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ANALYZED

PERIODIC PERFORMANCE REPORT

NO. 9

Dec. 1956

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Report no.: QC AVRO CF105 PPR-9

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by (Name): J.M.D. Henrie

(Dept.): DND Coordinate, Access to Information

Date: Aug 4 1992

Renee Auger
Signature

FORM 1316A

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

TECHNICAL DEPARTMENT (Aircraft)

ANALYZED

AIRCRAFT: CF-105 | REPORT NO: 9

SECRET
UNCLASSIFIED

FILE NO: | NO. OF SHEETS:

TITLE:

CF-105 PERIODIC PERFORMANCE REPORT

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Date of Issue ..19th DEC 56

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FORM 1316A

PERFORMANCE
DRAG
PROPULSION

SECRET UNCLASSIFIED

November 1956.

CF-105 PERIODIC PERFORMANCE REPORT - 9

Introduction

This is the ninth of a series of periodic performance reports for internal usage, to be issued from the Aerodynamics Department.

The pertinent changes are noted in the appropriate sections. For more detailed discussion of the drag changes see "Effect of N.A.C.A. Wind Tunnel and Free Flight Tests on the Estimated Performance of the CF-105".

As in the past, successive reports will present the latest data, with the alterations from the previous report noted. The report is divided into three major sections:

1) CF-105 Performance

2) CF-105 Drag

3) Propulsion

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PERFORMANCE

DRAG

PROPULSION

1: CF-105 PERFORMANCE WITH ORENDA IROQUOIS ENGINES

(C.G. at 29.5% M.A.C.)

The following CF-105 performance estimate is based on supersonic wind tunnel and free flight tests which were conducted at the Langley Laboratories of the N.A.C.A. The transonic and high subsonic regions are based on C.A.L. wind tunnel tests, whilst the low speed data is from N.A.E. tunnel tests.

The drag has been completely revised resulting in minor changes subsonically, but with larger changes in the supersonic region compared to Monthly Report Number 6, see Section 2.

The Orenda Iroquois engine data has also been completely revised, see Section 3.

The pertinent performance changes are listed below,

Combat 'g' at 1.5 M.N. at 50,000 feet ----------- -.25

Combat ceiling at 1.5 M.N. ---------------------- -2900 Ft.

Combat mission fuel (200 N.M. radius) ------------ +1119 Lb.

LOADING AND PERFORMANCE - 9 December, 1956.

Performance Under I.C.A.O. Standard Atmospheric Conditions

To R.C.A.F. Specification AIR 7-4

With Two Iroquois Engines

WEIGHT:

Take-Off Weight with 15,672 Lbs. Fuel (78.9% Max)	Lb.	59,336
Operation Weight Empty	Lb.	43,664
Combat Weight	Lb.	51,500
Normal design landing gross weight AIR 7-4 - MIL-S-5701	Lb.	45,854
Wing Loading at Normal Take-Off Weight	Lb/Sq.Ft.	48.4
Power Loading at Normal Take-Off Weight	Lb/Lb. Thrust	1.34

SPEED:

True Airspeed in Level Flight
at Sea Level at Combat Weight

Maximum Thrust A/B Lit	Kts.	700 *
Maximum Thrust A/B not Lit	Kts.	671

True Airspeed in Level Flight
at 50,000 Ft. at Combat Weight

Maximum Thrust A/B Lit	Kts.	1147 *

CEILING:

Combat Ceiling at Combat Weight, Rate of Climb = 500 F.P.M.

Maximum Thrust at 1.65 M.N. A/B Lit	Ft.	60,000

RATE OF CLIMB:

Steady Rate of Climb at Sea Level, Combat Weight

Maximum Thrust at M.N. = .92 A/B Lit	F.P.M.	60,600
Maximum Thrust at 527 Kts. A/B not Lit	F.P.M.	27,200

Steady Rate of Climb at 50,000 Ft., Combat Weight

Maximum Thrust at M.N. = 1.5 A/B Lit	F.P.M.	8,600

TIME TO HEIGHT:

Time to 50,000 Ft. M.N. = 1.5 from Engine Start at Take-Off Weight

Maximum Thrust A/B Lit	Mins.	4.33

MANOEUVRABILITY:

Combat Load Factor at Combat Weight

Maximum Thrust at M.N. = 1.50 at 50,000 Ft. A/B Lit		1.63

* AIR 7-4 Placard Speed

DRAG

PROPULSION

TAKE-OFF DISTANCE:

Take-Off Distance over 50 Ft. Obstacle at Sea Level at Take-Off Weight = 59,336 Lbs.

Maximum Thrust A/B Lit	Ft.	2,850
Maximum Thrust A/B not Lit	Ft.	4,430
Maximum Thrust Hot Day A/B Lit	Ft.	3,460

LANDING DISTANCE:

Landing Distance over 50 Ft. Obstacle at Sea Level at Normal Design Landing Gross Weight	Ft.	4,810

STALLING SPEED:

True Stalling Speed in Landing Configuration at Combat Weight at Sea Level	Kts.	111.5

RANGE:

Combat Radius of Action at 50,000 Ft. Climb at 527 Kts. T.A.S., Accel. to M = 1.5 @ 30,000', Climb @ M = 1.5 to 50,000', Cruise-out at M.N. = 1.5, Combat for 5 Mins. at M.N. = 1.50, Cruise-back at M.N. = .92, 15 Min. Stack at 40,000 Ft., 5 Min. Fuel Reserve on Landing

High Speed Mission with 15,672 Lbs. Fuel	N.M.	200.0
High Speed Mission with Full Internal Fuel (SG = 0.78)	N.M.	302.0

Combat Radius of Action at 50,000' Mission as above except Cruise-out at M.N. = .92

Maximum Range Mission with 15,744 Lbs. Fuel	N.M.	300.0
Maximum Range Mission with Full Internal Fuel (SG = 0.78)	N.M.	450.0

Ferry Range Mission at Economical Cruise Speed (Cruise climb from 36,500' to 41,500' at M = .92) including 15 Mins. Stacking at 40,000 Ft., 5 Min. Fuel Reserve on Landing

Range with Full Internal Fuel and 500 Gal. - External Tank (SG = 0.78)	N.M.	1460.0

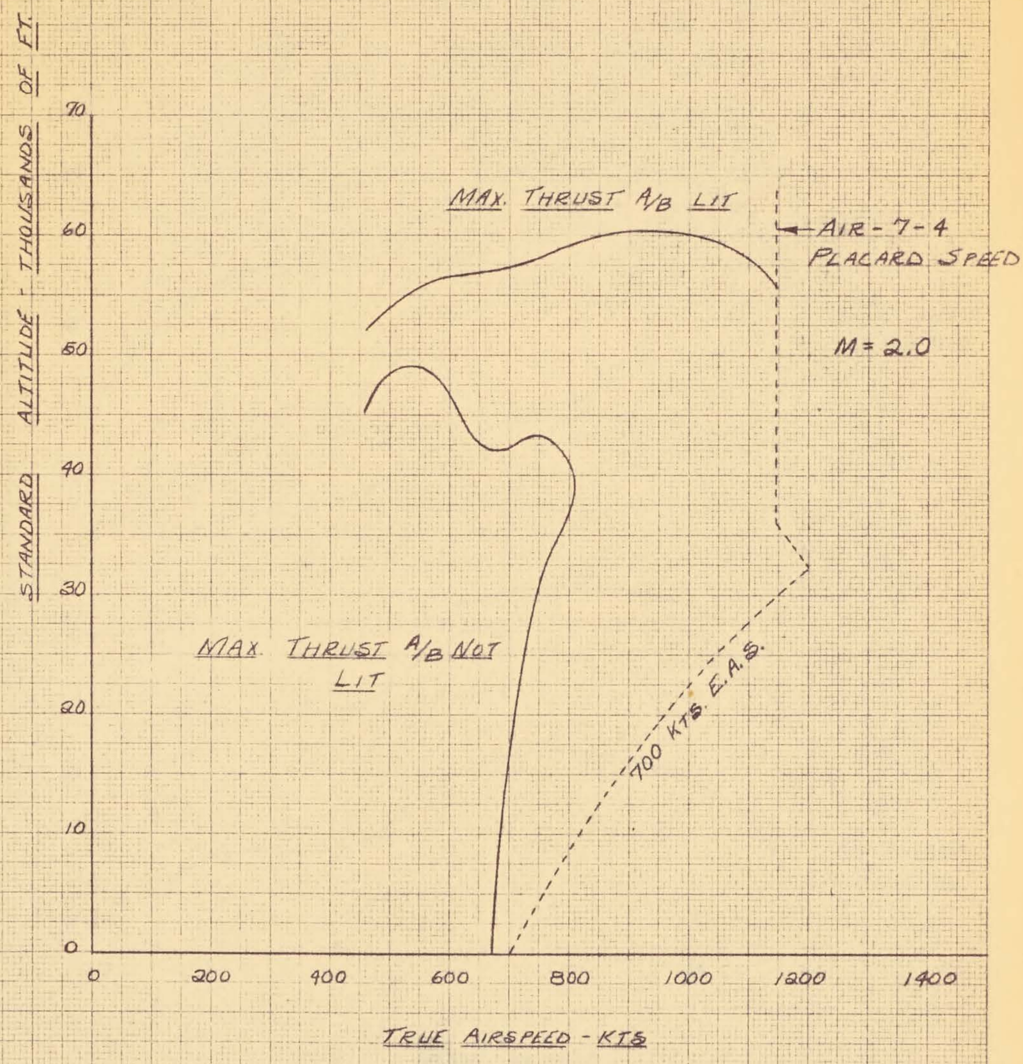
DRAG

PROPULSION

CF105

MAX. LEVEL SPEED AT COMBAT WEIGHT

IROQUOIS ENGINES



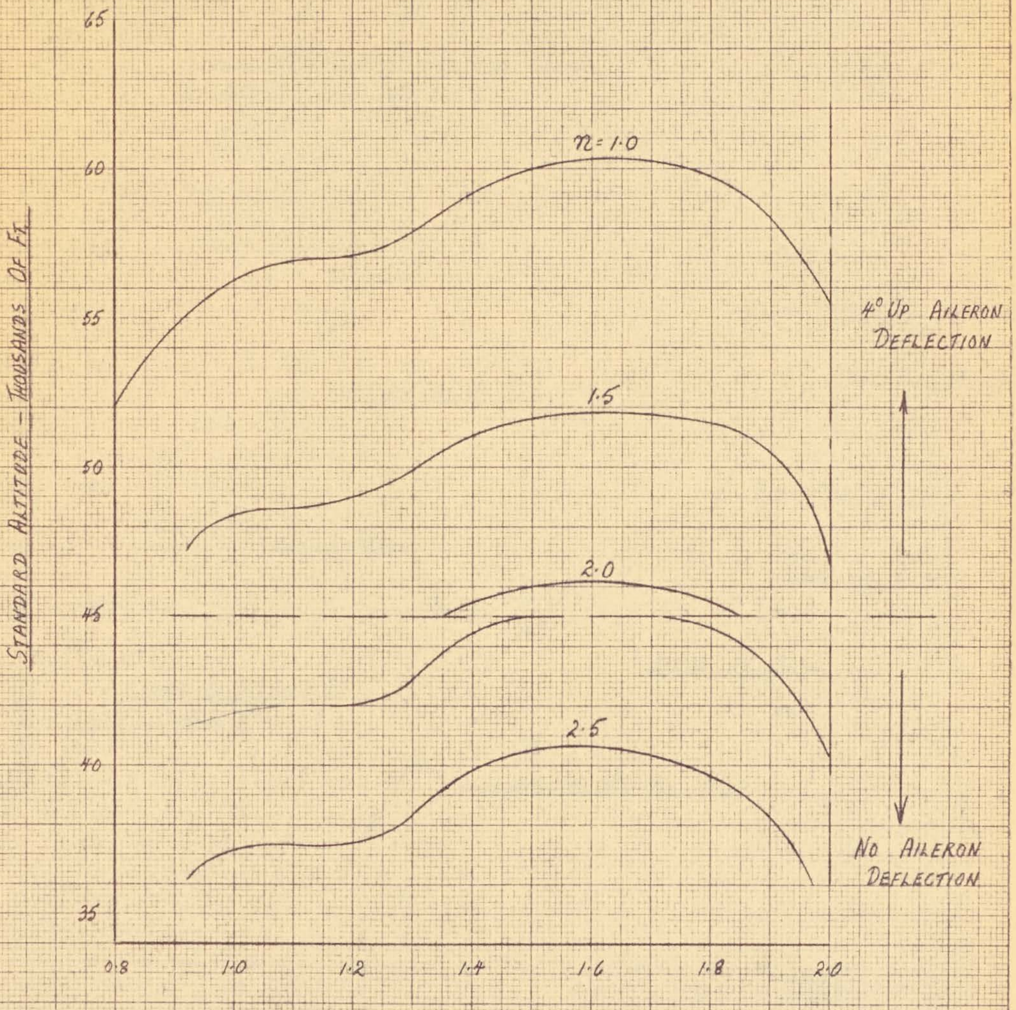
DRAG

PROPULSION

CF105 MANOEUVRABILITY

AVAILABLE STEADY g's AT COMBAT WEIGHT

IROQUOIS ENGINES WITH AFTERBURNERS LIT



K+E 10 X 10 TO THE CM. 359-14 KEUFFEL & ESSER CO. MADE IN U.S.A.

DRAG

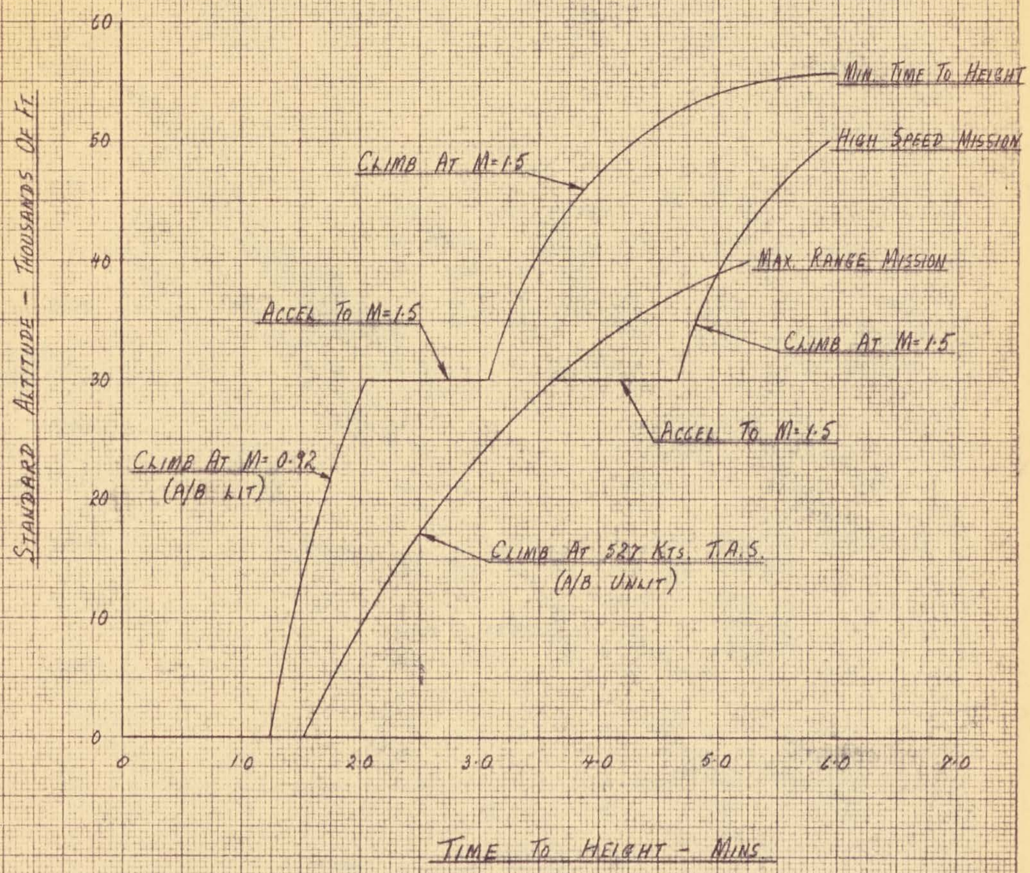
PROPULSION

CF105 TIME TO HEIGHT

IROQUOIS ENGINES

NOTE :- ½ MIN. ALLOWED FROM ENGINE START TO MAX. THRUST

ACCELERATION TO M=1.5 AND CLIMB
AT M=1.5 WITH A/B LIT.

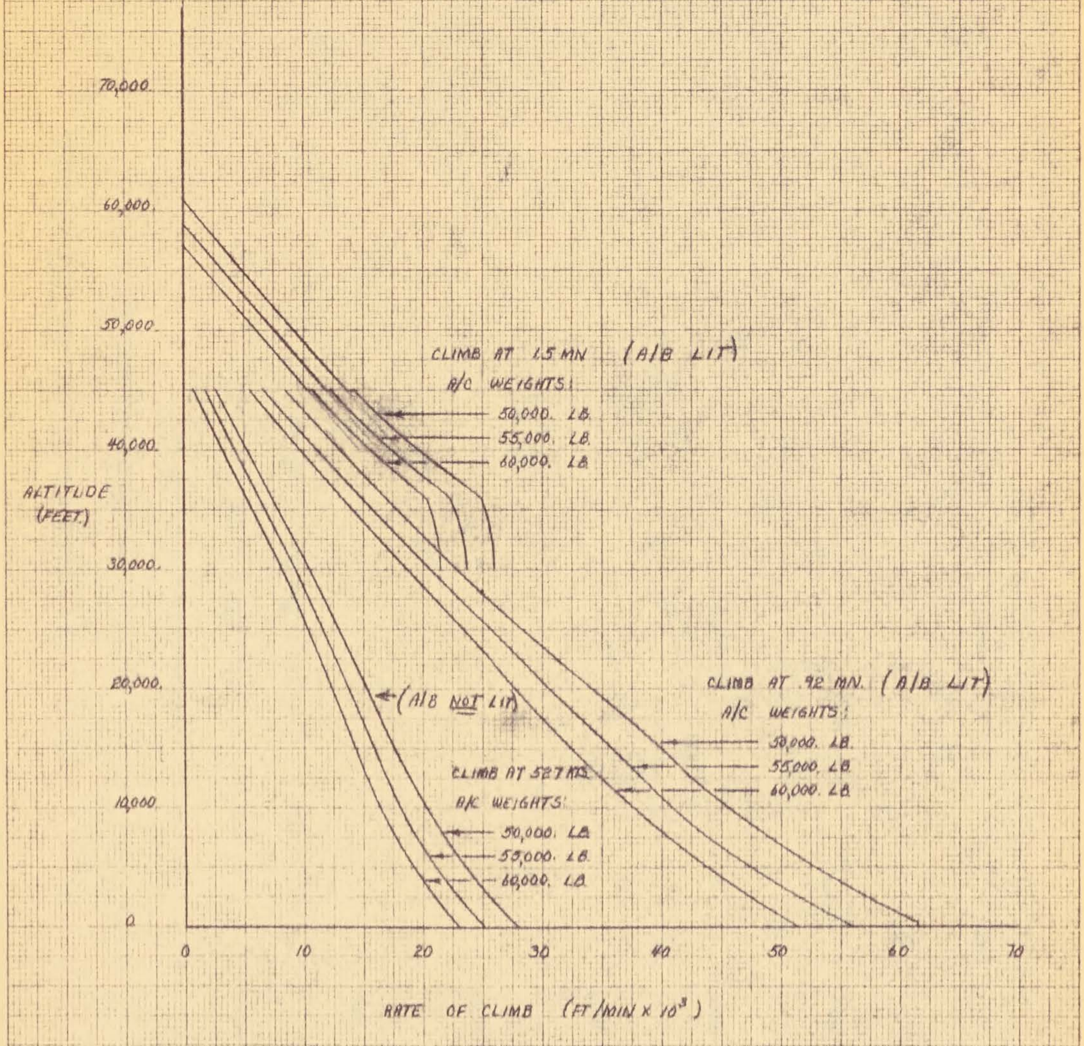


NOV. 1956

CF105

IROQUOIS ENGINE

RATE OF CLIMB (STEADY STATE)

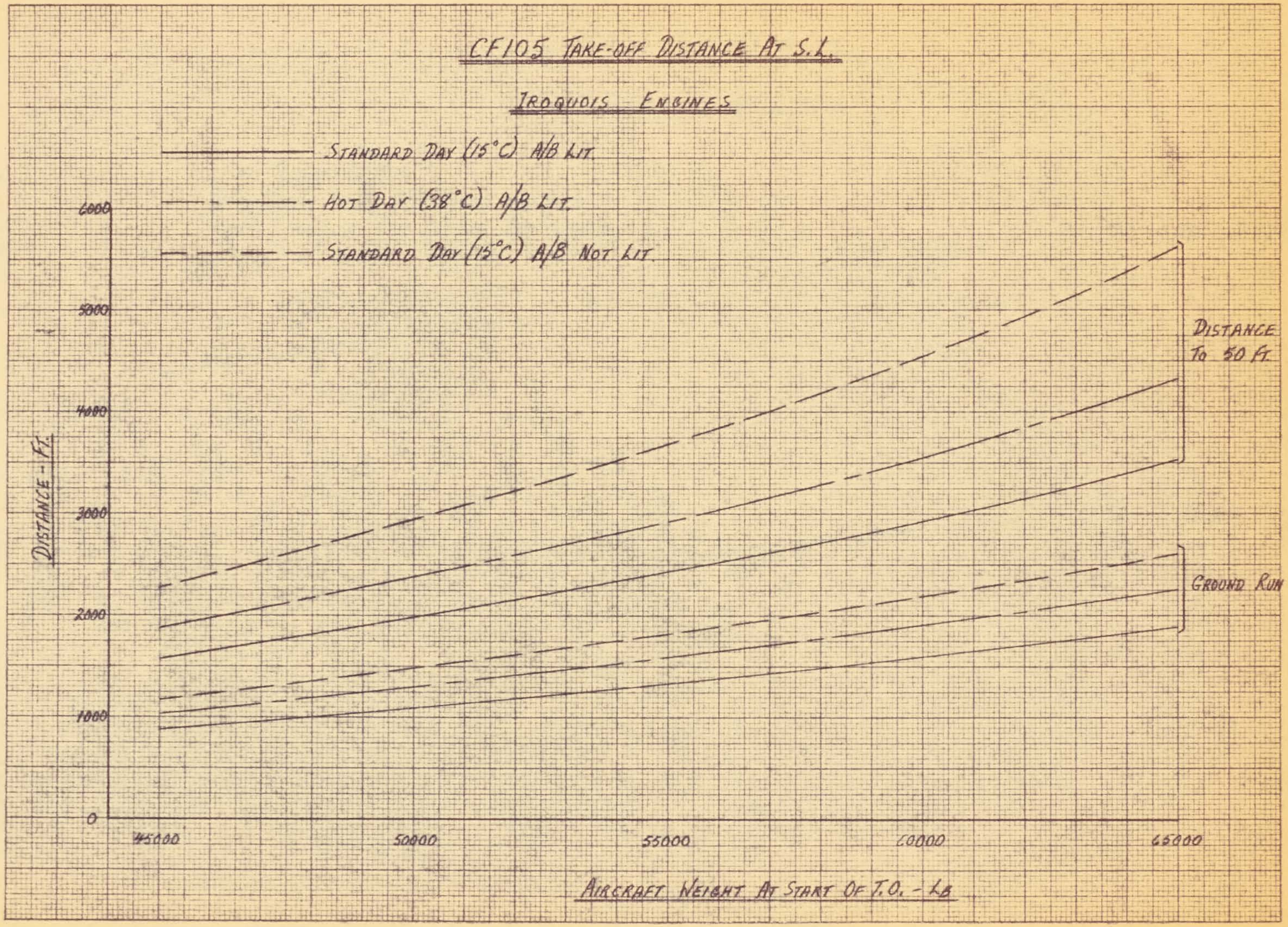


10 X 10 TO THE CM. 359-14
KEUFFEL & ESSER CO.

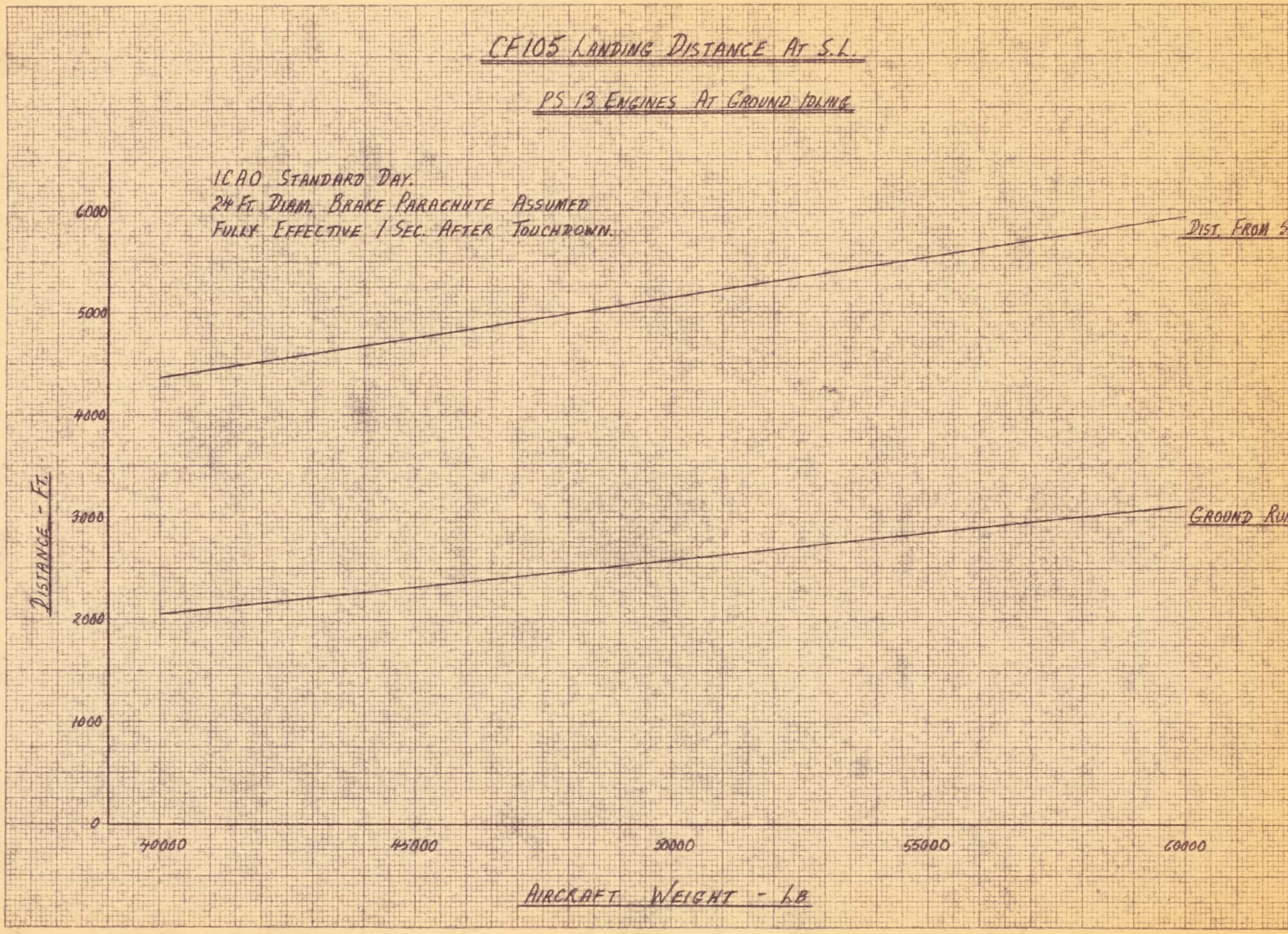
DRAG

PROPULSION

REPORT NO P/PERF/122



NOV 1956



OCT. 1956

DRAG

PROPULSION

2: CF-105 DRAG DATA

The following CF-105 pertinent drag data are based on the wind tunnel configuration designated $B_{11}\ V_1\ W_1\ E_{10}\ N_5\ D_{8-4}$. The particular features of this configuration are the extended, notched and cambered leading edge of the wing. Provision is made for 4^o up aileron deflection at altitudes greater than 45,000 feet. This data was obtained from wind tunnel tests at Langley Field, Cornell Aeronautical Laboratories and the National Aeronautical Establishment, together with "Free Flight Model" tests at the Langley Free Flight Test Range. The actual test data has been used for the first time. In the past this has not been possible, extrapolation being necessary beyond M = 1.23. Because of the non linearity of the derivatives it is considered preferable to use the actual test data in the manner outlined below. Combining this data from all sources enabled an overall drag picture from 0.2 Mach Number to Mach 2.0.

The wind tunnel data used is as follows:-

N.A.E.	Wind Tunnel Test	December, 1955	M = 0.21
C.A.L.	Wind Tunnel Test	May, 1955	M = 0.5 to 1.23
Langley	Wind Tunnel Test	March, 1956	M = 1.41
Langley	Wind Tunnel Test	June, 1956	M = 1.6; 1.8 and 2.0

The data of the three latter tests was corrected to the exact height of the c.g. below the reference axis, and allowance was made for the shift in c.g. from 28% M.A.C. on the model to 29.5% M.A.C. for the actual aircraft.

In the case of the N.A.E. tests investigation showed that C_D vs C_L could be taken as independent of M up to approximately M = 0.8. Furthermore it was found that satisfactory accuracy, in as far as the aircraft performance was concerned, could be achieved by ignoring trim drag and power effects within a c.g. range of 28% to 31% M.A.C.

Above M = 0.8 the data was corrected for c.g. height and position. A trim carpet, C_L vs δ_{Trim} and M.N., was drawn in order to determine the elevator angle required to trim the aircraft at various lift coefficients throughout the Mach Number range. Since all the above wind tunnel tests were conducted with a rigid model which did not simulate the structural elasticity of the full scale aircraft, these elevator angles were designated δ_{Trim} (Rigid). Investigation of the effect of structural elasticity and the thrust momentum change between model and full scale showed that these effects on the elevator angle required to trim were not significant together as they tend to cancel each other and can therefore be ignored.

PROPULSION

With this trim data and the tunnel drag data, the drag increment to trim for any given C_L throughout the Mach Number range was determined. This C_D for a δ_{trim} was found to be independent of C_L, thus enabling a single carpet of C_D vs δ_{trim} and M to be constructed.

Using actual elevator angles and corrected $C_{D_{min.}}$ corresponding to values obtained from Free Flight Tests a set of C_L vs C_D curves, for each Mach Number under test, with the elevator at zero degrees deflection was determined.

From this data the drag of the aircraft was evaluated in the following manner:-

$$C_{D_{Total}} = C_{D_{\delta_e = 0}} + \Delta C_D$$

i.e. Total Drag = Drag with zero elevator deflection plus the drag increment due to elevator deflection required to trim.

The trim drag of the elevator is proportional to the square of the deflection. Hence, if some assistance is given to the elevator by the aileron in producing a trimming moment, there will be a net reduction in drag.

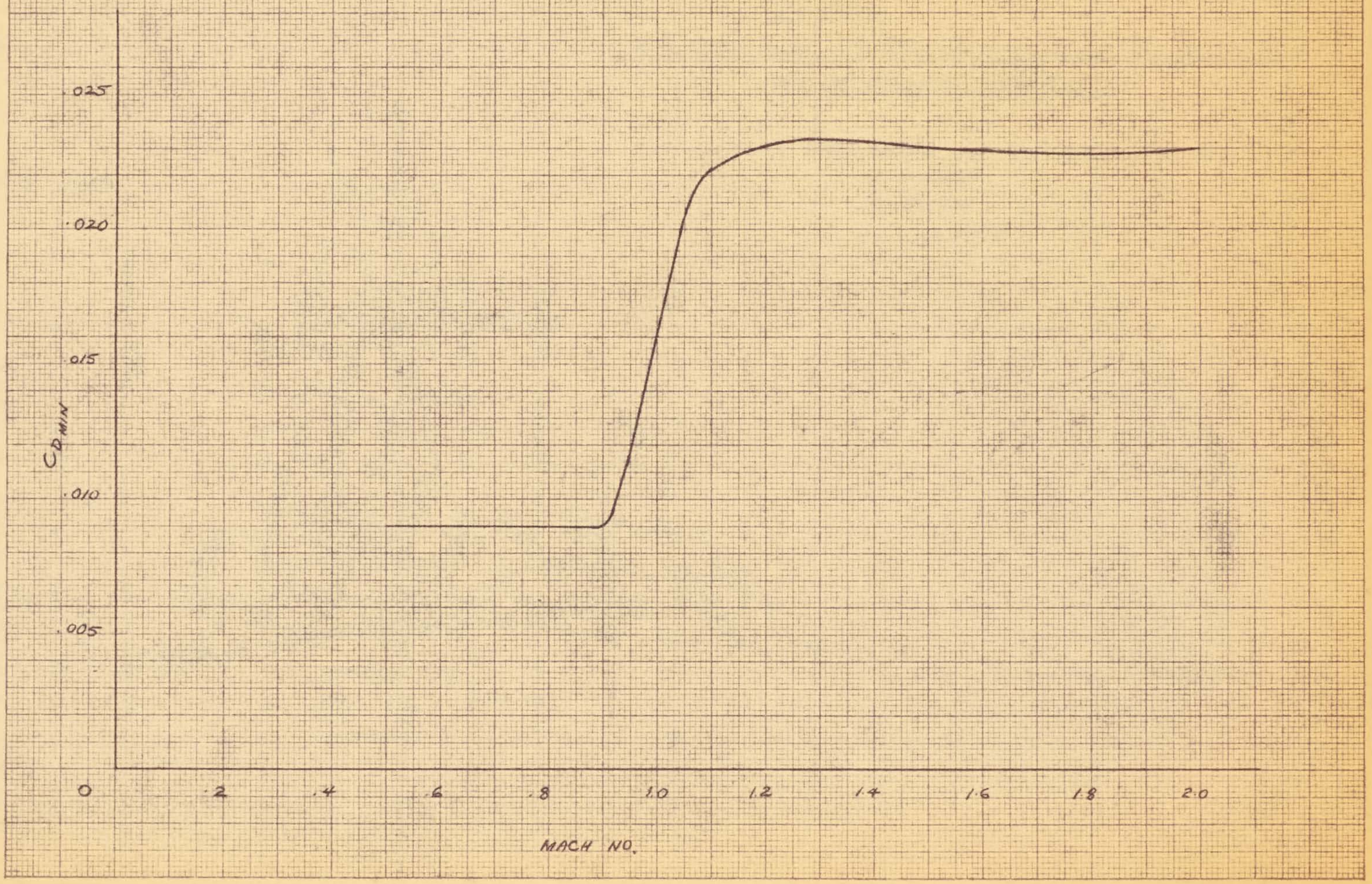
The drag of this aircraft with 4° of symmetrical aileron up-trim has therefore been evaluated. This trim will be automatically activated at altitudes above 45,000 feet, so as to avoid any difficulties arising from hinge moment limitations at the high values of "q" which occur at the lower altitudes.

The drag of the deflected ailerons is taken to be proportional to the relative effectiveness of the ailerons to the elevators, approximately 50%. The effect of structural elasticity, thrust momentum change and aileron deflection have been combined and evaluated as a $\Delta\delta_{e_{trim}}$ at 50,000 feet and nominal weight of 47,000 lb. with varying g's to cover the requisite range of C_L's. The altitude of 50,000 feet was selected as being of greatest significance in as far as the aircraft performance is concerned. Thus for the configuration with 4° up aileron deflection we have,

$$\delta_{e_{Trim}} = \delta_{e_{Trim\ (Rigid)}} + \Delta\delta_{e_{Trim}}$$

PROPULSION

C 105 ~ $C_{D_{MIN}}$ vs MACH NO.



P/AERO DATA/74

C-105

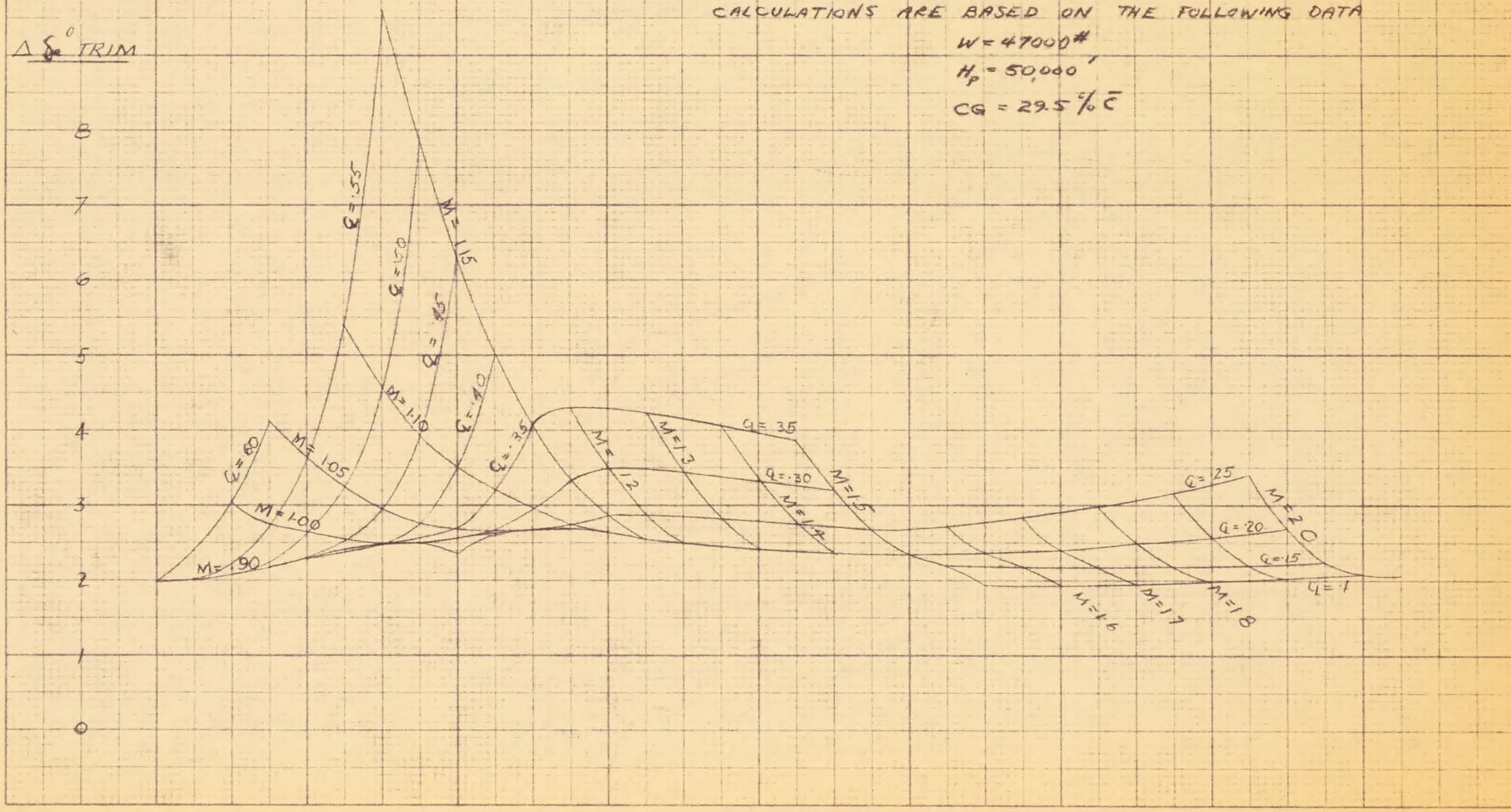
$\Delta\delta_e$ TRIM DUE TO POWER AND ELASTIC EFFECTS AND -4° AILERON DEFLECTION

CALCULATIONS ARE BASED ON THE FOLLOWING DATA

$W = 47000$#

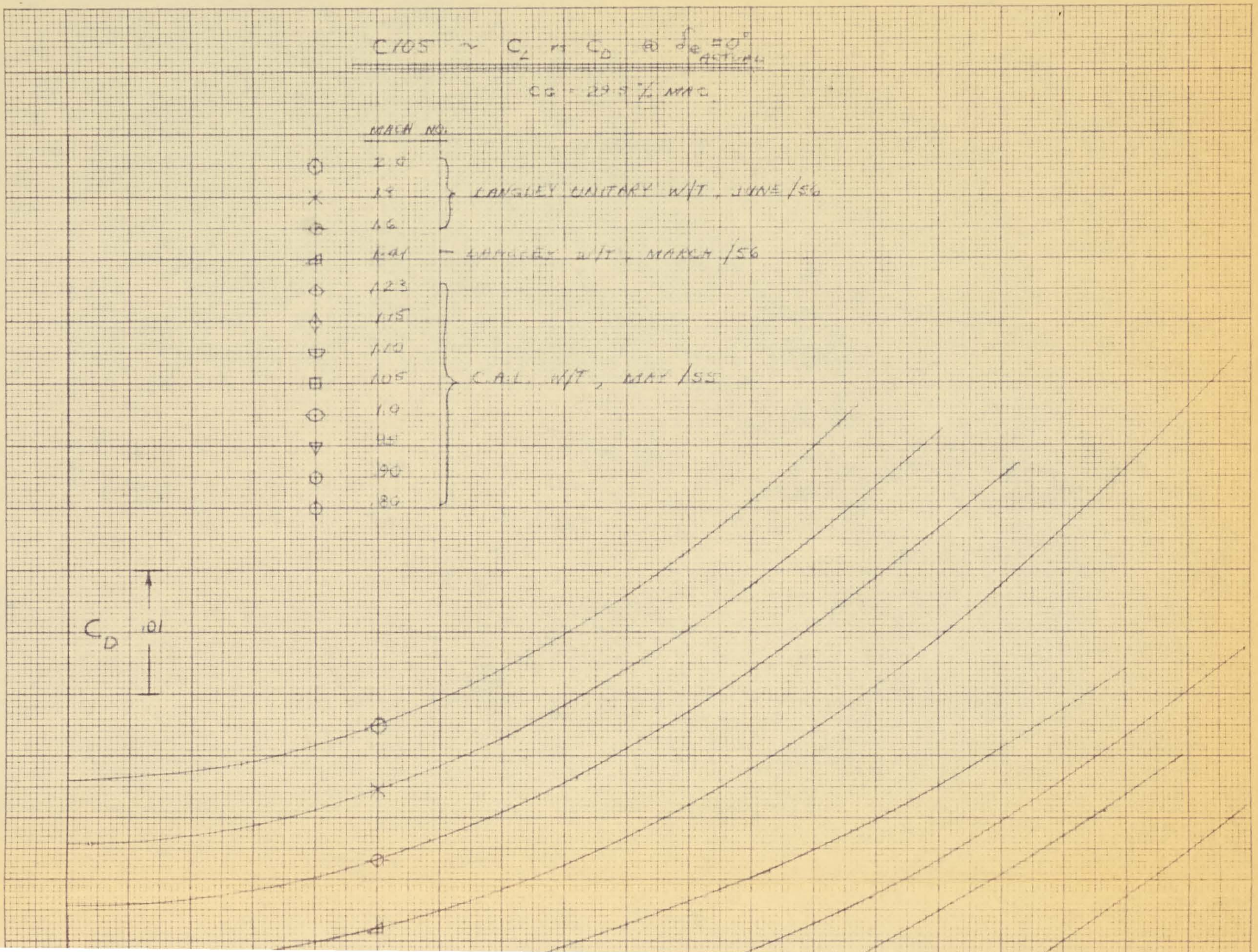
$H_p = 50,000'$

$CG = 29.5\%\bar{c}$

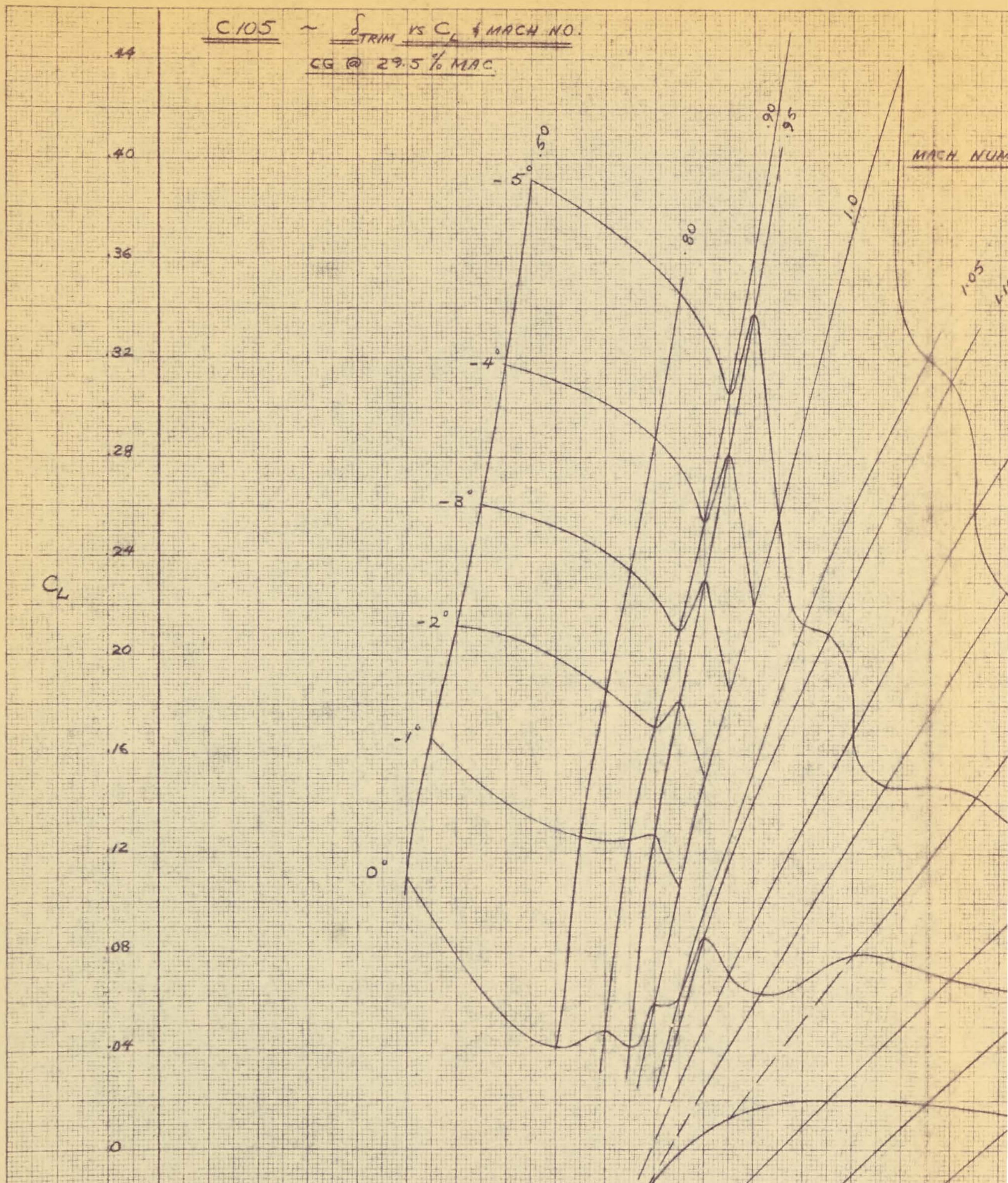


DEC /56

PROPULSION

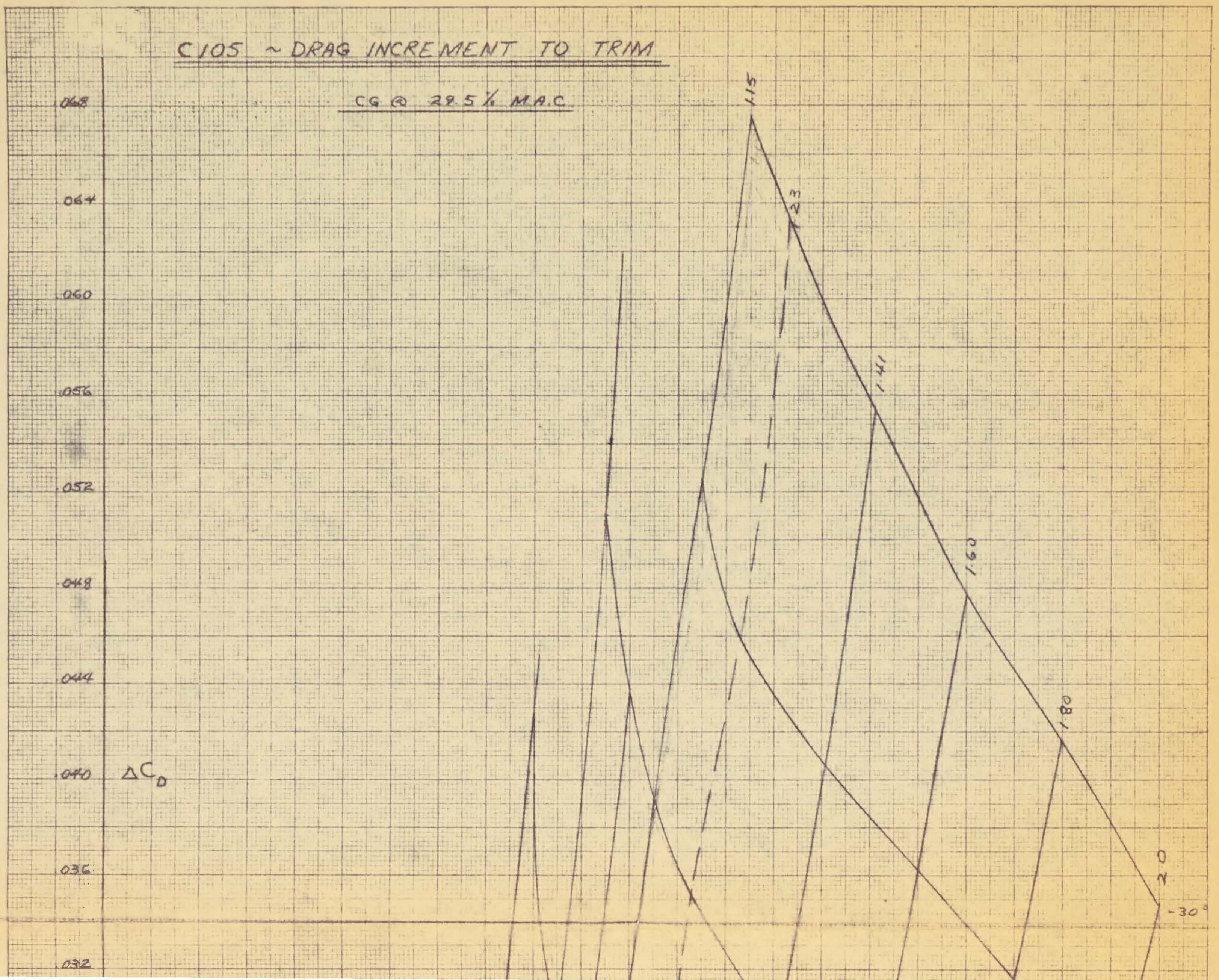


C105 ~ C_L vs C_D @ δe = 0° ACTUAL
CG = 29.9 % MAC
MACH NO.
2.0
1.6
1.41
1.23
1.15
1.10
1.05
1.0
.90
.80
LANGLEY UNITARY W/T, JUNE /56
MARCH /56
C.A.L. W/T, MAY /55
C_D
.01



K+E 10 X 10 TO THE CM. 359-14L
KEUFFEL & ESSER CO.

10 X 10 TO THE CM. 359-14L
KEUFFEL & ESSER CO.

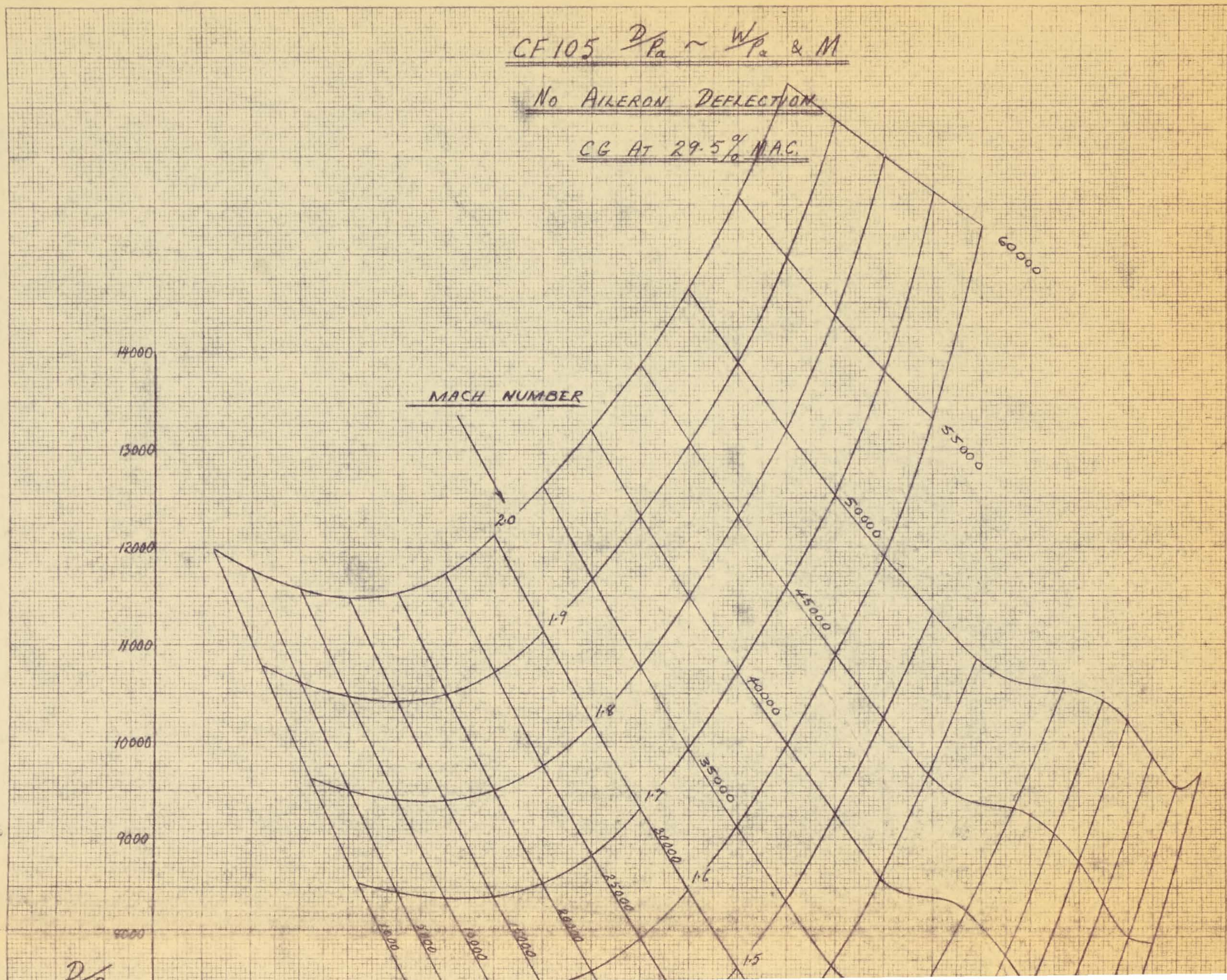


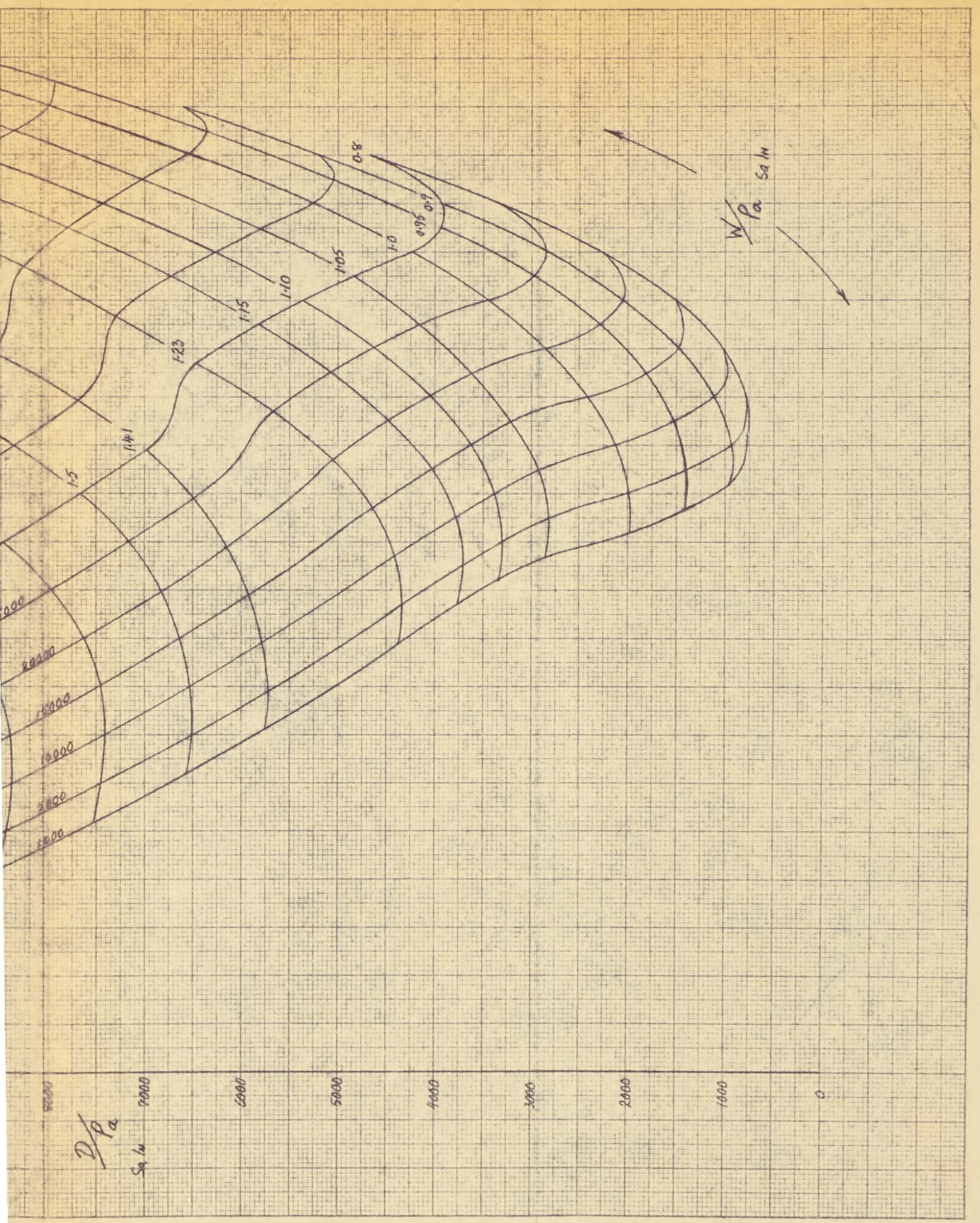
Nov 1956

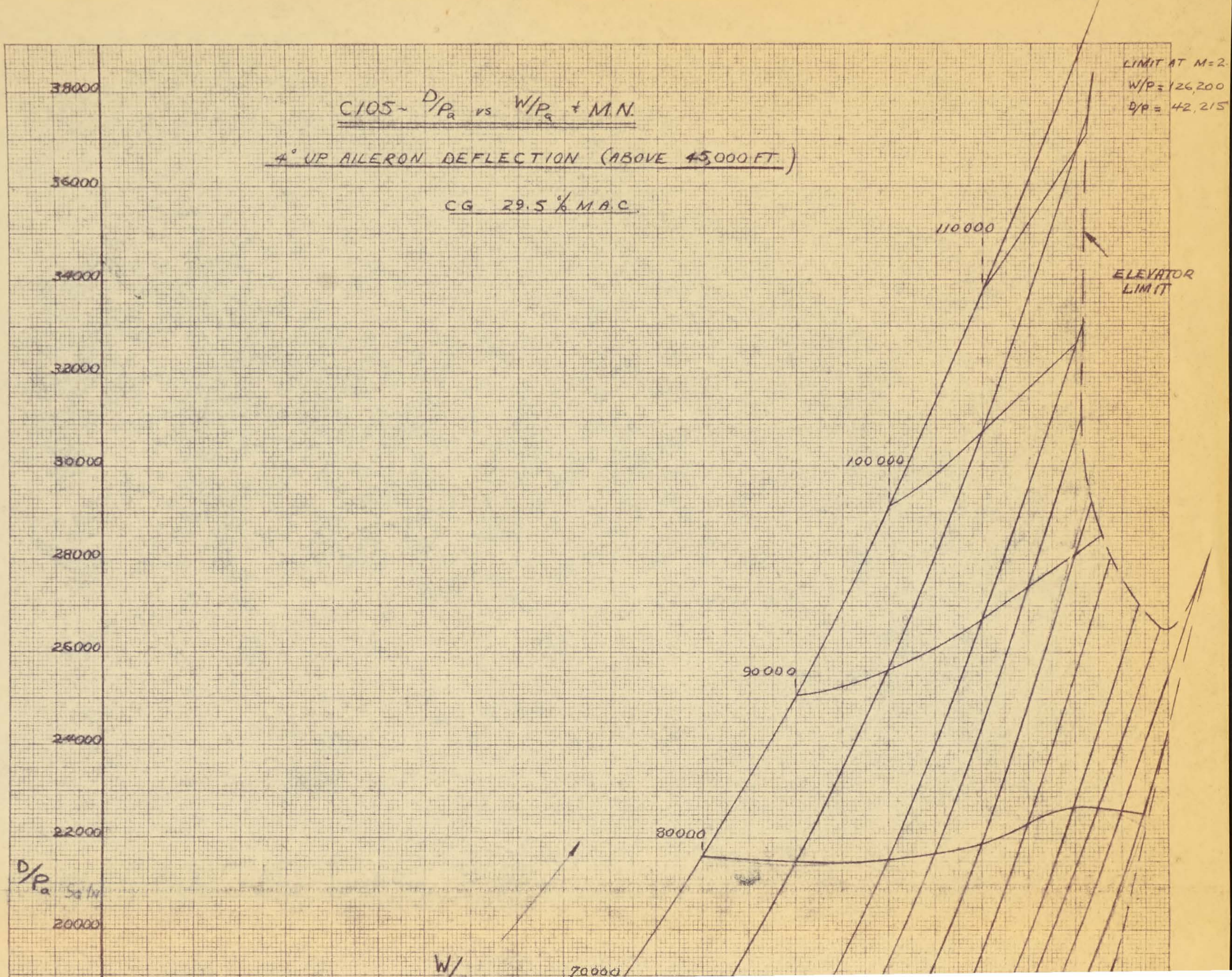
CF105 D/Pa ~ W/Pa & M

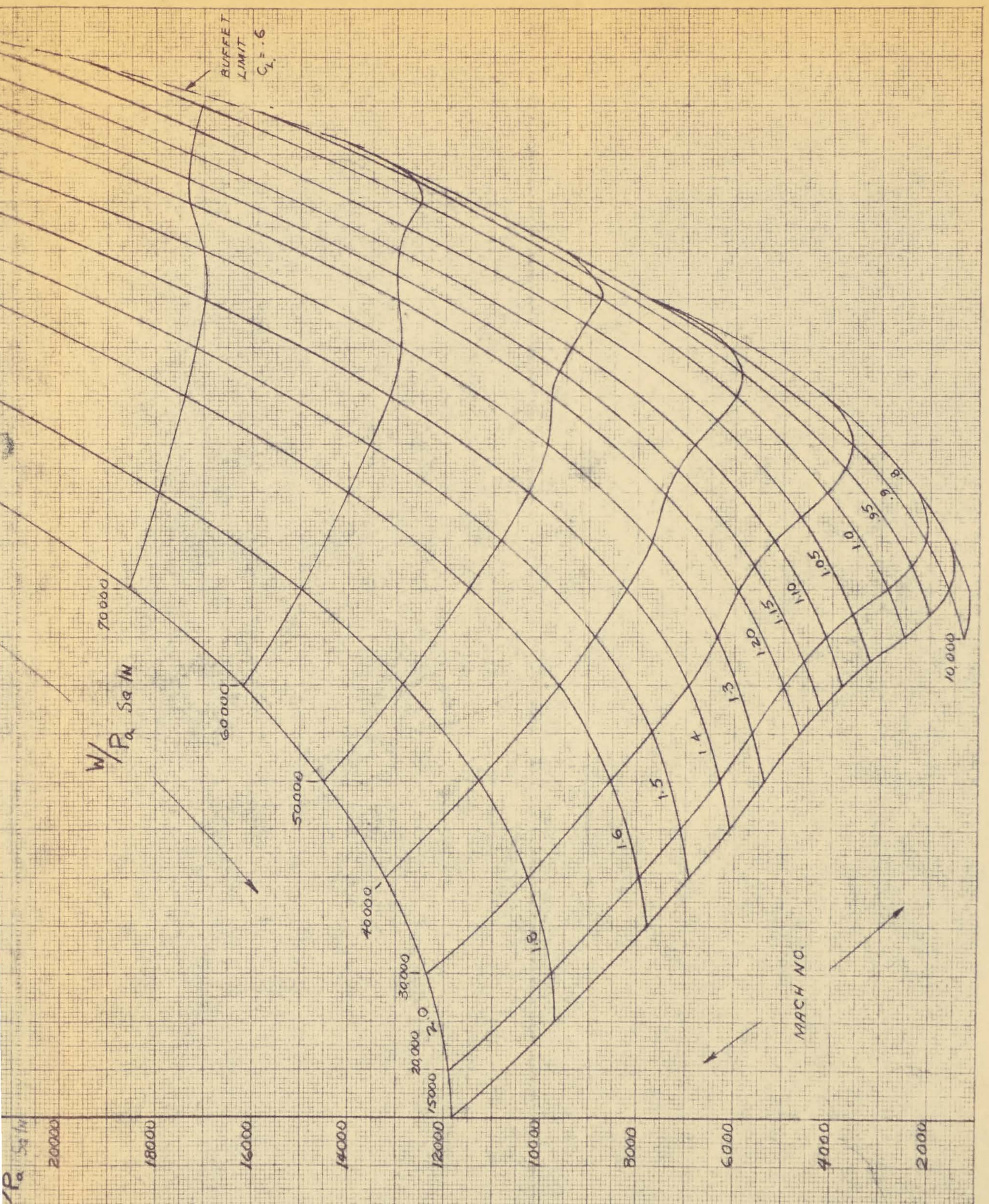
No Aileron Deflection

CG At 29.5% MAC.




W/Pa Sq In
0.8
0.9
0.95
1.0
1.05
1.10
1.15
1.23
1.41
1.5
20000
15000
10000
5000
1000
D/Pa Sq In
7000
6000
5000
4000
3000
2000
1000
0


C105 ~ D/Pa vs W/Pa + M.N.
4° UP AILERON DEFLECTION (ABOVE 45,000 FT)
CG 29.5 % M.A.C
LIMIT AT M=2.
W/P = 126,200
D/P = 42,215
ELEVATOR LIMIT
110000
100000
90000
80000
70000
38000
36000
34000
32000
30000
28000
26000
24000
22000
20000
D/Pa
W/


BUFFET LIMIT
$C_L = .6$
W/Pa Sq In
70000
60000
50000
40000
30,000
20,000
15000
10,000
2.0
1.8
1.6
1.5
1.4
1.3
1.20
1.15
1.10
1.05
1.0
.95
.9
.8
MACH NO.
20000
18000
16000
14000
12000
10000
8000
6000
4000
2000

PROPULSION

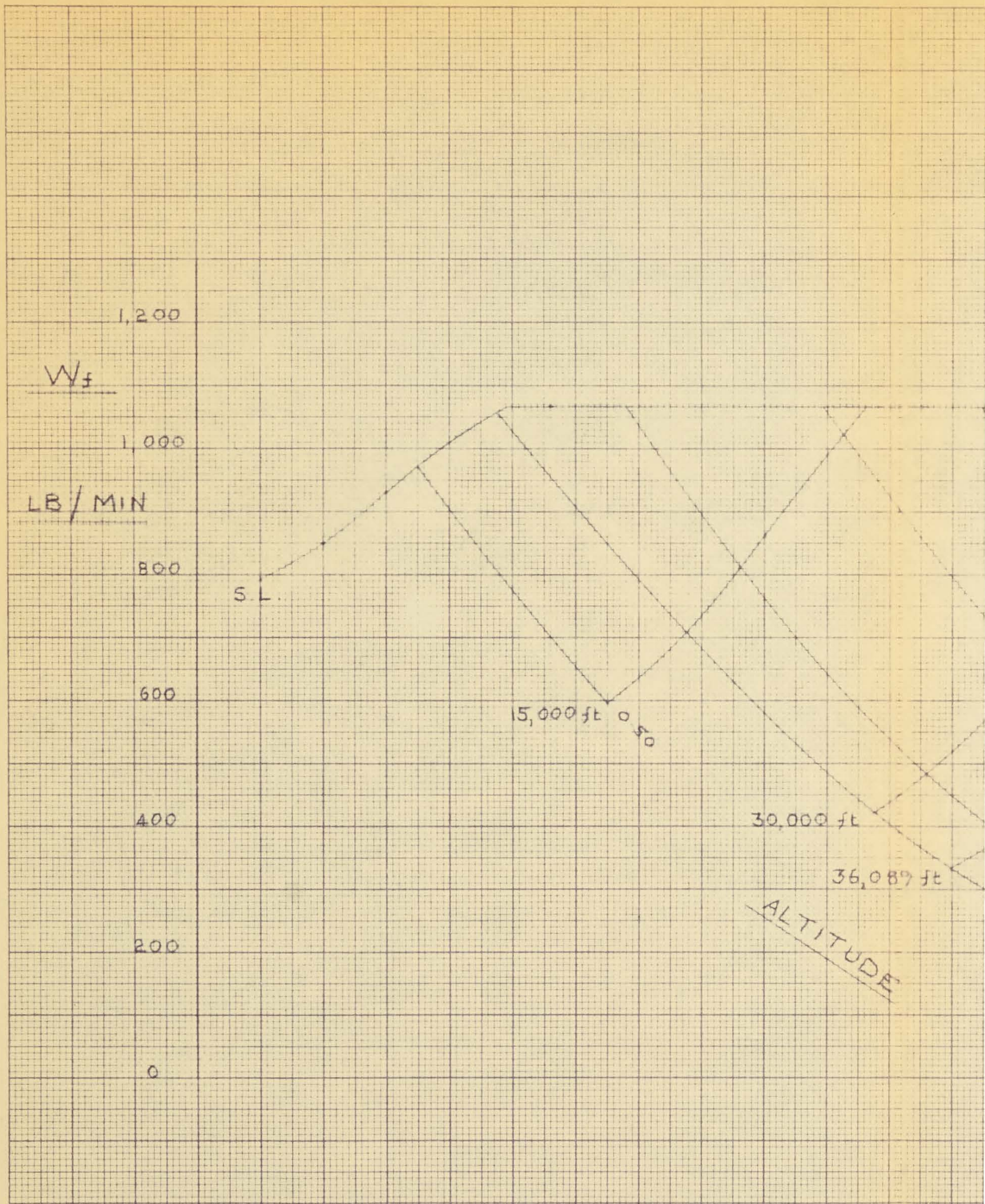
3: PROPULSION

The installed Iroquois engine data has been revised. Essentially the change is to a 42.8" divergent ejector instead of the 40" cylindrical ejector for which the performance is given in Monthly Report #8. All new data and revisions during the interval have also been included. The more significant of these are as follows:

- altered bypass geometry; i.e. the bypass area has been increased from 60 sq.in. to 180 sq.in.; the diffusion area ratio has been increased from 1.25 to 1.50; re-estimate of bypass temperature rise.
- additional data from Orenda on fuel limitations and afterburner fuel flows.
- correction of error in partial non-afterburning thrust vs fuel flows.

NOTE:

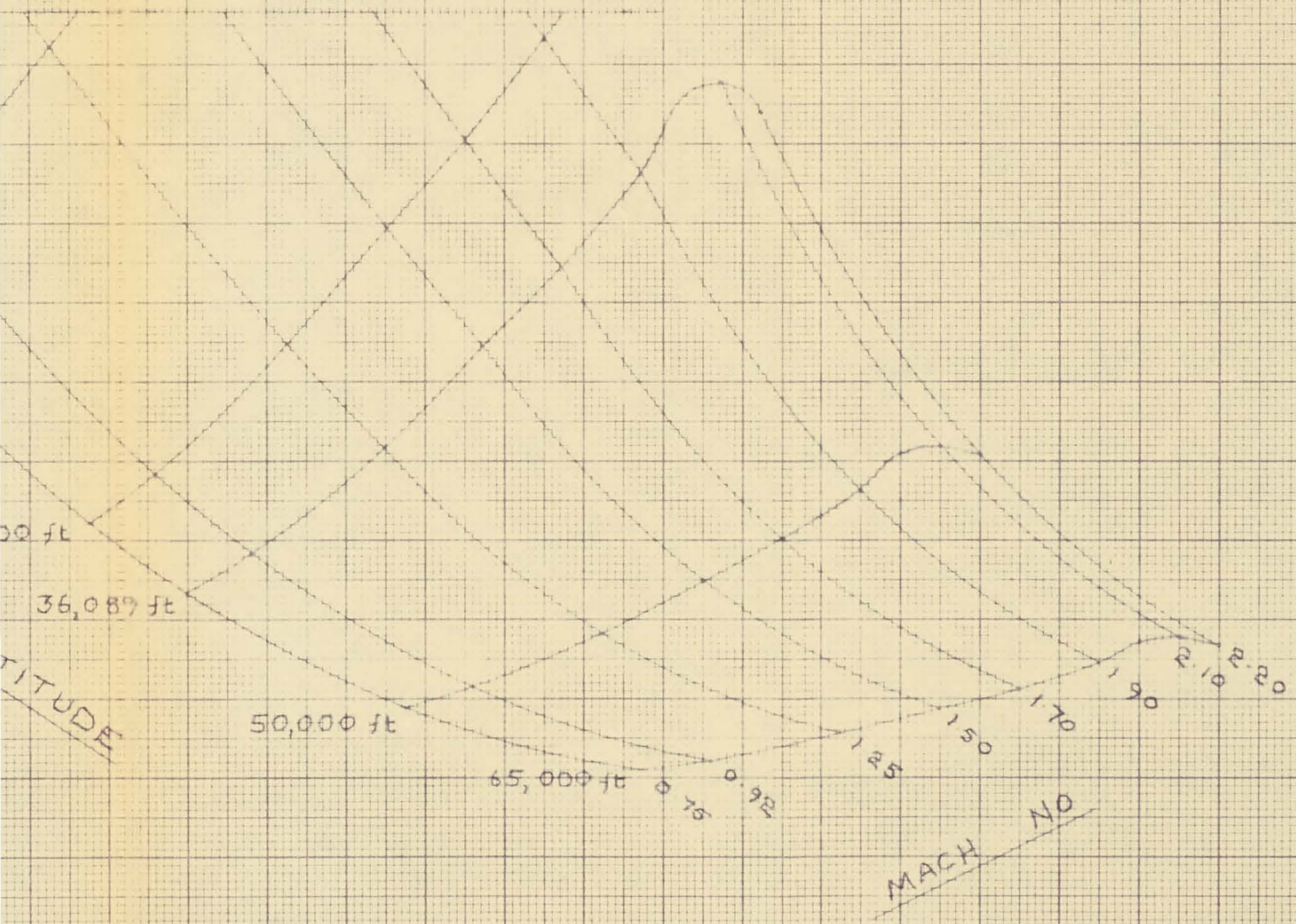
Since data prepared for C.A.S. Presentation on December 6th/56 the Iroquois A/B fuel flows have been revised, showing an increase in fuel for the high speed mission of 875 lb. This results in an increased combat weight, decreasing the combat 'g' from 1.65 to 1.63.

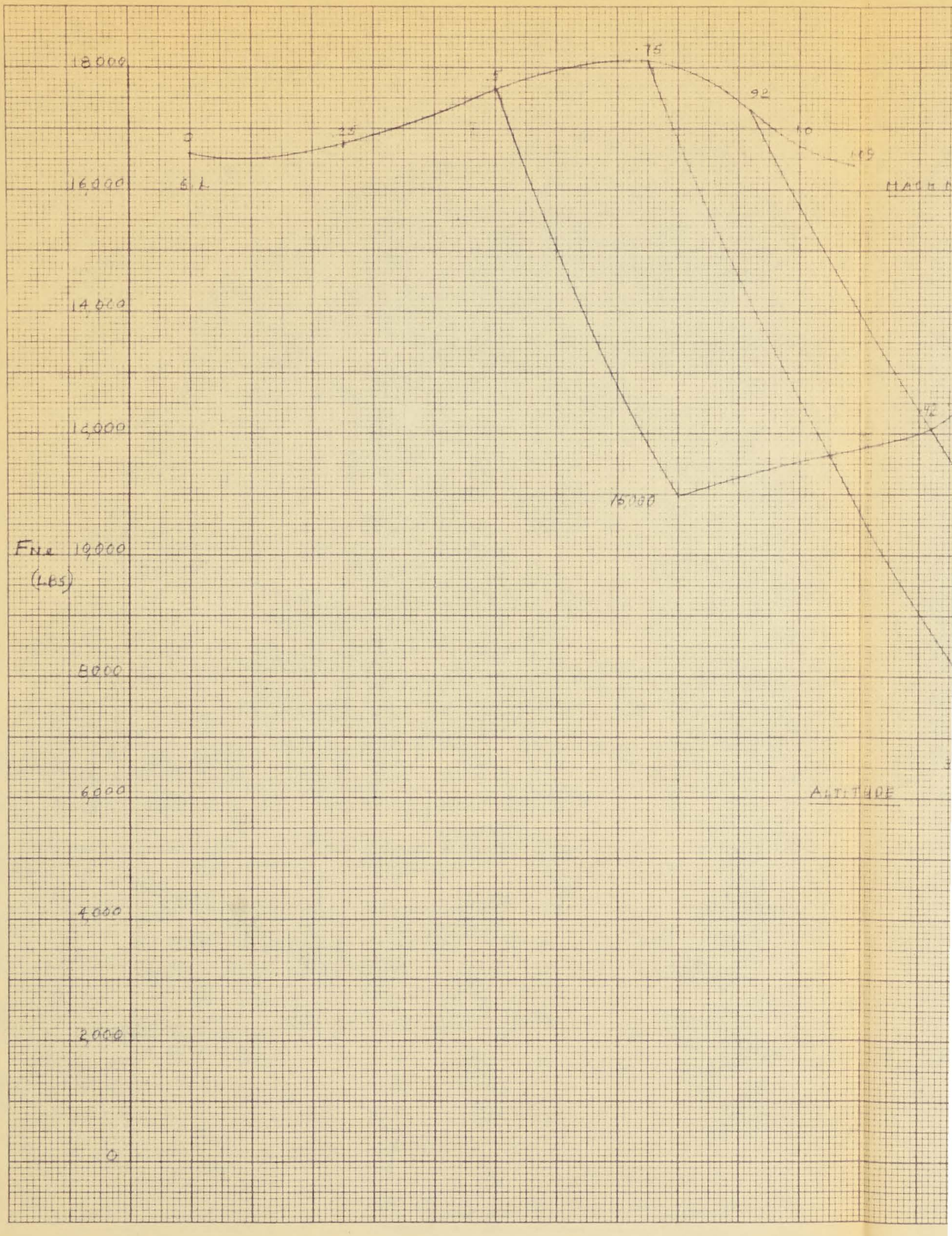

1,200
W_f
1,000
LB / MIN
800
S.L.
600
15,000 ft
400
30,000 ft
36,089 ft
200
0
ALTITUDE

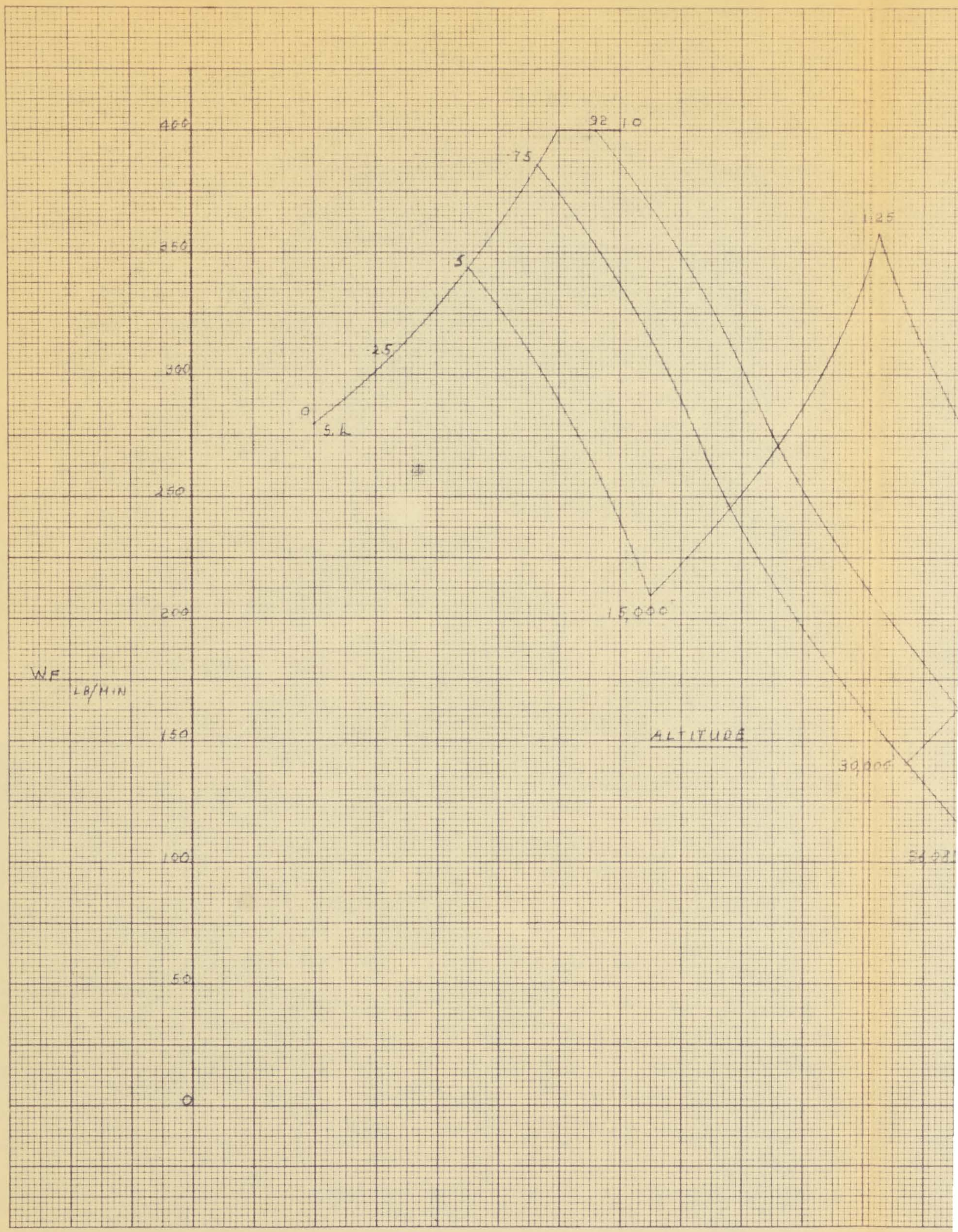
IROQUOIS FUEL FLOW

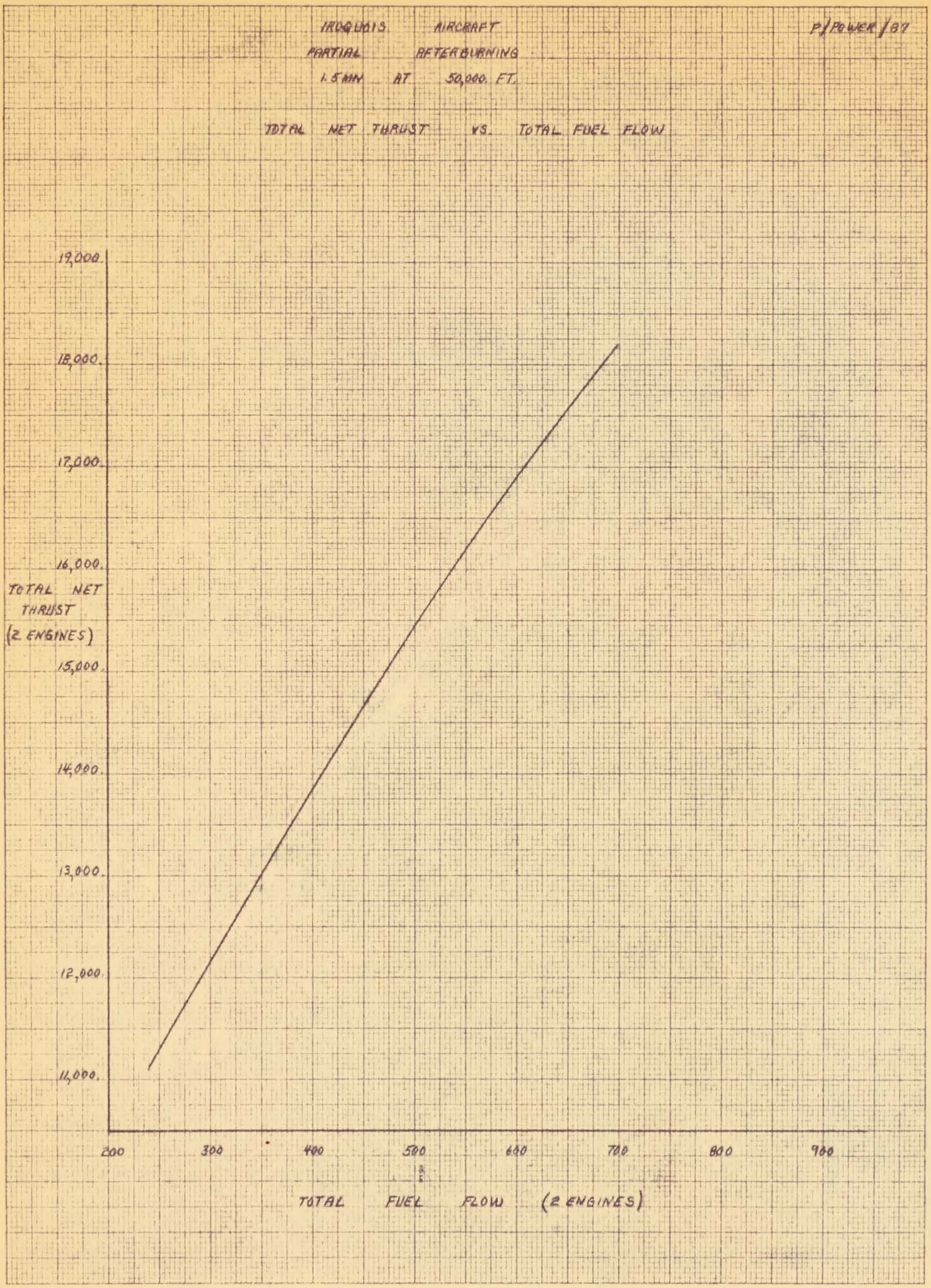
ONE ENGINE

AFTERBURNER LIT




18000
16000
14000
12000
10000
8000
6000
4000
2000
0
FNe
(LBS)
S.L.
.75
92
ALTITUDE


400
350
300
250
200
150
100
50
0
WF
LB/MIN
.92
1.0
.75
1.25
.5
.25
0
S.L.
15,000
30,000
ALTITUDE



IROQUOIS AIRCRAFT
PARTIAL AFTERBURNING
1.5 MN AT 50,000 FT.
P/POWER/87
TOTAL NET THRUST VS. TOTAL FUEL FLOW
19,000
18,000
17,000
16,000
15,000
14,000
13,000
12,000
11,000
TOTAL NET THRUST (2 ENGINES)
200
300
400
500
600
700
800
900
TOTAL FUEL FLOW (2 ENGINES)