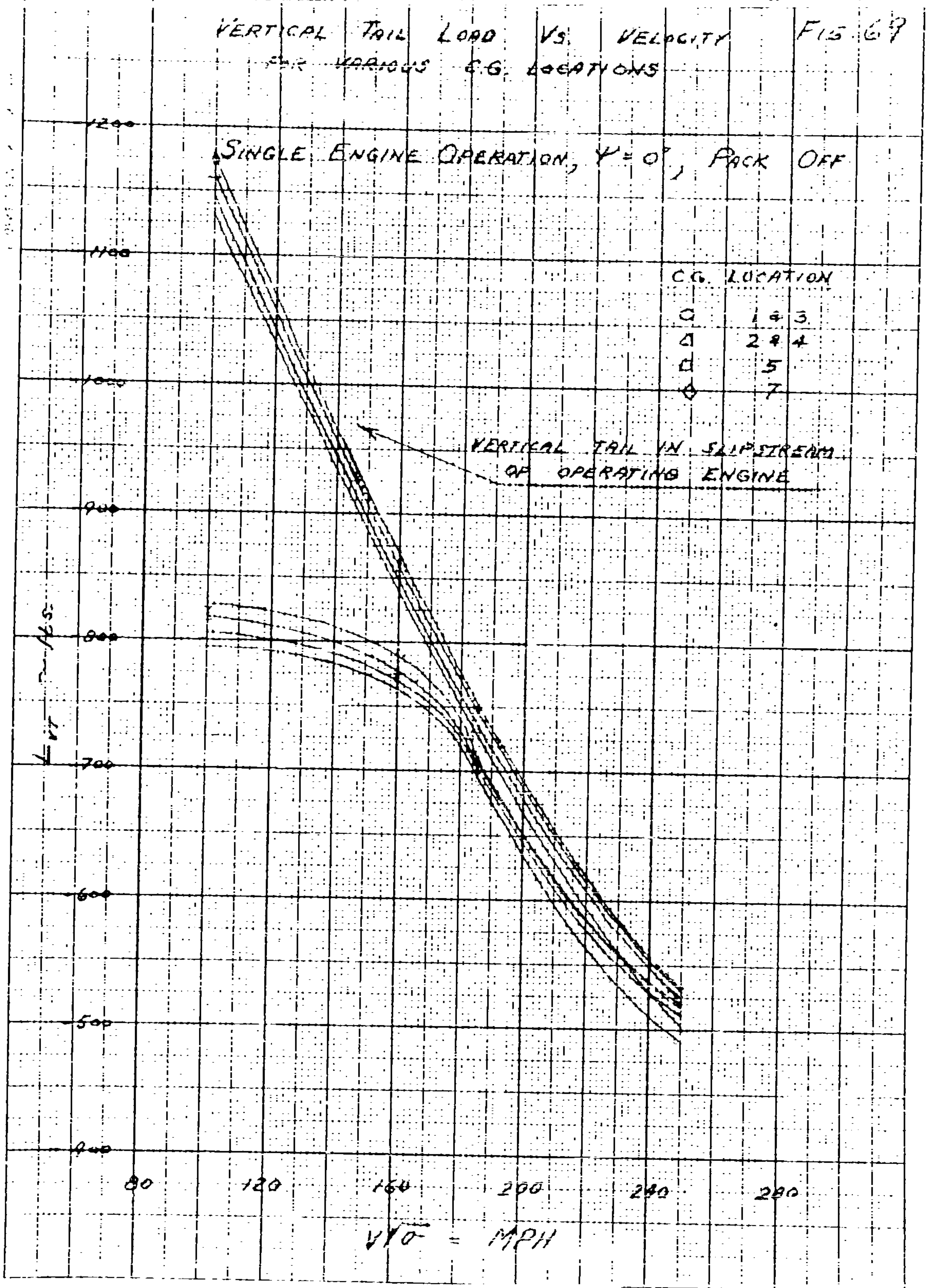
VERTICAL TAIL LOAD VS VELOCITY
VARIOUS C.G. LOCATIONS
FIG 69
SINGLE ENGINE OPERATION, Ψ = 0°, PACK OFF
C.G. LOCATION
1 & 3
2 & 4
5
7
VERTICAL TAIL IN SLIPSTREAM OF OPERATING ENGINE
1200
1100
1000
900
800
700
600
500
400
LVT ~ LBS.
80
120
160
200
240
280
V√σ = MPH

PART III-D

Vertical tail loads with one engine dead and the other engine delivering military power at V_C = 200 mph plus a side gust of 30 fps at zero yaw

$$\Delta\alpha_{vt} \text{ due to gust} = \frac{\pm 30}{200 \times 1.467} = \pm .1022 = 5.833°$$

C.G. Location	(1) & (3)		(2) & (4)		(5)		(7)	
$C_{N_{AT}}$	-.0043							
$C_{N_{VT}}$	.0043							
V_R	-3.22		-3.29		-3.26		-3.32	
q #/□'	103							
$\frac{q_{vt}}{q}$ *	1.042							
$\frac{q_{vt}}{q}$	1.00							
Initial α_{vt}° *	0°							
Initial α_{vt}°	0°							
gust	+	-	+	-	+	-	+	-
effective α_{vt}° *	5.833	-5.833	5.833	-5.833	5.833	-5.833	5.833	-5.833
effective α_{vt}°	5.833	-5.833	5.833	-5.833	5.833	-5.833	5.833	-5.833
$\eta_{vt} C_{L_{VT}}$ *	+.15[illegible]	-.3099	.1786	-.31136	.1792	-.3100	.1780	-.3120
$\eta_{vt} C_{L_{VT}}$	+.1800	-.3099	.1786	-.31136	.1792	-.3108	.1780	-.3120
L_{VT} * #	+1924	-3313	1909	-3328	1916	-.3322	1903	-3335
L_{VT} #	+1847	-3180	1832	-3194	1839	-.3188	1826	-3200

* Indicates vertical tail in slipstream of operating engine.

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PART XII-D (Cont.)

Vertical Tail Loads for 5° Yaw at $V\sigma^{\frac{1}{2}}$ = 331.3 mph

C. G. Location	1 & 3	2 & 4	5	7
$C_{N_{A-T}}$	.00278	.00289	.00284	.00293
$C_{N_{VT}}$	-.00276	-.00286	-.00281	-.00293
q #/sq. ft.	281	281	281	281
q_{vt}/q	1.016	1.016	1.016	1.016
$\delta_r{}^\circ$	-6.95	-6.8	-6.84	-6.76
$\alpha_{vt}{}^\circ$	4.327	4.327	4.327	4.327
$\eta_{vt}\, C_{C_{VT}}$	.0415	.04454	.04374	.04535
L_{VT} lbs.	1173	1267	1244	1290

Vertical Tail Loads for 5° Yaw At $V\sigma^{\frac{1}{2}}$ = 331.3 mph with Zero Rudder Deflection

C. G. Location	1 & 3	2 & 4	5	7
$C_{N_{A-T}}$	.00275	.002869	.00284	.00293
$C_{N_{VT}}$	-.0119	-.01156	-.01185	-.01158
C_{N_A}	-.00914	-.00870	-.00904	
q #/sq. ft.	281	281	281	281
q_{vt}/q	1.016	1.016	1.016	1.016
$\delta_r{}^\circ$	0	0	0	0
$\alpha_{vt}{}^\circ$	4.327	4.327	4.327	4.327
$\eta_{vt}\, C_{C_{VT}}$	.182	.182	.182	.182
L_{VT}	5150	5150	5150	5150

TABULATION OF VERTICAL TAIL LOADS ~ PACK OFF

CONDITION	ψ	C.G.	ENG. OPER.	$V_0^{-\frac{1}{2}}$	q #/FT²	$\frac{q_{vt}}{q}$ *	$\frac{q_{vt}}{q}$	δ_r	α_{vt} DEG. *	α_{vt} DEG.	L_{vt} LBS. *	L_{vt} LBS.
Side gust 50 f.p.s. at 265 m.p.h. with init. α_{vt} = 0	0	ALL	R&L	265	179.6	1.020	—	0	7.3	—	5594	—
Zero yaw; one engine dead, other delivering Military Power	0	1&3	R	100	25.6	1.42	1.00	-15.53	0	0	-130	-795
	0	2&4	R	100	25.6	1.42	1.00	-15.95	0	0	-163	-818
	0	5	R	100	25.6	1.42	1.00	-15.77	0	0	-1144	-805
	0	1&3	R	160	65.5	1.098	1.00	-5.83	0	0	-842	-765
	0	2&4	R	160	65.5	1.098	1.00	-5.95	0	0	-857	-777
	0	5	R	160	65.5	1.098	1.00	-5.92	0	0	-850	-773
	0	1&3	R	185	87.5	1.057	1.00	-3.95	0	0	720	-692
	0	2&4	R	185	87.5	1.057	1.00	-4.03	0	0	749	-709
	0	5	R	185	87.5	1.057	1.00	-4.01	0	0	740	-701
	0	1&3	R	250	160	1.022	1.00	-1.53	0	0	501	-490
	0	2&4	R	250	160	1.022	1.00	-1.61	0	0	-529	-518
	0	5	R	250	160	1.022	1.00	-1.58	0	0	-519	-508
One eng. dead; other deliv. Military Power — + gust 35 f.p.s.	0	1&3	R	200	103	1.042	1.00	-3.22	+5.833	+5.833	+1914	+1847
	0	2&4	R	200	103	1.042	1.00	-3.29	+5.833	+5.833	+1909	+1832
	0	5	R	200	103	1.042	1.00	-3.26	+5.833	+5.833	1916	+1835
— gust 36 f.p.s.	0	1&3	R	200	103	1.042	1.00	-3.22	-5.833	-5.833	-3315	-3180
	0	2&4	R	200	103	1.042	1.00	-3.29	-5.833	-5.833	-3328	-3194
	0	5	R	200	103	1.042	1.00	-3.26	-5.833	-5.833	-3312	-3188
5° yaw with resulting δ_r @ 331.3 m.p.h.	5	1&3	R&L	331.3	281	1.016	1.016	-6.95	4.327	4.327	173	
	5	2&4	R&L	331.3	281	1.016	1.016	-6.8	4.327	4.327	267	
	5	5	R&L	331.3	281	1.016	1.016	-6.84	4.327	4.327	1244	
5° yaw with δ_r = 0° @ 331.3 m.p.h.	5	ALL	R&L	331.3	281	1.016	1.016	0	4.327	4.327	1250	

* Indicates tail in slipstream of operating engine

REPORT NO. R107-014

OF FAIRCHILD ENGINE & AIRPLANE CORPORATION

MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY

DATE 1 May 1949

REVISED

Subject:— BASIC FLIGHT CRITERIA – PACK OFF

TABULATION OF VERTICAL LOADS ~ PACK OFF

CONDITION	ψ	C.G	ENG. OPER.	$V\sigma^{\frac{1}{2}}$	q #/FT.²	$\frac{q_{vt}}{q}$ *	$\frac{q_{vt}}{q}$	σ_n	$\alpha_{vt}°$ *	$\alpha_{vt}°$	L_{VT} LBS. *	L_{VT} LBS.
Zero yaw; one engine dead, other delivering Military Power	0	7	R	100	25.6	1.42	1.00	-16.1	0	0	-1176	-826
	0	7	R	160	65.5	1.098	1.00	-6.0	0	0	-867	-790
	0	7	R	185	87.5	1.057	1.00	-4.05	0	0	-753	-712
	0	7	R	250	160	1.022	1.00	-1.62	0	0	-532	-521
One eng. dead, other deliv. Military Power, +gust 30 fps.	0	7	R	200	103	1.042	1.00	-3.32	5.833	5.833	1903	1626
-gust 30 fps.	0	7	R	200	103	1.042	1.00	-3.32	-5.833	-5.833	-3335	-3200
5° yaw with resulting σ_n @ 331.3 mph.		7	R&L	331.3	281	1.016	1.016	-6.76	4.327	4.327	1240	

* Indicates tail in slipstream of operating engine.

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PART III-E

E. Rudder Tab Loads

Rudder tab loads were determined for maximum tab deflection 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 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 (331.3 mph).

The rudder trim tab hinge moment is determined as follows:

$$H_{r_{tt}} = \frac{d\,C_{H_{r_{tt}}}}{d\,\delta_{r_{tt}}}\ \delta_{r_{tt}} \times q \times \frac{q_{vt}}{q} \times S_{r_{tt}} \times c_{r_{tt}}$$

$S_{r_{tt}}$ = 3.01 sq. ft. $c_{r_{tt}}$ = .673 ft.

$\delta_{r_{tt}}$ maximum = 15° page 17 reference (1)

$\frac{d\,C_{H_{r_{tt}}}}{d\,\delta_{r_{tt}}}$ = -.0043 per degree page 246 reference (1)

q = 280 $\frac{q_{vt}}{q}$ = 1.016 figure 65 reference (1)

$$H_{r_{tt}} = (-.0043)(15)(280)(1.016)(3.01)(.673)$$

$$= -38.4 \text{ ft. lb.}$$

$$N_{r_{tt}} = \frac{-H_{r_{tt}}}{-(4/9 \times .673)} = 128 \text{ lbs.}$$

$\frac{N_{r_{tt}}}{S_{r_{tt}}} = \frac{128}{3.01}$ = 42.5 lbs. per sq. ft. of trim tab area

MODEL XC-120 PREPARED BY CHECKED BY APPROVED BY

Subject:— BASIC FLIGHT CRITERIA - PACK OFF

DATE 1 May 1949

REVISED

PART II-E-2

2. Rudder Spring Tab

The load on the rudder spring tab is determined for the same conditions of maximum deflection and V_D (331.3 mph).

The rudder spring tab hinge moment is determined as follows:

$$H_{r_{st}} = \frac{d\,C_{H_{r_{st}}}}{d\,\delta_{r_{st}}}\ \delta_{r_{st}} \times q \times q_{vt} \times S_{r_{st}} \times c_{r_{st}}$$

$S_{r_{st}} = 1.66$ sq. ft. $\qquad c_{r_{st}} = .673$

$\delta_{r_{st}}$ maximum $= 17.5°$ page 17 reference (1)

$\frac{d\,C_{H_{r_{st}}}}{d\,\delta_{r_{st}}} = -.0043$ per degree, page 246 reference (1)

$q = 280 \qquad \frac{q_{vt}}{q} = 1.016$ figure 65 reference (1)

$$H_{r_{st}} = (-.0043)(17.5)(280)(1.016)(1.66)(.673)$$

$$= -23.8 \text{ ft. lbs.}$$

$$N_{r_{st}} = \frac{-H_{r_{st}}}{-(4/9 \times .673)} = 79.5 \text{ lbs.}$$

$$\frac{N_{r_{st}}}{S_{r_{st}}} = \frac{79.5}{1.66} = 47.8$$ lbs. per sq. ft. of spring tab area

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PART IV -

ASYMMETRICAL FLIGHT - ROLL

The rate of roll is calculated as a function of aileron deflections and of the dynamic pressure due to free stream velocity.

A. Rolling Moments

The rolling moments are calculated for the wing alone and are due primarily to the opposed deflections of the ailerons. The wing and ailerons on the XC-120 airplane are identical to those of the C-119B airplane. The constants and data required for rate of roll equation are developed in Part IV-B of reference (5) and the final equation is as follows:

$$\frac{p\,B_w}{2\,V}\,(133.435 + 66.497\,q + .002179\,q^2 \text{---}) = (345.708 - .349342\,q - 2.5704 \times 10^{-5}\,q^2 \text{---})\,\delta_L - (363.465 - .459607\,q - 2.8259 \times 10^{-5}\,q^2 \text{---})\,\delta_R$$

Thus given either the wing tip helix angle or one aileron deflection the other function may be determined in terms of q . The relationship of up aileron to down aileron is shown in figure 97 of reference (5).

PART IV-B

B. Design Criteria Requirements for Roll

Reference (2) specifies a wing tip helix angle of .08 degrees per second at $.8V_H$ and .015 degrees per second at V_D.

For the XC-120 airplane, pack off, V_H is 265 mph and V_D is 331.3 mph.

For $.8V_H$ equal to 212 mph $q = 114.9$ lbs./sq. ft. Substituting full aileron deflections ($\delta_{a_L} = 12^\circ$ and $\delta_{a_R} = -24^\circ$) in the above equation we obtain

$$\frac{p\,B_w}{2\,V} = .08 \quad \text{for 212 mph}$$

For V_D equal to 331.3 mph the following compressibility correction for dynamic pressure was used.

$$V_i = V_c\,(1 - M)^{-\frac{1}{4}}$$

where $M = \dfrac{V_c}{c}$

M = Mach number

V_c = velocity in a compressible fluid = 331.3 mph

C = velocity of sound

V_i = equivalent velocity in an incompressible fluid

Then $M = .4354$ and $V_i = 349.2$ mph

and the dynamic pressure for 331.3 mph corrected for compressibility is 311.9 lbs./sq. ft.

For $\dfrac{pB_w}{2\,V} = .015$ at 331.3 mph

$$\delta_{a_R} = -5.49^\circ \quad \text{and} \quad \delta_{a_L} = 4.78^\circ$$

PART IV-C

C. Aileron Loads

The aileron loads on the XC-120 airplane at .8 V_H are estimated from the data shown in Part IV-D of reference (5) which shows aileron loads on the up and down ailerons for various load factors and lift coefficients at 200 mph.

The maximum loads occur on the up aileron (δa_R = -24°) and are plotted on figure 70 versus lift coefficient. The load for 212 mph were obtained by ratioing the loads to the dynamic pressures, and are shown on the same figure.

The aileron loads for .8 V_H are as follows:

Gross Weight 54,000 lbs. $V\sqrt{\sigma}$ = 212 mph δa_R = -24°			
n	+1.0	+2.0	-1.0
C_L	.3246	.6492	-.3246
Load, lbs.	-1446	-1408	-1515

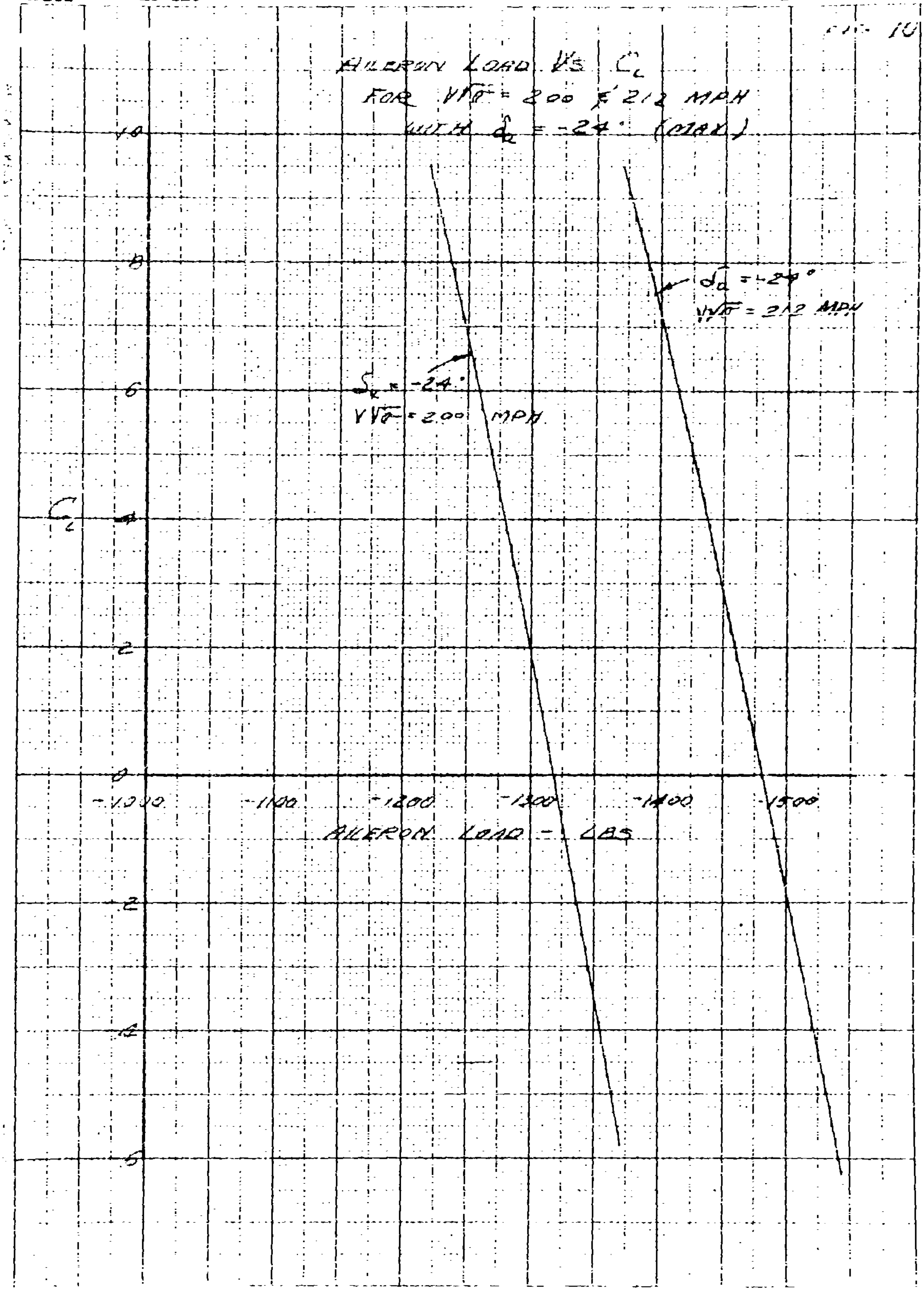


FIG. 10
AILERON LOAD VS C_L
FOR $V\sqrt{\sigma}$ = 200 & 212 MPH
WITH δ_a = -24° (MAX)
δ_a = -24°
$V\sqrt{\sigma}$ = 212 MPH
δ_a = -24°
$V\sqrt{\sigma}$ = 200 MPH
C_L
1.0
.8
.6
.4
.2
0
-.2
-.4
-.6
-1000
-1100
-1200
-1300
-1400
-1500
AILERON LOAD - LBS

PART IV-C

The aileron loads for V_D are also based on data shown in Part IV-D of reference (5) which shows the loads on the up and down aileron at V_D for the C-119B airplane (313 mph).

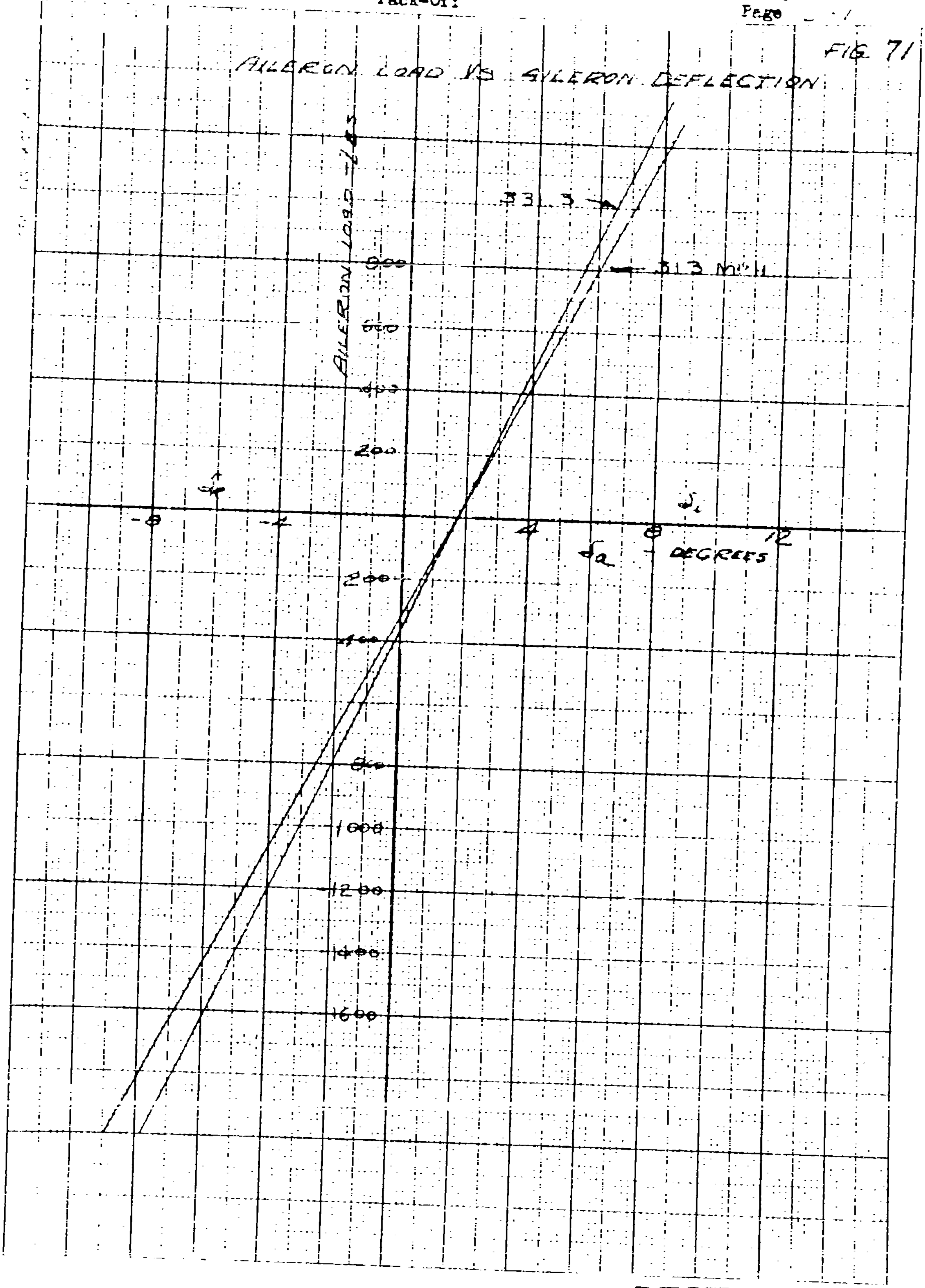
These loads are shown plotted versus aileron deflection on figure 71 for speed of 313 mph and 331.3 mph; loads at 331.3 mph being obtained by ratioing the loads to the dynamic pressures.

The aileron loads for V_D are as follows:

Gross Weight 54,000 lbs.		$V\sqrt{\sigma}$ = 331.3 mph
δ_a	4.78	-5.49
n	2.0	2.0
Load, lbs.	635	-1510

The maximum aileron load is -1515 lbs. and occurs at $V\sqrt{\sigma}$ = 212 mph
$n = -1.0$ $\delta_{a_R} = -24°$ ($\delta_{a_L} = 12°$)

FIG. 71



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PART IV-D

D. Aileron Tab Loads

1. Aileron Trim Tab

The aileron trim tab load is determined for maximum deflection at V_D (331.3 mph) for tab design load.

The aileron trim tab hinge moment is determined as follows:

$$H_{att} = \frac{d\,CH_{att}}{d\,\delta_{att}}\,\delta_{att} \times q \times S_{att} \times c_{att}$$

$$\frac{d\,CH_{att}}{d\,\delta_{att}} = -.0022 \quad \text{figure 13, reference (1)}$$

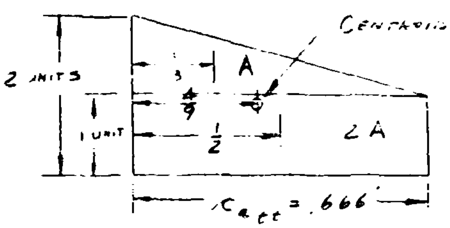
area $= S_{att} = 2.72$ sq. ft.

span $= b_{att} = 49$ inches

chord aft of hinge line $c_{att} = .666$ ft.

maximum deflections $= \pm 15^\circ$

Using the chordwise tab loading as specified in reference (2) the tab loads are determined.



H_t = tab hinge moment
N_t = normal force on tab, lbs.
y = force arm
A = area

From Above Data:

$$-y\,N_t = H_t$$

$$N_t = \frac{H_t}{-y} = \frac{H_t}{-4/9 \times .666 \text{ ft.}}$$

$$N_t = -3.38\,H_t$$

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PART IV-D (Cont.)

Calculating the hinge moment

$$H_{att} = (-.0022)(15)\ 280.7 \times 2.72 \times .666$$

$$= -16.80 \text{ ft. lbs.}$$

$$H_{att} = -3.38\ H_t$$

$$= -3.38 \times -16.78$$

$$= +56.72 \text{ lbs.}$$

So

$$\text{lbs./sq. ft.} = \frac{+56.72}{2.72} = +20.9 \text{ lbs./sq. ft. of tab area}$$

Since the maximum loading is less than the minimum specified design load of reference (2), design load will be the minimum specification value of 50 lbs. per sq. ft.

PART IV-D-2

2. Aileron Balance Tabs

Aileron balance tabs are checked for the two design conditions investigated in Part IV-B as follows:

a. $\frac{p\ B_w}{2\ V} = .08$ at $V\sqrt{\sigma} = 212$ mph

b. $\frac{p\ B_w}{2\ V} = .015$ at $V\sqrt{\sigma} = 331.3$ mph

The following physical characteristics of the aileron balance tabs are from reference (1).

$S_{abt} = 2.74$ sq. ft.

$c_{abt} = .666$ ft.

Maximum deflections $= +36^{o}, \ -16^{o}$

Using the same chordwise loading as in Part IV-D-1

$$-y\ N_t = H_t$$
$$N_t = -3.38\ H_t$$

The hinge moments for various deflections are calculated using the following equation:

$$H_{abt} = \frac{d\ CH_{abt}}{d\ \delta_{abt}} \quad \delta_{abt} \times q \times S_{abt} \times c_{abt}$$

$V\sqrt{\sigma}$ mph	q #/ft.^{2}	Aileron	δ_a Part IV-B	δ_{abt} Fig. 14 Ref. (1)	$\frac{d\ H_{abt}}{d\ \delta_{abt}}$	H_{abt} ft. lbs.	N_{abt} lbs.	N_{abt} #/ft.^{2}
212	115.0	left	+12	-22.25	-.0017	7.94	-26.83	-9.79
212	115.0	right	-24	+32.5	-.0017	-11.58	39.19	14.30
331.3	280.7	left	4.78	- 7.76	-.0017	6.73	22.[illegible]	-8.33
331.3	280.7	right	-5.49	8.3	-.0017	- 7.23	24.43	8.91

Since the maximum loading is less than the minimum specified design load of reference (2), design will be to the minimum specification value of 50 lbs. per sq. ft.

REEL-C 4546 B

A.T.I 99725

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(UNPUBLISHED CARD)

ATI 99 725 **AD-B803 065 U**

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

EO 1CEC1 dd 5 NOV 1953

BASIC FLIGHT CRITERIA - PACK OFF - MODEL XC-120

A. ATHANASIAN; G. POOLE; A. GRENIS AND OTHERS
1 MAY'49 298PP GRAPHS, DRWGS

STRUCTURES (7) LOADS, ~~CRITICAL~~
LOADS AND CRITERIA (1) ~~AIRPLANES~~ - ~~AERODYNAMICS~~ LOADING
C-120 Aircraft
RESTRICTED

P1/3, 5 Transport Aircraft

DEFENSE TECHNICAL INFORMATION CENTER
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3. DATE OF REQUEST: 14 Mar 00

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8. METHOD OF PAYMENT (X ONE): [ ] VISA [ ] MC [ ] AMX [X] Charge to NTIS Deposit Account No. [redacted] [ ] Bill My Organization to the Attention of:

9. CONTRACT MONITOR AND TELEPHONE NUMBER (Contractors Only)

10. NAME, TITLE, PHONE NUMBER OF REQUESTING OFFICIAL: [redacted])

SECTION II. BIBLIOGRAPHIC INFORMATION

11. AD NUMBER (If Known): AD-B803065

12. (TITLE, REPORT NUMBER, AUTHOR(S)), ETC: Basic Flight Criteria - Pack Off - Model XC-120, 298 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.

SECTION IV. RELEASING AGENCY

1. RELEASING AGENCY ADDRESS (If Known):

WRIGHT AIR DEVELOPMENT CTR
WRIGHT-PATTERSON AFB OH

2. RELEASING AGENCY DECISION (If the report was developed under the SBIR Program refer to instruction B 2 on the reverse of this form)

- [ ] APPROVED FOR RELEASE TO THE ABOVE REQUESTER
- [ ] DISAPPROVED. REASON FOR DISAPPROVAL
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28 APR 2000

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3. NAME AND TITLE OF RELEASING OFFICIAL: [illegible handwritten name]

4. TELEPHONE NO: 937-656-9431

5. SIGNATURE: [illegible signature]

6. DATE: 14 Apr 00

DTIC Form 55

PREVIOUS EDITIONS MAY BE USED UNTIL STOCK IS EXHAUSTED