CF-105 

MONTHLY PERFORMANCE REPORT

NO. 1

UNLIMITED SECRET

October 1955.

TO
A

DATE

Report no.: QC AVRO CF105 MR-1

has been
☐ downgraded to:
☑ de-classified

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

(Dept.): DND Coordinator - Access to Information

Date: Aug 4 '92

Renée Auger
Signature

A. V. ROE CANADA LIMITED
MALTON - ONTARIO

# TECHNICAL DEPARTMENT (Aircraft)



AIRCRAFT CF-105

REPORT NO. Monthly Report No. 1

FILE NO.

NO. OF SHEETS:

TITLE



PERFORMANCE

DRAG

ENGINE

## CF-105 MONTHLY PERFORMANCE REPORT

(Issued Mid-Monthly)

This is Copy Number 9
Issued to: R.C.A.F.
Date October 15th, 1955.



PREPARED BY DATE Oct. 1955.

CHECKED BY DATE

SUPERVISED BY DATE

APPROVED BY DATE



| ISSUE No | REVISION No | REVISED BY | APPROVED BY | DATE | REMARKS |
|---|---|---|---|---|---|
| | | | | | |
| | | | | | |
| | | | | | |
| | | | | |  |



FORM 1316A

# CF-105 MONTHLY PERFORMANCE REPORT

(Issued Mid-Monthly)

## INTRODUCTION

This is the first of a series of monthly performance reports for internal usage, to be issued from the Aerodynamics Department. 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. Engine Data

PERFORMANCE

PERFORMANCE

SECRET

UNLIMITED

1. <u>CF-105 PERFORMANCE</u>

The performance in this issue is sub-divided into two parts:

1A. CF-105 Performance with Pratt and Whittney JT4A-25 Engines

1B. CF-105 Performance with Orenda PS 13 Engines

SECRET

UNLIMITED

## 1A. CF-105 PERFORMANCE WITH PRATT AND WHITTNEY JT4A-25 ENGINES
(C.G. = 29% M.A.C.)

The following CF-105 - JT4A-25 performance estimate is based on the wind tunnel configuration designated $B_2V_1W_1E_{10}N_5D_8$-4. The particular feature of this configuration is the extended, notched, and cambered leading edge of the wing. The drag of this configuration is summarized (Extract P/Aero Data/58) and is presented in section 2 of this report.

The installed engine data is summarized (Extract P/Power/51) and is presented in section 3 of this report. Of particular interest, is the use of an ejector for improved performance.

LOADING AND PERFORMANCE

Performance Under N.A.C.A. Standard Atmospheric Conditions

SECRET

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

(With Two J75 Engines)

WEIGHT:

| | | |
|---|---|---|
| Take-Off Weight with 15,298 Lb. Fuel (77.1% Max.) | Lb. | 58,982 |
| Operational Weight Empty | Lb. | 43,684 |
| Combat Weight (1/2 Fuel) | Lb. | 51,333 |
| Landing Weight (With Reserve Fuel + Missiles) | Lb. | 44,200 |
| Wing Loading at Normal Take-Off Weight | Lb. /Sq.Ft. | 47.0 |
| Power Loading at Normal Take-Off Weight | Lb. /Lb. Thrust | 1.61 |

SPEED

True Air Speed In Level Flight
At Sea Level at Combat Weight

| | | |
|---|---|---|
| Maximum Thrust | Kts. | ☆ 755 |
| Military Thrust | Kts. | 640 |

True Air Speed in Level Flight
At 50,000 Ft. at Combat Weight

| | | |
|---|---|---|
| Maximum Thrust | Kts. | 1,147 |

CEILING

| | | |
|---|---|---|
| Combat Ceiling at Combat Weight, Rate of Climb = 500 F.P.M. Maximum Thrust at 1.5 M.N. | Ft. | 57,200 |

RATE OF CLIMB

Steady Rate of Climb at Sea Level, Combat Weight

| | | |
|---|---|---|
| Maximum Thrust at M.N. = .92 | F.P.M. | 51,400 |
| Military Thrust at 530 Kts. | F.P.M. | 15,800 |

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

| | | |
|---|---|---|
| Maximum Thrust at M.N. = 1.5 | F.P.M. | 7,700 |

TIME TO HEIGHT

Time to 50,000 Ft. M.N. = 1.5 from Engine Start at Take-Off Weight = 58,982

| | | |
|---|---|---|
| Maximum Thrust | Mins. | 4.4 |

MANOEUVRABILITY

| | |
|---|---|
| Combat Load Factor at Combat Weight Maximum Thrust at M.N. = 1.50 at 50,000 Ft. | 1.50 |
| Combat Load Factor at Combat Weight Maximum Thrust at M.N. = 1.70 at 50,000 Ft. | 1.65 |

☆ Placard Speed = 720 Kts.

SECRET

SECRET

TAKE-OFF DISTANCE

Take-Off Distance over 50 Ft. Obstacle at Sea Level
Take-Off Weight = 58,982 Lb.

| | | |
|---|---|---|
| Maximum Thrust ........................................ | Ft. | 3,400 |
| Military Thrust ........................................ | Ft. | 6,700 |
| Maximum Thrust, Hot Day ................................ | Ft. | 4,600 |

LANDING DISTANCE

Landing Distance over 50 Ft. Obstacle at Sea Level at Combat Weight Ft. 5,300

STALLING SPEED

True Stalling Speed in Landing Configuration at Combat Weight at Sea Level .................................... Kts. 110

RANGE

Combat Radius of Action at 50,000 Ft., Climb at M.N. = .92, 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 15,298 Lb. Fuel .................. | N.M. | 200 |
| High Speed Mission with Full Internal Fuel ................ | N.M. | 309 |

Combat Radius of Action at 50,000 Ft., Mission as above except climb at 530 Kts. and cruise out at M.N. = .92

| | | |
|---|---|---|
| Maximum Range Mission with 15,298 Lb. Fuel ................ | N.M. | 406 |
| Maximum Range Mission with Full Internal Fuel ............. | N.M. | 605 |

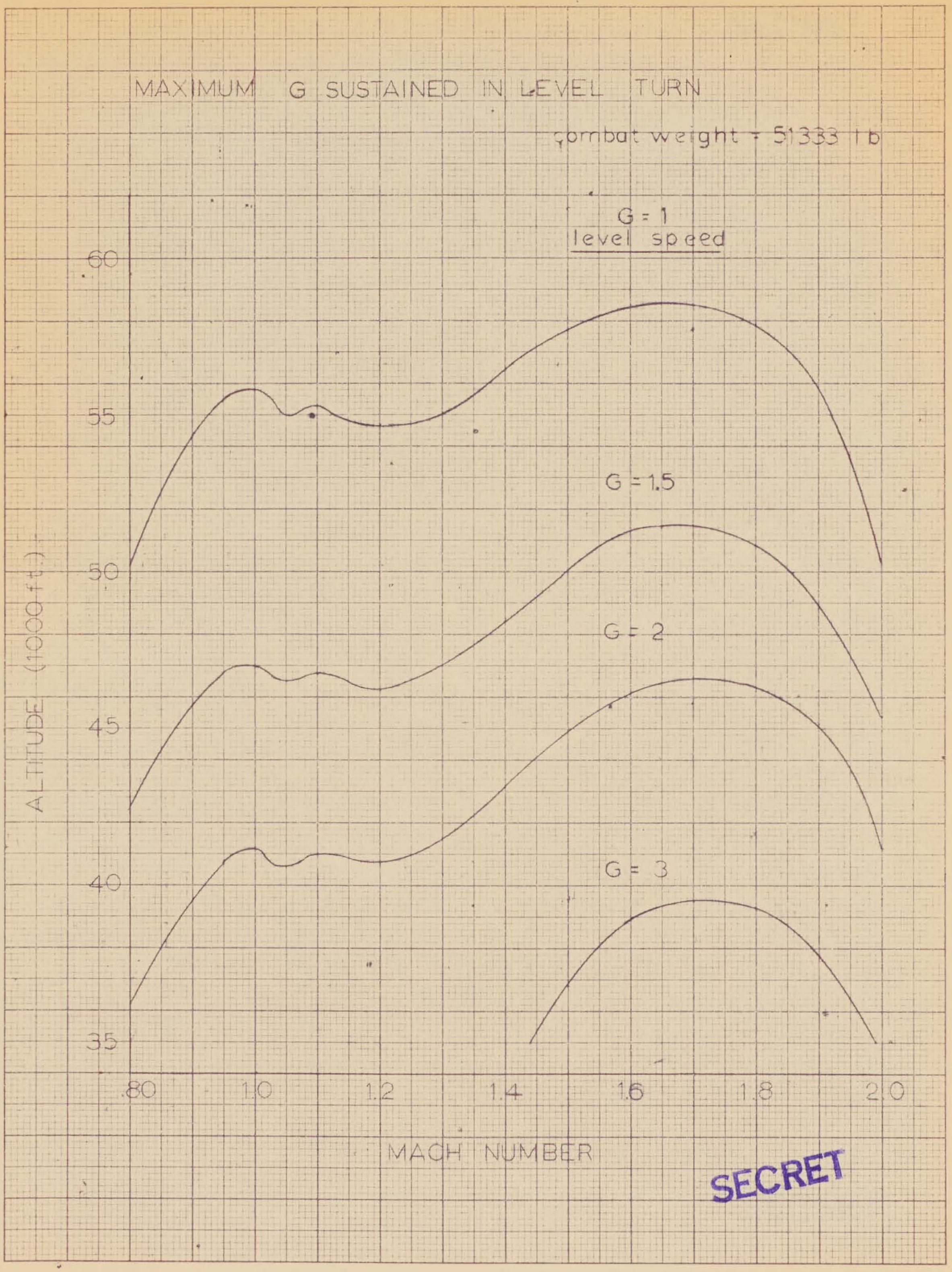
Combat Radius of Action at Sea Level, Cruise out at .6 M.N. and Combat at M.N. = .92 at Sea Level, Cruise Back at .92 M.N. at 40,000 Ft., 15 Min. Stack, 5 Min. Fuel Reserve on Landing

| | | |
|---|---|---|
| Sea Level Mission with 15,298 Lb. of Fuel ................. | N.M. | 325 |
| Sea Level Mission with Full Internal Fuel ................. | N.M. | 470 |

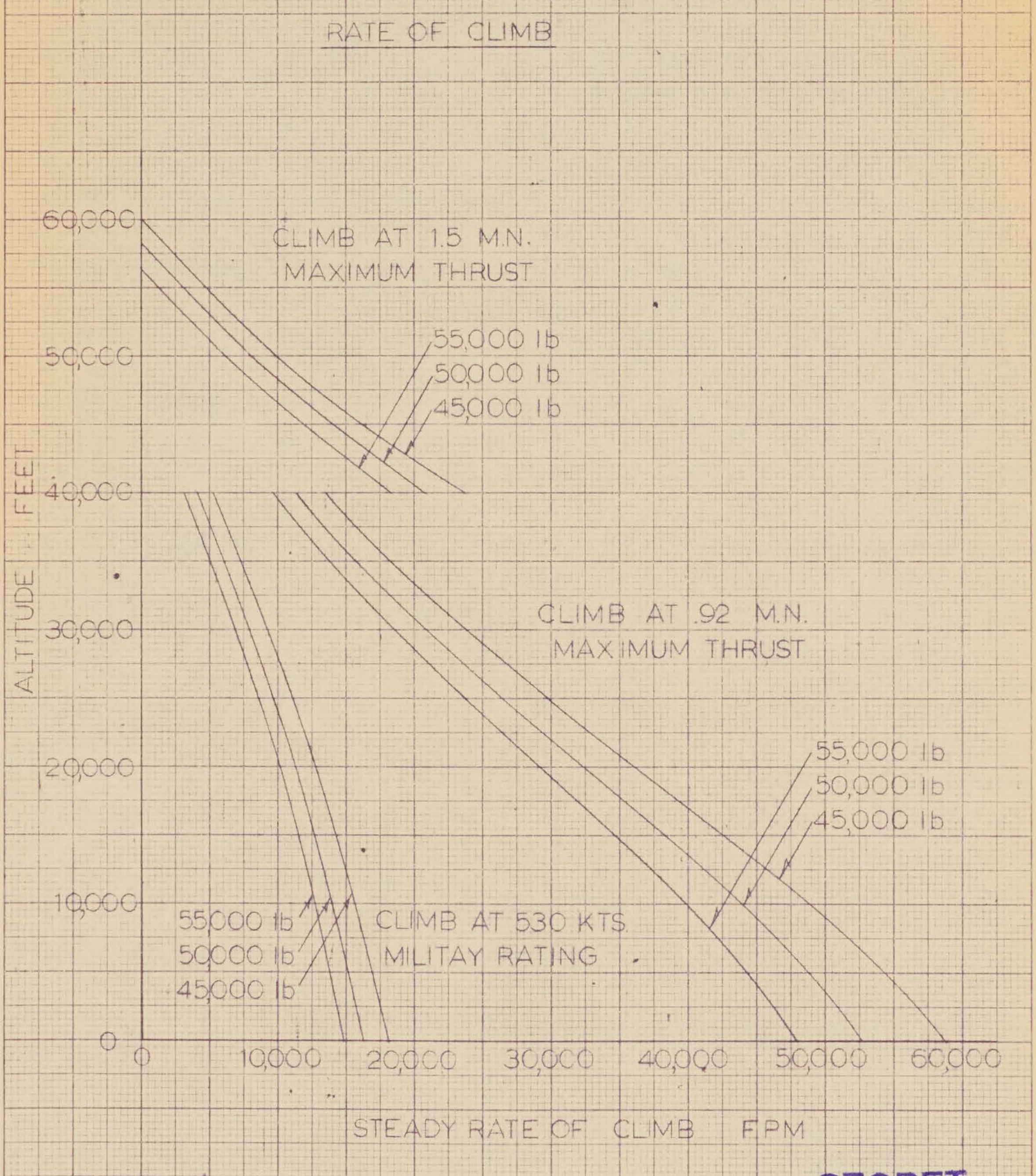
Ferry Range Mission at Economical Cruise Speed (M = .92 and Height, including 15 Mins. Stacking at 40,000 Ft., 5 Min. Fuel Reserve on Landing

| | | |
|---|---|---|
| Range with Full Internal Fuel and 500 Gal. - External Tank . | N.M. | 1,859 |
| Range with full internal fuel ............................ | N.M. | 1,609 |

SECRET

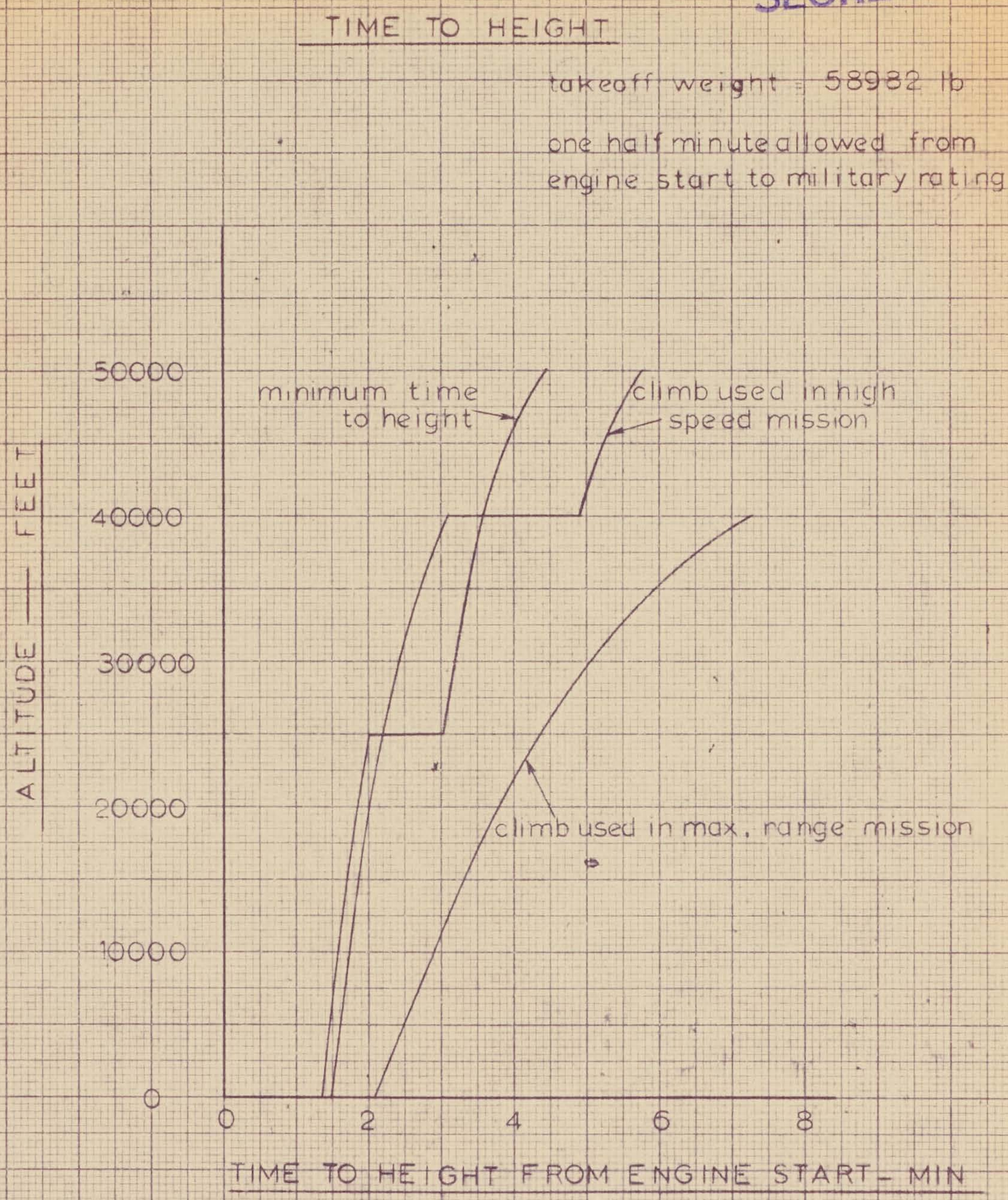

MAXIMUM G SUSTAINED IN LEVEL TURN
combat weight = 51333 lb
G = 1
level speed
G = 1.5
G = 2
G = 3
ALTITUDE (1000 ft)
60
55
50
45
40
35
.80
1.0
1.2
1.4
1.6
1.8
2.0
MACH NUMBER
10 X 10 TO THE 1/2 INCH 359-12
KEUFFEL & ESSER CO.

SECRET




SECRET

SECRET

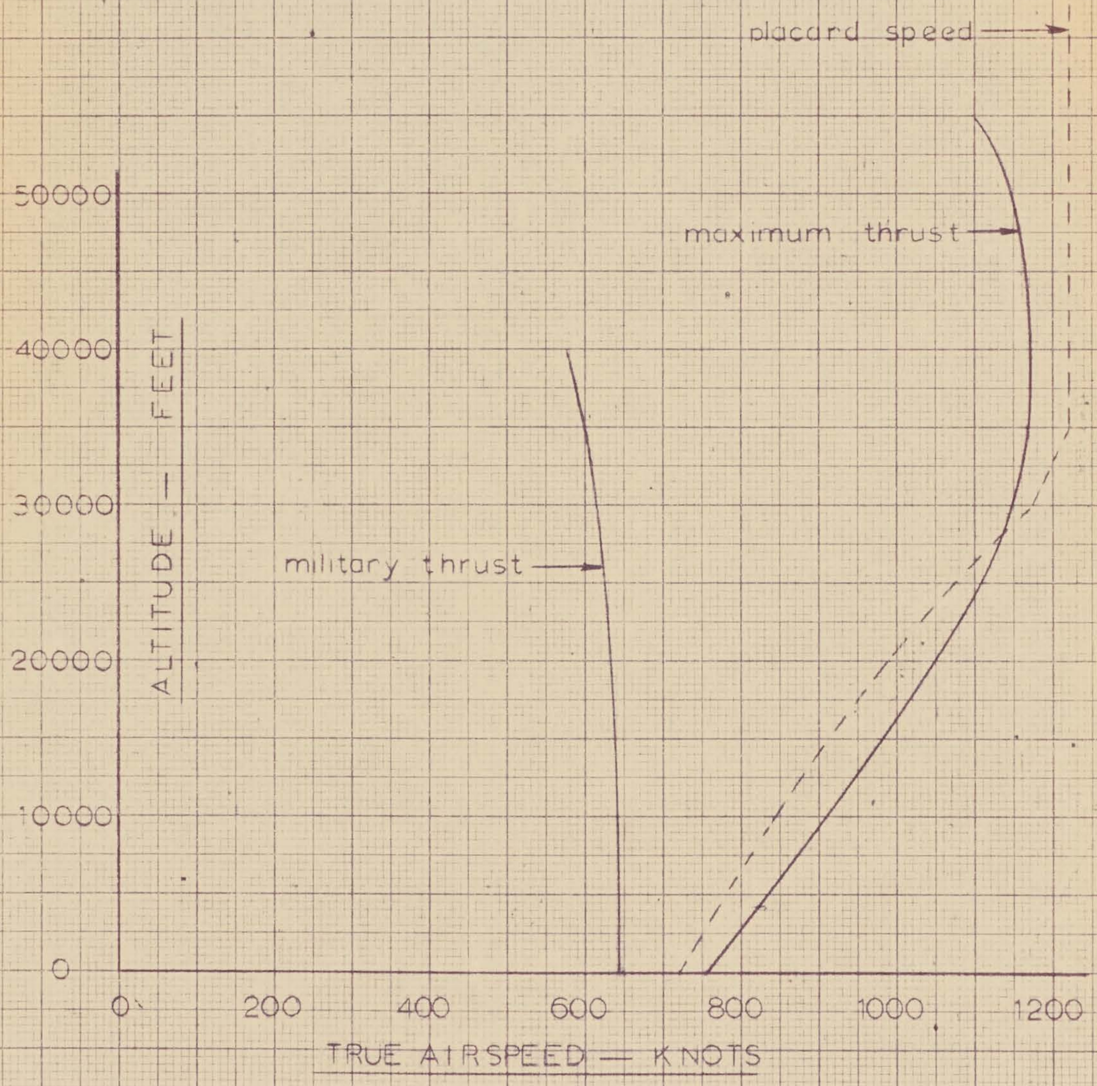



SECRET

SECRET

## LEVEL FLIGHT TRUE AIRSPEED

combat weight = 51333 lb



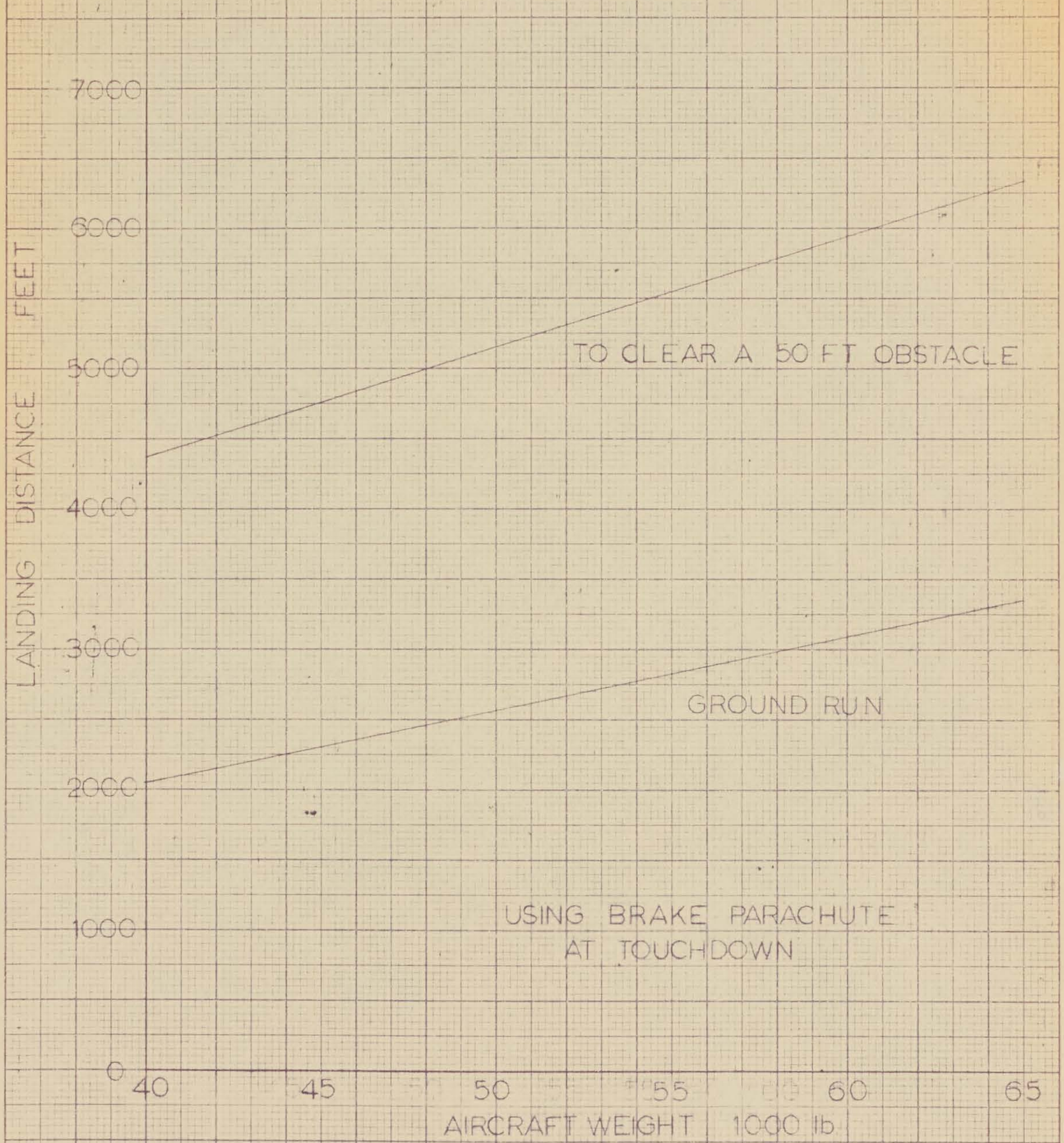

SECRET



DRAG

SECRET

LANDING DISTANCE AT SEA LEVEL STANDARD




SECRET



DRAG

## 1B. CF-105 PERFORMANCE WITH ORENDA PS 13 ENGINES

(C.G. = 29% M.A.C.)

The following CF-105 - PS 13 performance estimate is based on the wind tunnel configuration designated $B_2V_1W_1E_{10}N_5D_8$-4 over the subsonic portion, and configuration $W_9$, $N_{A5}$, $B_4$, $G_3$, $V_2$, $R_s$, over the supersonic range. The particular feature of the former configuration is the extended, notched, and cambered leading edge of the wing. The drag of this configuration is summarized, (Extract P/Aero Data/58), and is presented in section 2 of this report. The latter configuration differs chiefly by not having a cambered leading edge. This drag data is given in P/Aero Data/48 but has not been summarized for this report. This constitutes little change under supersonic cruise conditions, and only decreases the supersonic drag by about 4% at maximum 'g' due to less elevator angle for trim. Thus, the performance does reasonably represent that for the one configuration, $B_2V_1W_1E_{10}N_5D_8$ - 4.

The PS 13 engine data is in a more incomplete state. The engine data above the tropopause was taken from the Dec. '54 Memo, (Ref. Orenda P11-1-1) on the PS 13, with the exception of the cruise operation at .92 M.N. and 40,000 Ft., where insufficient data was available from the Memo, and we were forced to use the original PS 13 Brochure (EMS 8) April '54. The memo of Dec. '54 assumes a 6.5 Sq. Ft. intake, and pressure recovery curve from P/Power/23 APP/A/10. It also considers the effect of a 39" ejector, as well as a bypass which opens to 118 sq. Inches. For engine performance below the tropopause the original PS 13 Brochure was used. The above mentioned pressure recovery correction were applied to this data, but no account was taken of the bypass effect. It should be noted that revised thrust estimates now being prepared indicate an increase in maximum thrust at 1.5 M.N. of approximately 4%. This offsets the slightly optimistic supersonic drags used in this report for the performance of the PS 13 engines version.

## Performance Under N.A.C.A. Standard Atmospheric Conditions

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

## With Two PS 13 Engines

WEIGHT:

| | | |
|---|---|---|
| Take-Off Weight with 15,510 Lb. Fuel (78.2% Max.) | Lb. | 55,889 |
| Operational Weight Empty | Lb. | 40,379 |
| Combat Weight (1/2 Fuel) | Lb. | 48,130 |
| Landing Weight (With Reserve Fuel + Missiles) | Lb. | 42,200 |
| Wing Loading at Normal Take-Off Weight | Lb./Sq.Ft. | 44.5 |
| Power Loading at Normal Take-Off Weight | Lb./Lb. Thrust | 1.19 |

SPEED

True Air Speed in Level Flight
At Sea Level at Combat Weight

| | | |
|---|---|---|
| Maximum Thrust | Kts. | ★ 720 |
| Military Thrust | Kts. | 650 |

True Air Speed in Level Flight
At 50,000 Ft. at Combat Weight

| | | |
|---|---|---|
| Maximum Thrust | Kts. | 1,110 |

CEILING

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

| | | |
|---|---|---|
| Maximum Thrust at 1.5 M.N. | Ft. | 62,200 |

RATE OF CLIMB

Steady Rate of Climb at Sea Level, Combat Weight

| | | |
|---|---|---|
| Maximum Thrust at M.N. = .92 | F.P.M. | 50,000 |
| Military Thrust at 530 Kts. | F.P.M. | 25,200 |

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

| | | |
|---|---|---|
| Maximum Thrust at M.N. = 1.5 | F.P.M. | 11,500 |

TIME TO HEIGHT

Time to 50,000 Ft. M.N. = 1.5 from Engine Start at Take-Off
Weight = 55,889 Lb.

| | | |
|---|---|---|
| Maximum Thrust | Mins. | 4.1 |

MANOEUVRABILITY

Combat Load Factor at Combat Weight
Maximum Thrust at M.N. = 1.50 at 50,000 Ft. 1.84

★ Placard Speed = 720 Kts.

SECRET

TAKE-OFF DISTANCE

Take-Off Distance over 50 Ft. Obstacle at Sea Level ............
Take-Off Weight = 55,889 Lb.

| | | |
|---|---|---|
| Maximum Thrust .......................................... | Ft. | 2,500 |
| Military Thrust ......................................... | Ft. | 3,800 |
| Maximum Thrust Hot Day .................................. | Ft. | 3,300 |

LANDING DISTANCE

Landing Distance over 50 Ft. Obstacle at Sea Level at Combat Weight Ft. 5,000

STALLING SPEED

True Stalling Speed in Landing Configuration at Combat Weight at Sea Level .............................................. Kts. 105

RANGE

Combat Radius of Action at 50,000 Ft., Climb at M.N. = .92, 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., 5Min. Fuel Reserve on Landing

| | | |
|---|---|---|
| High Speed Mission with 15,510 Lb. Fuel .................... | N.M. | 200 |
| High Speed Mission with Full Internal Fuel ................. | N.M. | 318 |

Combat Radius of Action at 50,000 Ft., Mission as above except Cruise Out at M.N. = .92

| | | |
|---|---|---|
| Maximum Range Mission with 15,510 Lb. Fuel ................. | N.M. | 315 |
| Maximum Range Mission with Full Internal Fuel .............. | N.M. | 491 |

Combat Radius of Action at Sea Level, Cruise Out at .6 M.N. and Combat at M.N. = .92 at Sea Level, Cruise Back at .92 M.N. at 40,000 Ft., 15 Min. Stack, 5 Min. Fuel Reserve on Landing

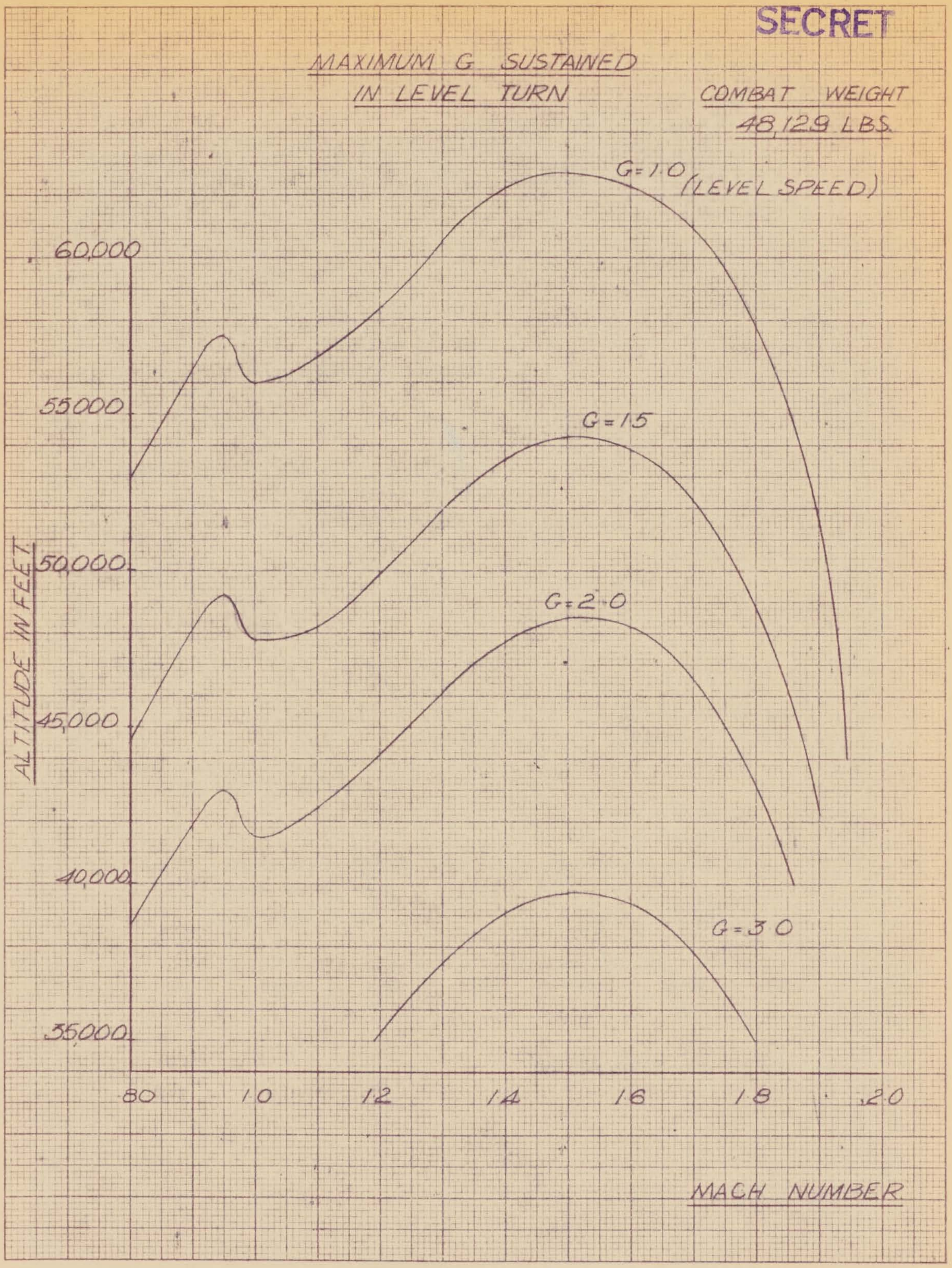
| | | |
|---|---|---|
| Sea Level Mission with 15,510 Lb. of Fuel .................. | N.M. | 217 |
| Sea Level Mission with Full Internal Fuel .................. | N.M. | 318 |

Ferry Range Mission at Economical Cruise Speed (M = .92 and Height, including 15 Mins. Stacking at 40,000 Ft., 5 Min. Fuel Reserve on Landing

| | | |
|---|---|---|
| Range with Full Internal Fuel and 500 Gal. - External Tank . | N.M. | 1,675 |

SECRET

SECRET





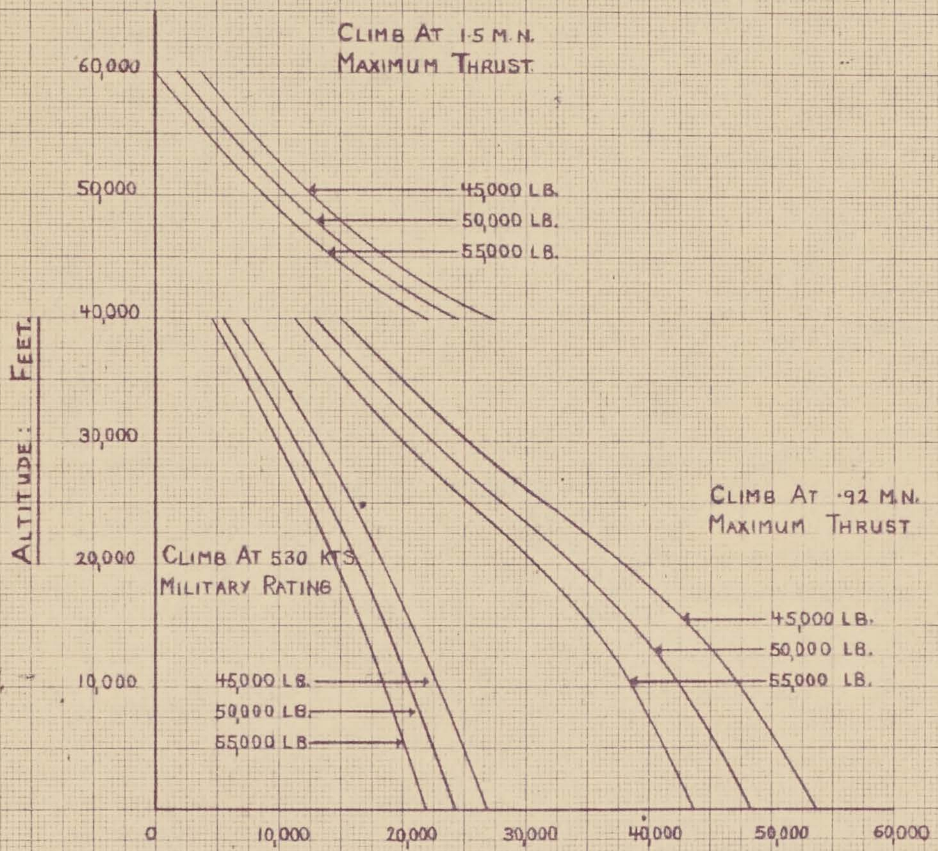

SECRET JULY 55 T. GRAYSON.

PS 13 ENGINES

PIPERF/102

SECRET

# RATE OF CLIMB




SECRET

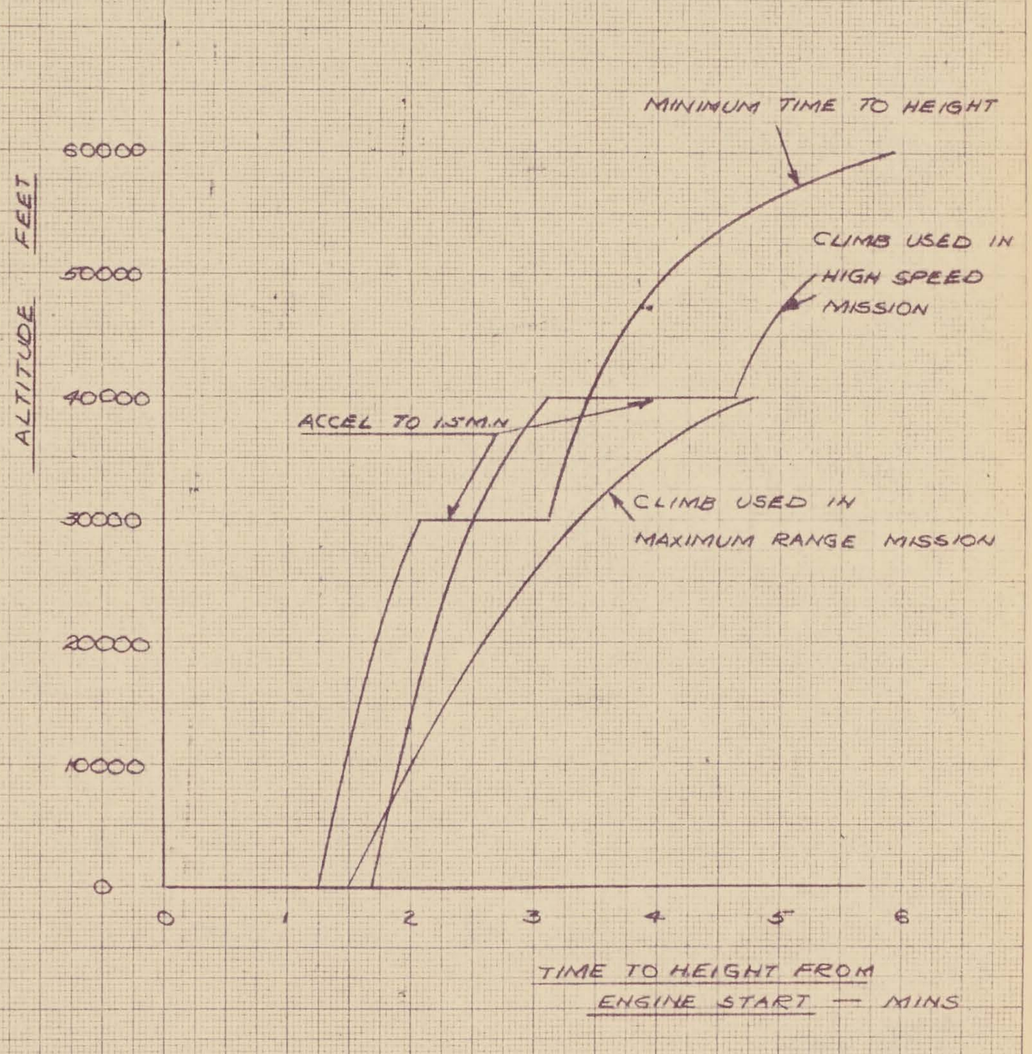
DRAG

SECRET

TIME TO HEIGHT

TAKEOFF WEIGHT = 55889 LB

NOTE: ONE HALF MINUTE ALLOWED FROM ENGINE START TO MILITARY RATING




SECRET

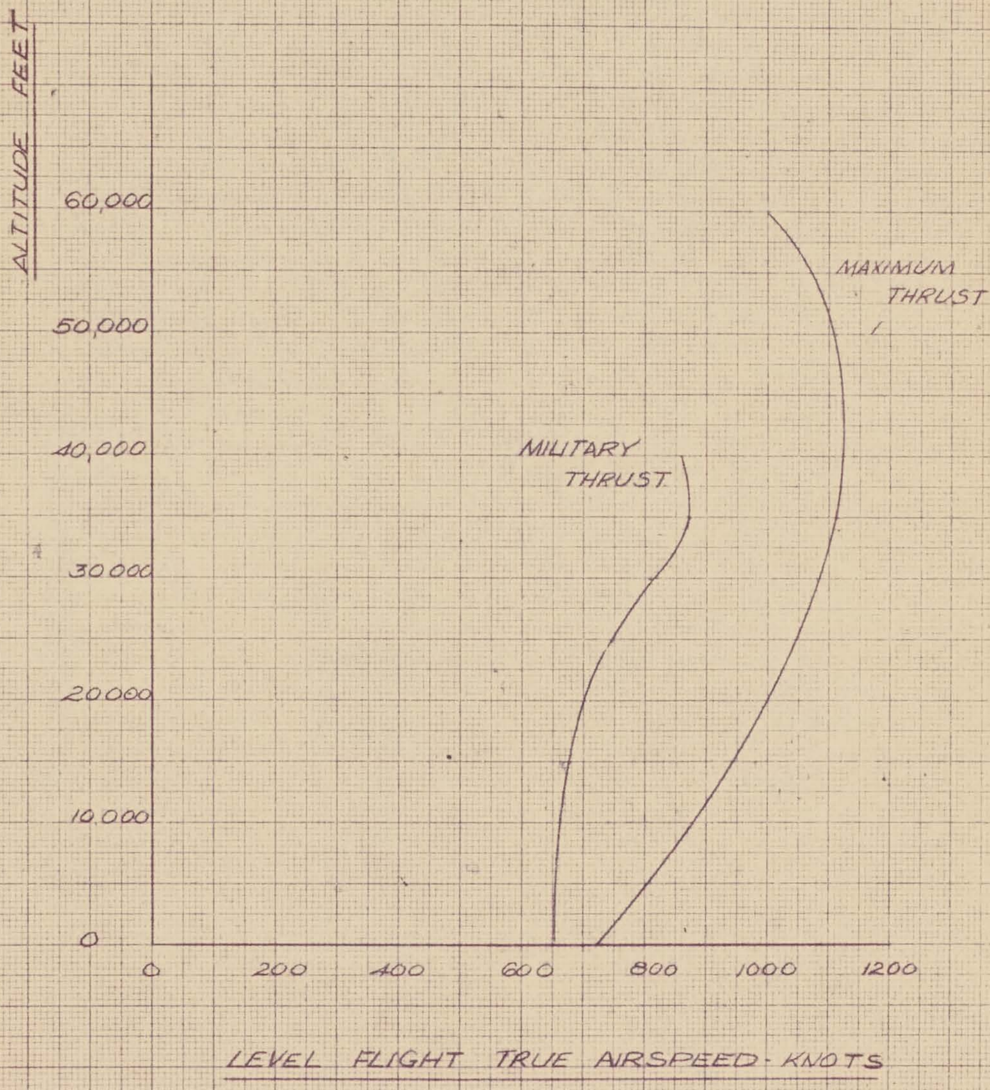


DRAG

SECRET

# LEVEL FLIGHT TRUE AIRSPEED

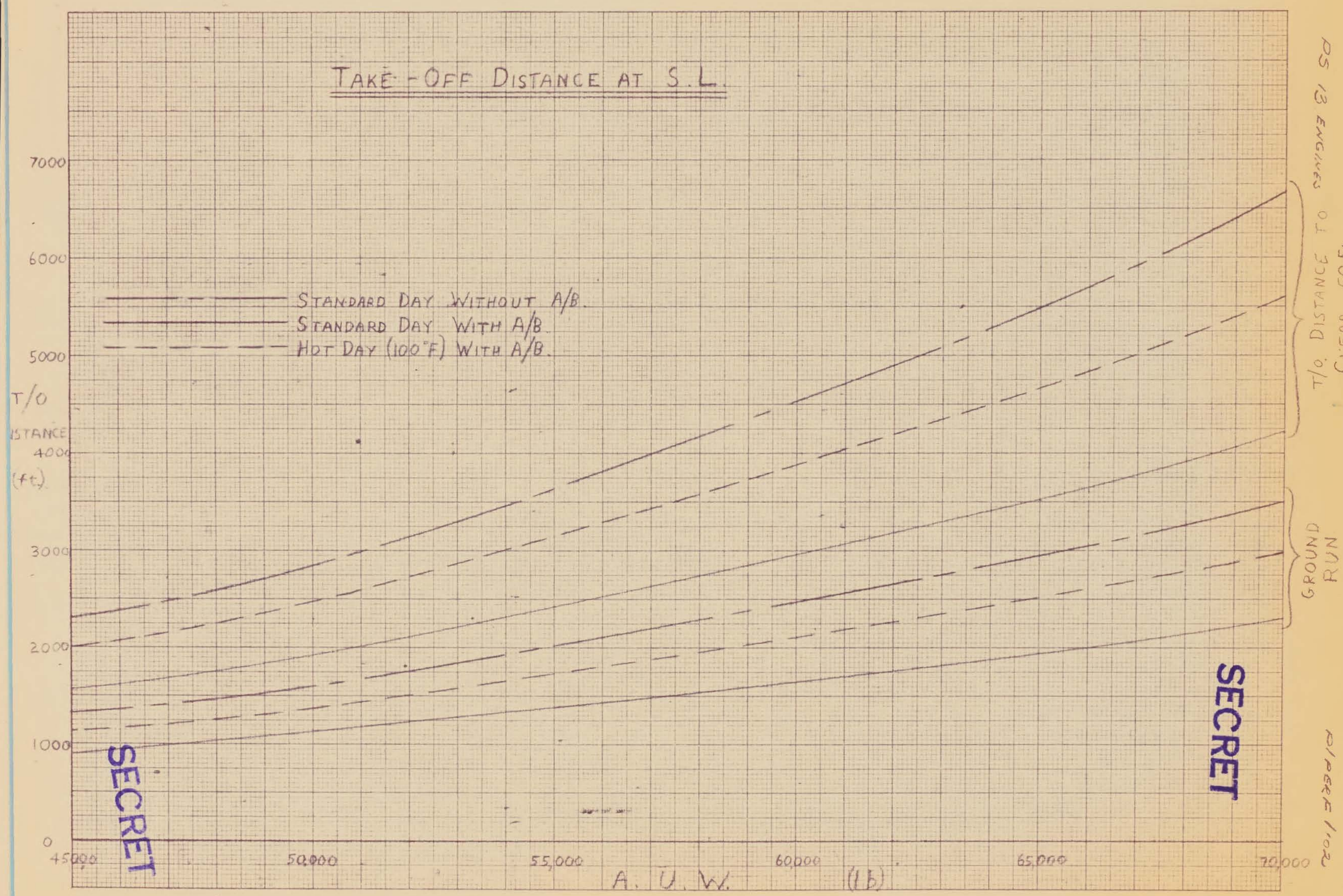
COMBAT WEIGHT = 48129LB.




SECRET

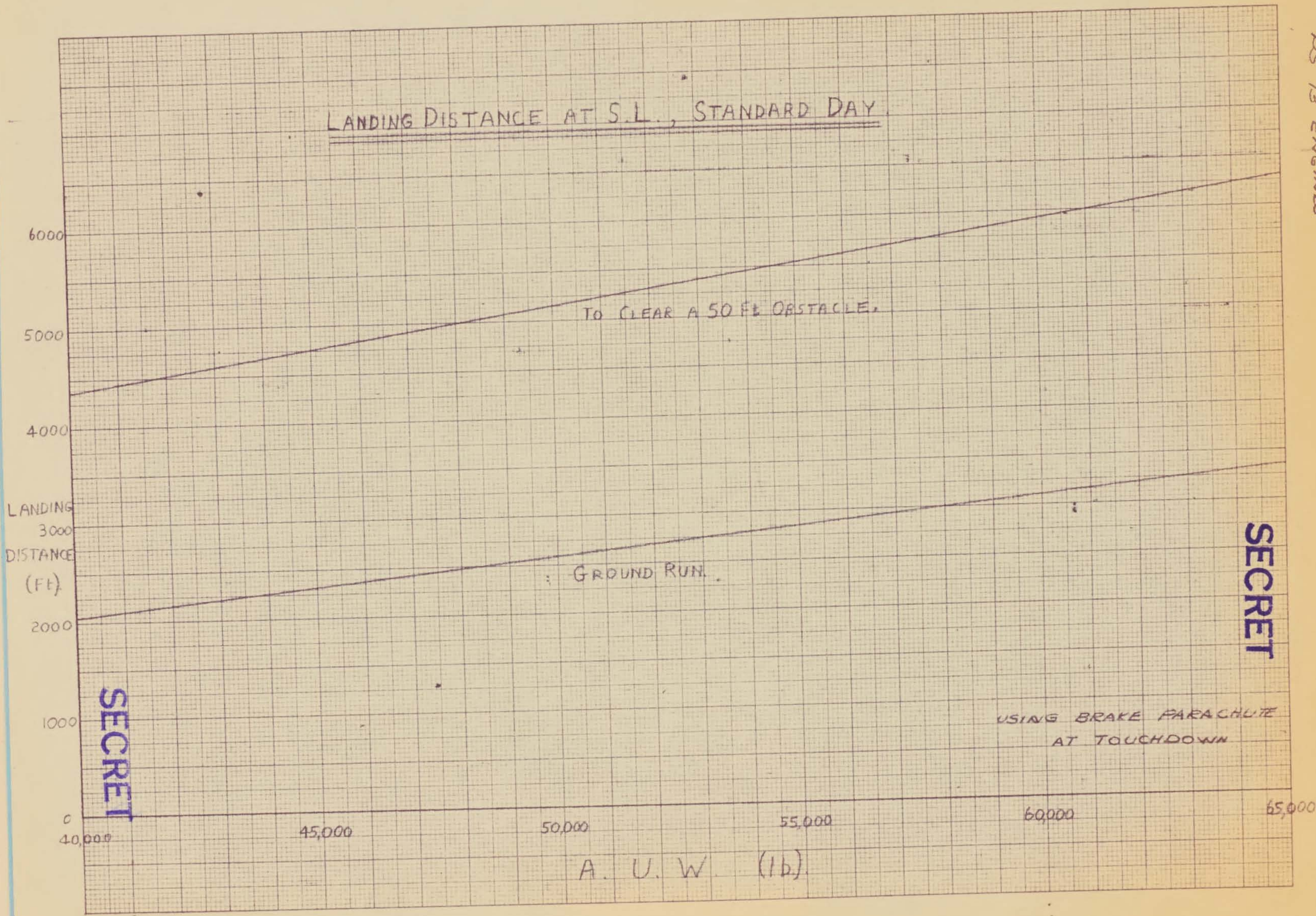


DRAG


TAKE-OFF DISTANCE AT S.L.
STANDARD DAY WITHOUT A/B.
STANDARD DAY WITH A/B.
HOT DAY (100°F) WITH A/B.
T/O DISTANCE (ft)
7000
6000
5000
4000
3000
2000
1000
0
45,000
50,000
55,000
60,000
65,000
70,000
A. U. W. (lb)
T/O DISTANCE TO CLEAR 50 Ft.
GROUND RUN
PS 13 ENGINES
SECRET
SECRET
P/PERF 1/02

DRAG

RS 13 ENGINES




SECRET

SECRET

PIPERF/102



DRAG

DRAG

Extract P/Aero Data/58

September 30, 1955.

SECRET

CF-105 (Configuration $B_2V_1W_1E_{10}N_5D_{8-4}$) DRAG NOTE

This extract contains the latest CF-105 drag data used for performance estimations. The particular feature of this configuration is the extended, notched, and cambered leading edge of the wing.

The supersonic $C_{D_{MIN}\,\delta C_{D_{MIN}}}$ has been anchored by the selection of $C_{D_{MIN}\,\delta C_{D_{MIN}}} = .02$ at 1.5 M.N. This selection has been based on a compromise between CF-105 C.A.L. Wind Tunnel Tests, the first CF-105 Free Flight Model and estimates. Similarly, the subsonic value of $C_{D_{MIN}\,\delta C_{D_{MIN}}}$ has been selected.

The drag due to lift, including elevator drag to trim, has been obtained from C.A.L. Wind Tunnel Project No. W.A. 844-DD3 results. The model was .04 scale, the Mach Number range was from .5M to 1.23M with a Reynold's Number range from 1.6 to 2.5 x $10^6$. No allowance has been made for scale effect. Beyond 1.23M, the drag coefficients have been extrapolated where possible by data from N.A.C.A. reports. The total drag is then determined from -

$$D/P = 126500M^2 \left[ C_{D_{MIN}\,\delta C_{D_{MIN}}} + \frac{\left( \frac{1}{126500M^2} \times \frac{W}{P} - C_{L_{C_{DMIN}}\,\delta_{TRIM}} \right)^2}{\pi \, AR \, e_{\delta_{TRIM}}} \right.$$

$$\left. + \frac{\Delta C_{D_{MIN}}}{\left[ \delta_{TRIM} - \delta C_{D_{MIN}} \right]^2} \times \left[ \delta_{TRIM} - \delta C_{D_{MIN}} \right]^2 \right]$$

and $$\delta_{TRIM} = \frac{C_{L_A}(h - a.c.) + C_{M_0}}{-K \, C_{M_\delta}}$$

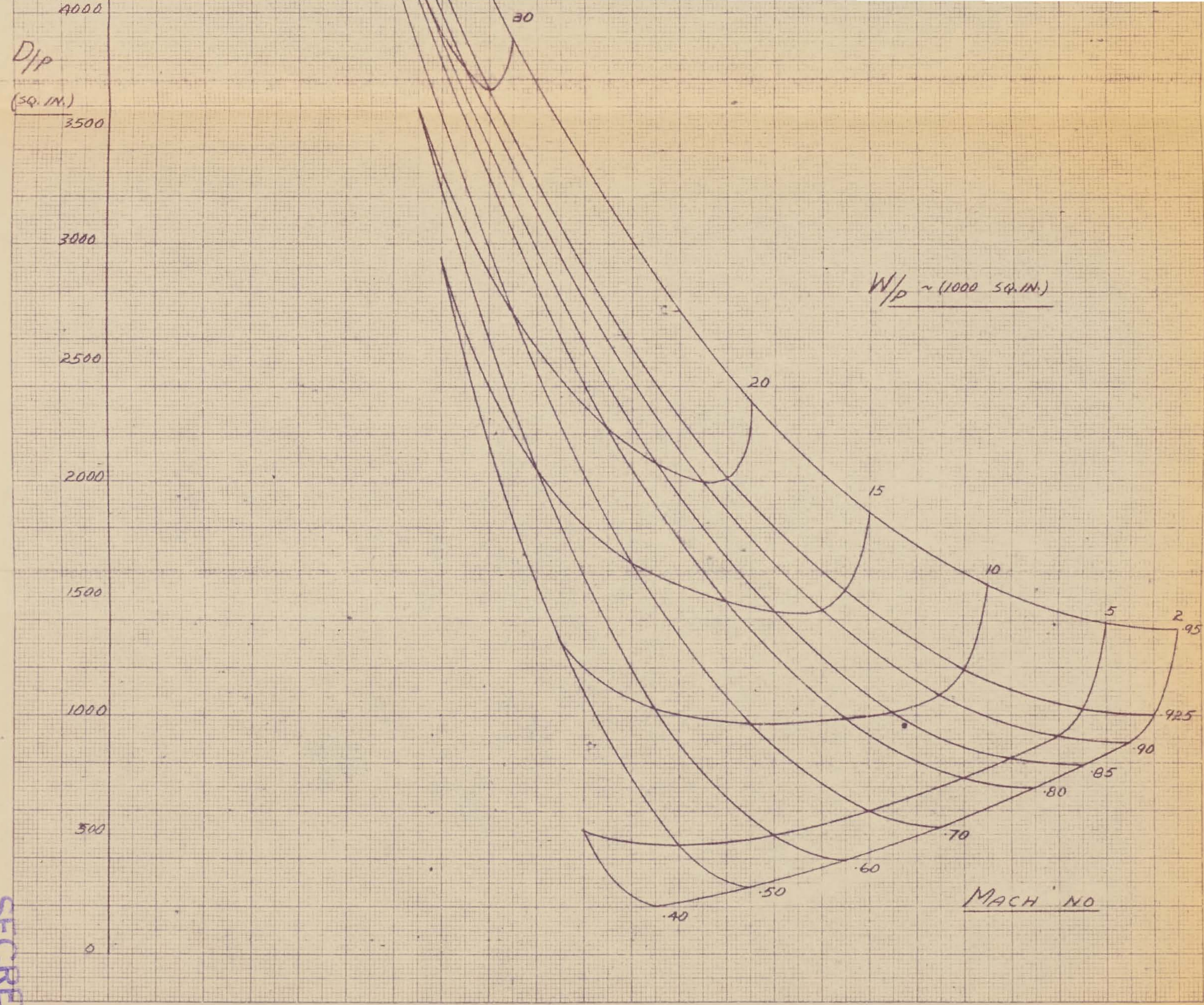
SECRET

| | | |
|---|---|---|
| Where | $D$ | - Total Drag - Lb. |
| | $P$ | - Ambient Pressure - Lb./Sq.In. |
| | $M$ | - Mach Number |
| | $W$ | - Aircraft Effective Weight (Normal load factor x aircraft weight) - Lb. |
| | $AR$ | - Aspect Ratio (1.995) |
| | a.c. | - Aerodynamic centre % M.A.C. |
| | $h$ | - Centre of Gravity % M.A.C. |
| | $\delta_{TRIM}$ | - Elevator Angle for Trim of Aircraft at effective weight - degrees |
| | $\delta_{C_{D_{MIN}}}$ | - Elevator Angle for Minimum drag - degrees |
| | $K$ | - Non-linearity factor for $C_{M_\delta}$ |
| | $C_{D_{MIN}\delta_{C_{D_{MIN}}}}$ | - Minimum Drag Coefficient |
| | $C_{L_{C_{D_{MIN}}\delta_{TRIM}}}$ | - Lift Coefficient at $C_{D_{MIN}}$ corresponding to $\delta_{TRIM}$ |
| | $C_{L_A}$ | - Aircraft Lift Coefficient |
| | $e_{\delta_{TRIM}}$ | - Aerodynamic Drag Efficiency Factor at $\delta_{TRIM}$ |
| | $C_{M_o}$ | - Pitching Moment Coefficient at Zero Lift |
| | $C_{M_\delta}$ | - Elevator Pitching Effectiveness at constant $C_L$ |

From the drag curve, thus determined, the coefficients can be approximately reduced to the more conventional form.

$$C_D \approx C_{D_{intercept}} + \frac{C_L^2}{AR\, e_{\delta_{TRIM}(C_{D_{intercept}})}}$$

SECRET



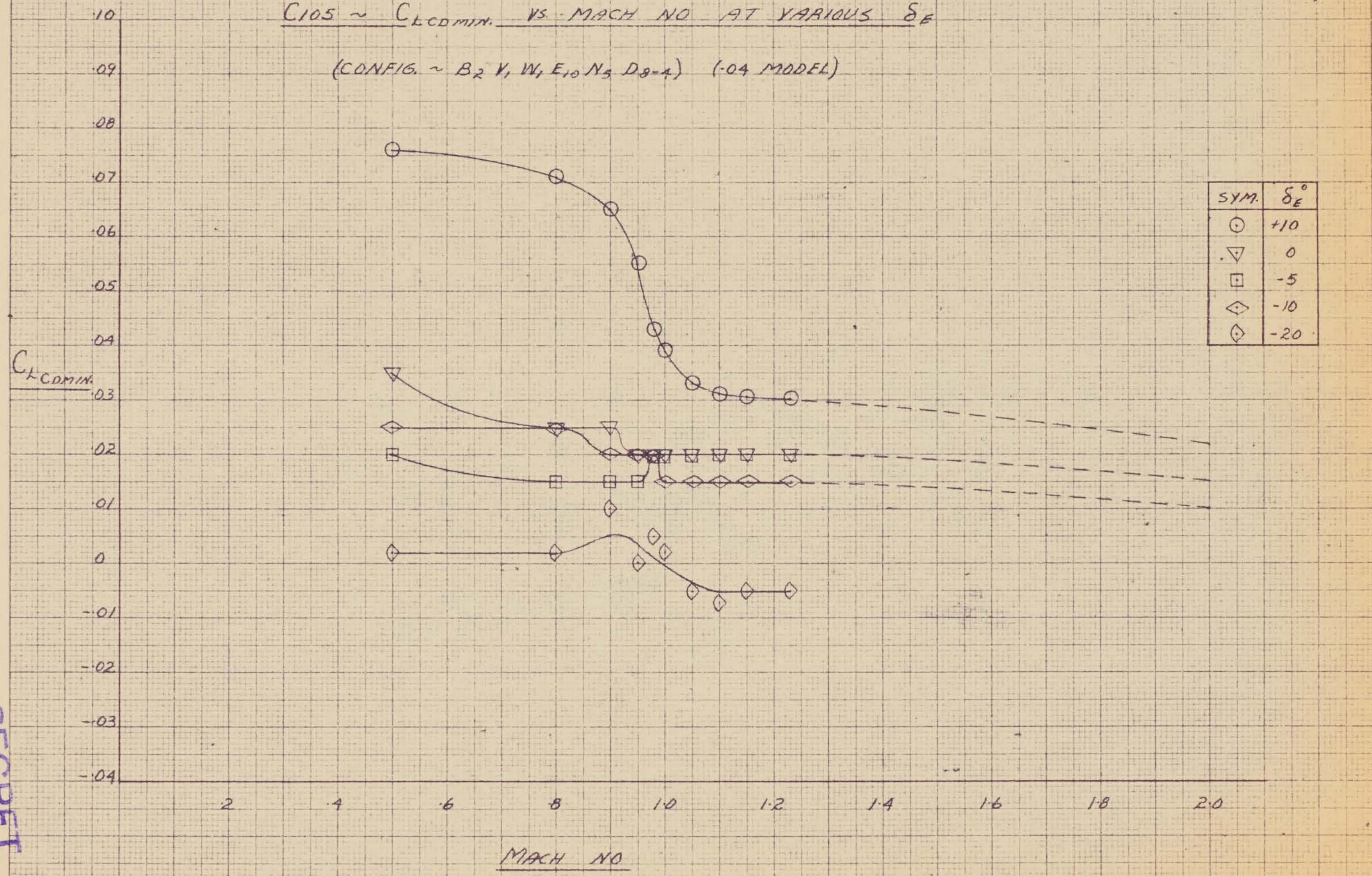

SECRET

P/AERO DATA/58
R. SKULSKY
SEPT./55

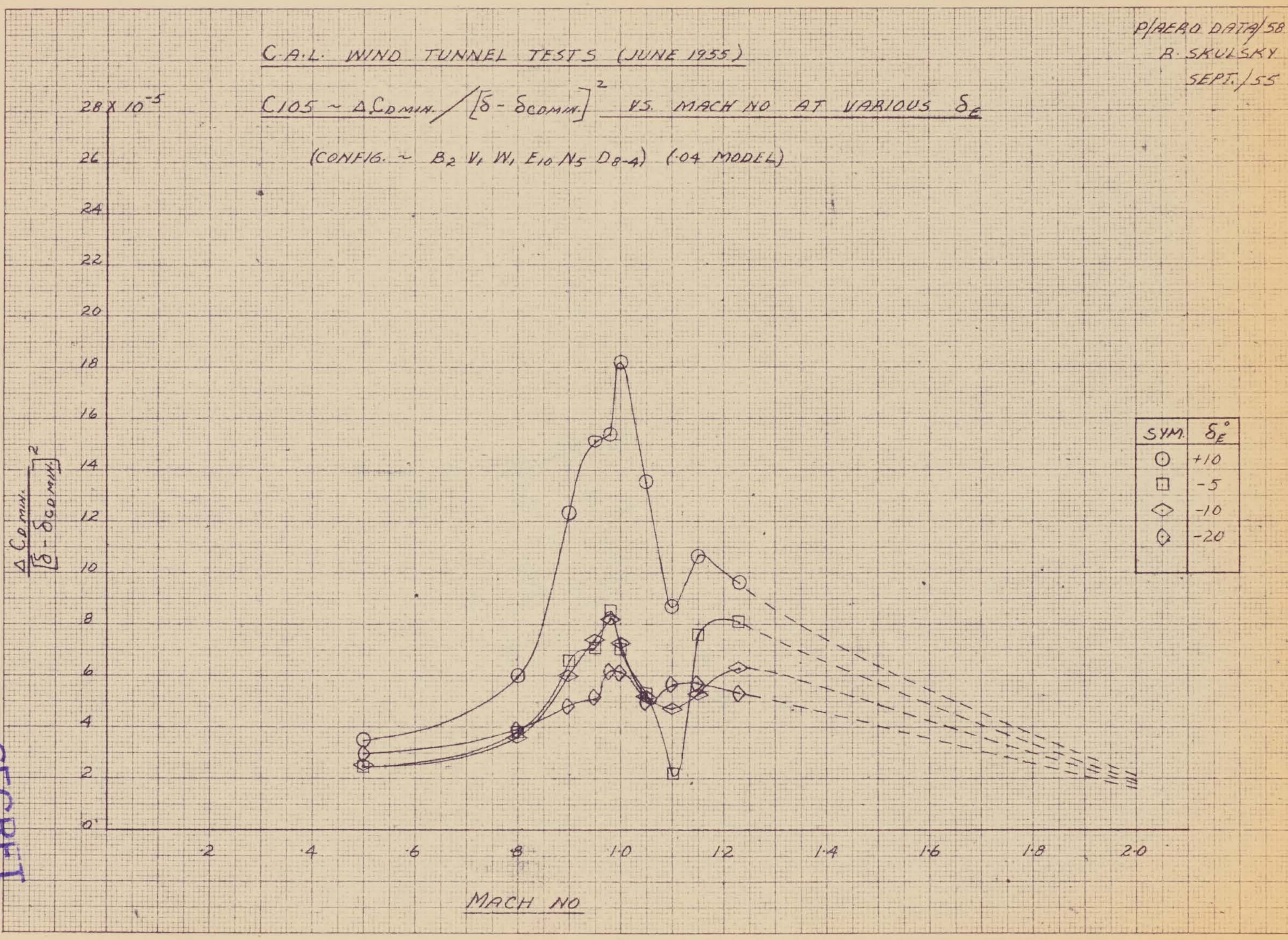
C.A.L. WIND TUNNEL TESTS (JUNE 1955)

$C105 \sim C_{L_{CD\,MIN.}}$ VS. MACH NO AT VARIOUS $\delta_E$

(CONFIG. ~ $B_2 V_1 W_1 E_{10} N_5 D_{8-4}$) (.04 MODEL)



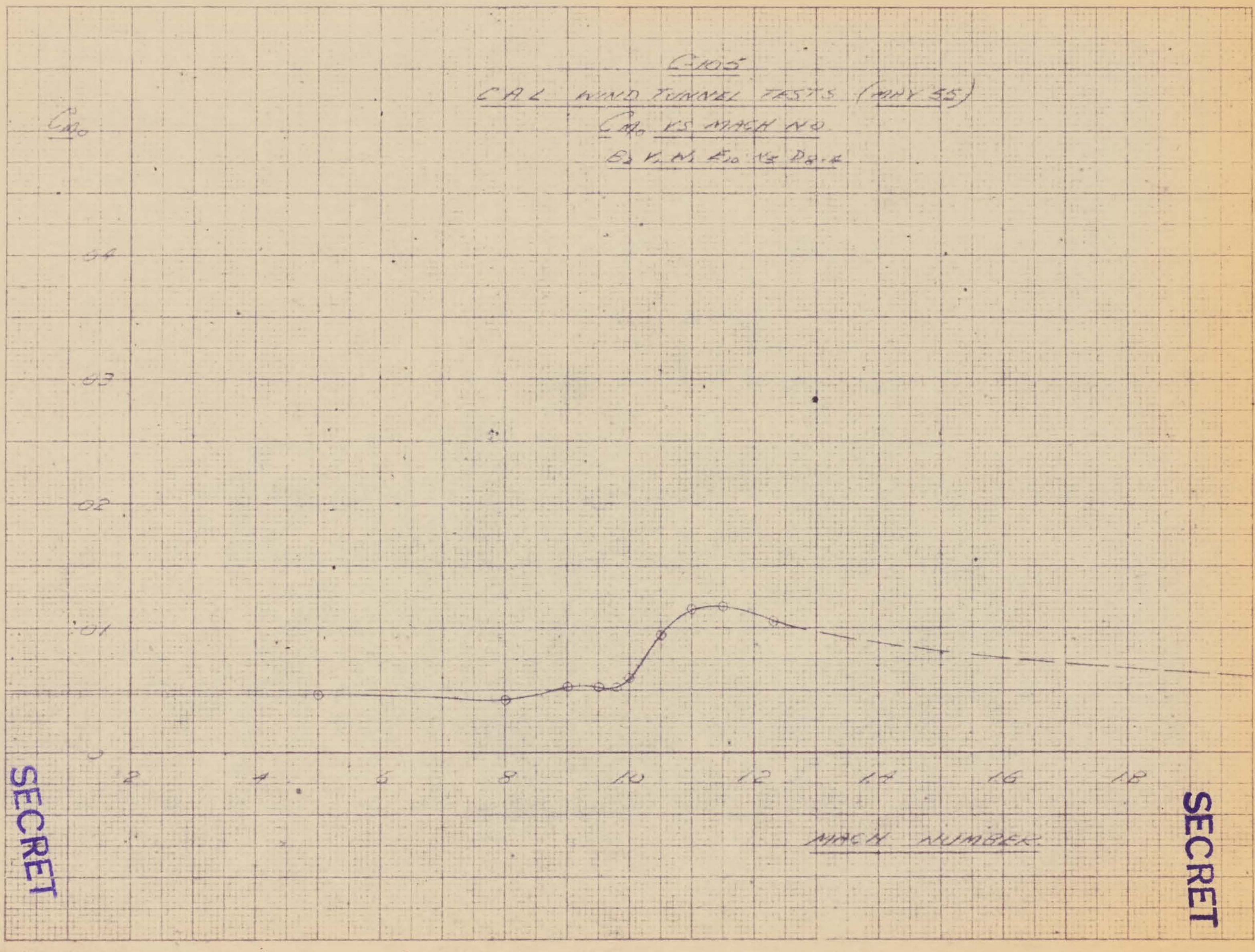
P/AERO DATA/58
R. SKULSKY
SEPT./55






ENGINE

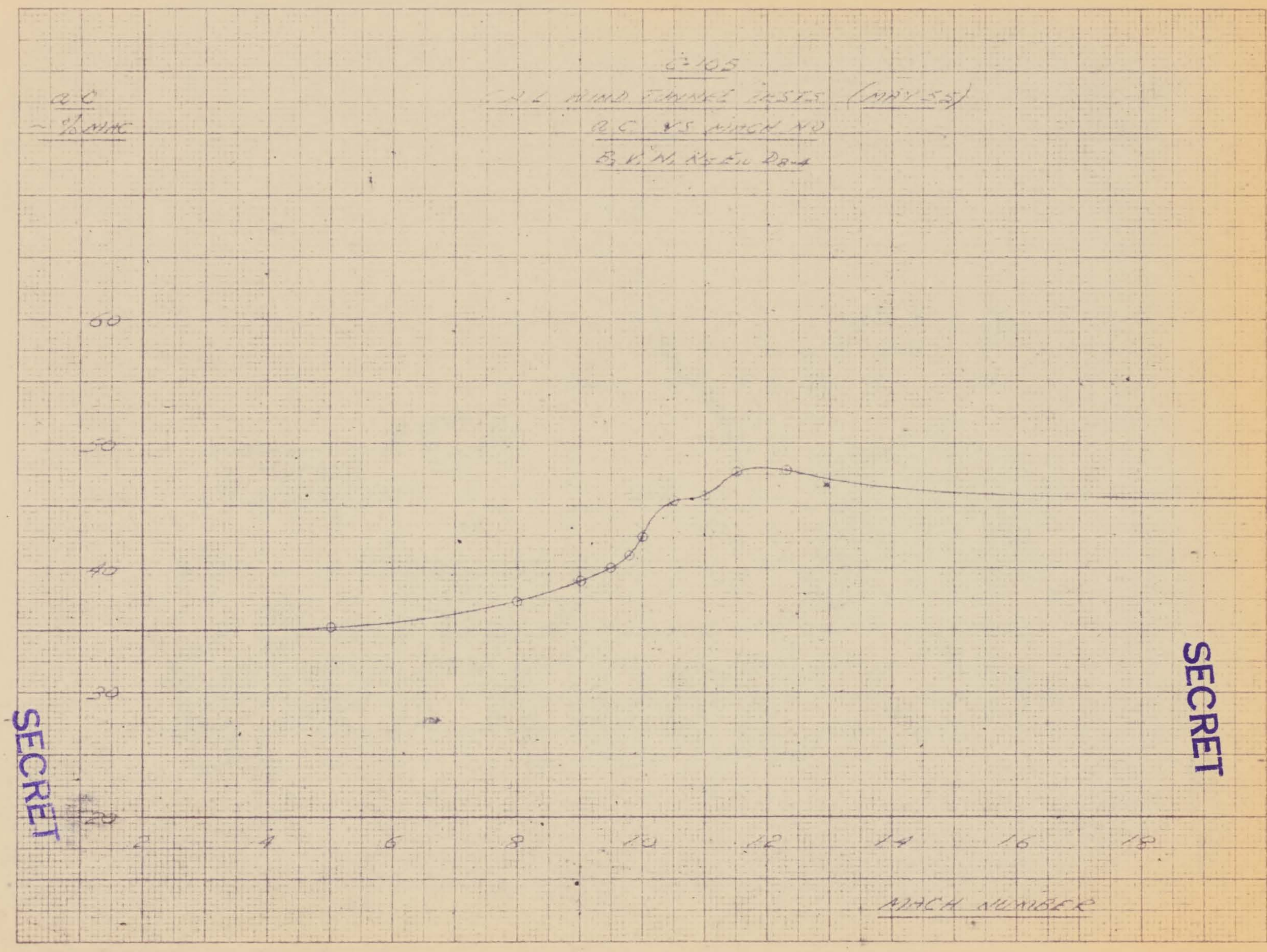
SECRET




SECRET

2.2 MAY 55 P/NT/180 CLARK

ENGINE

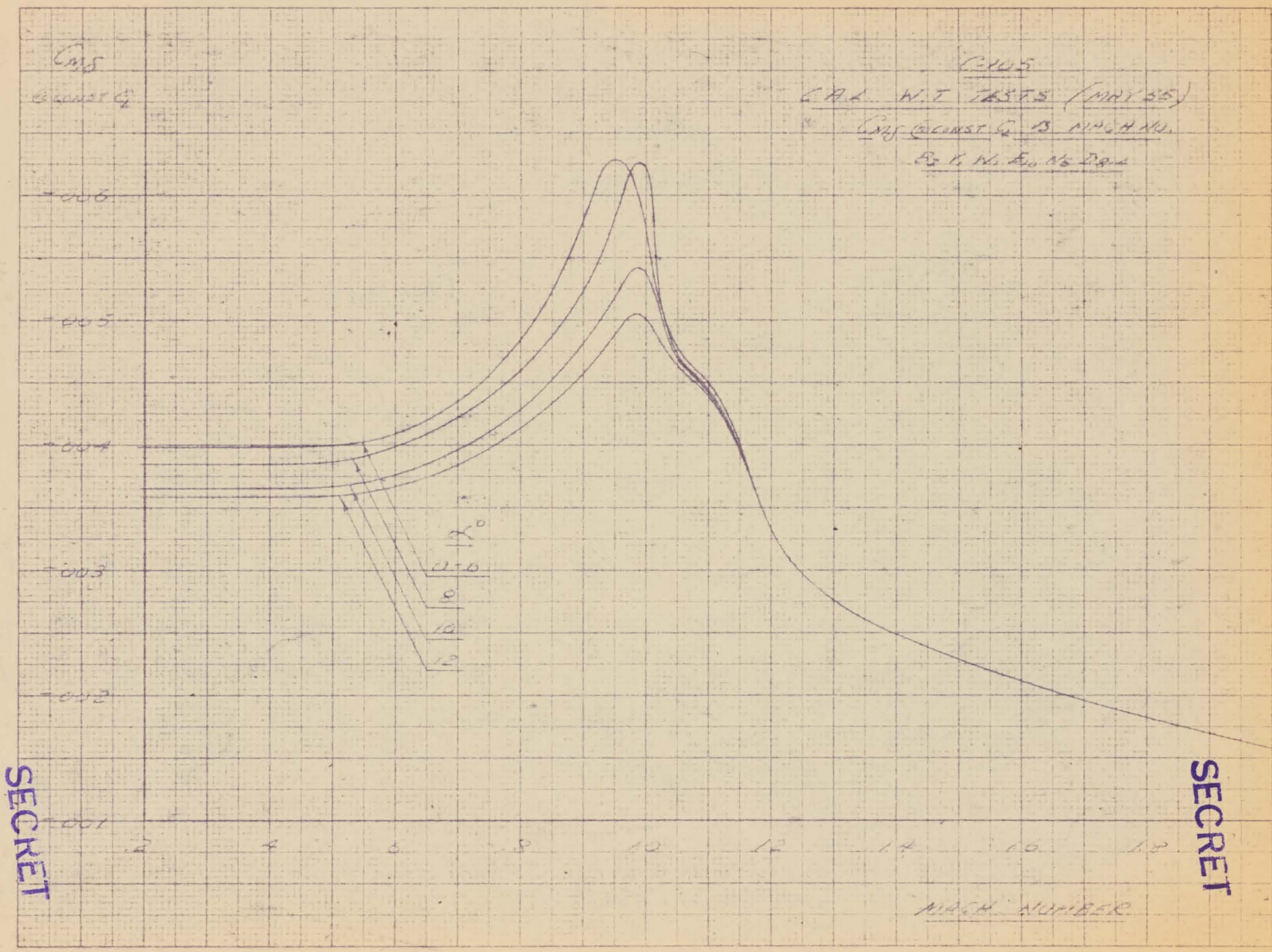



SECRET

SECRET

ENGINE

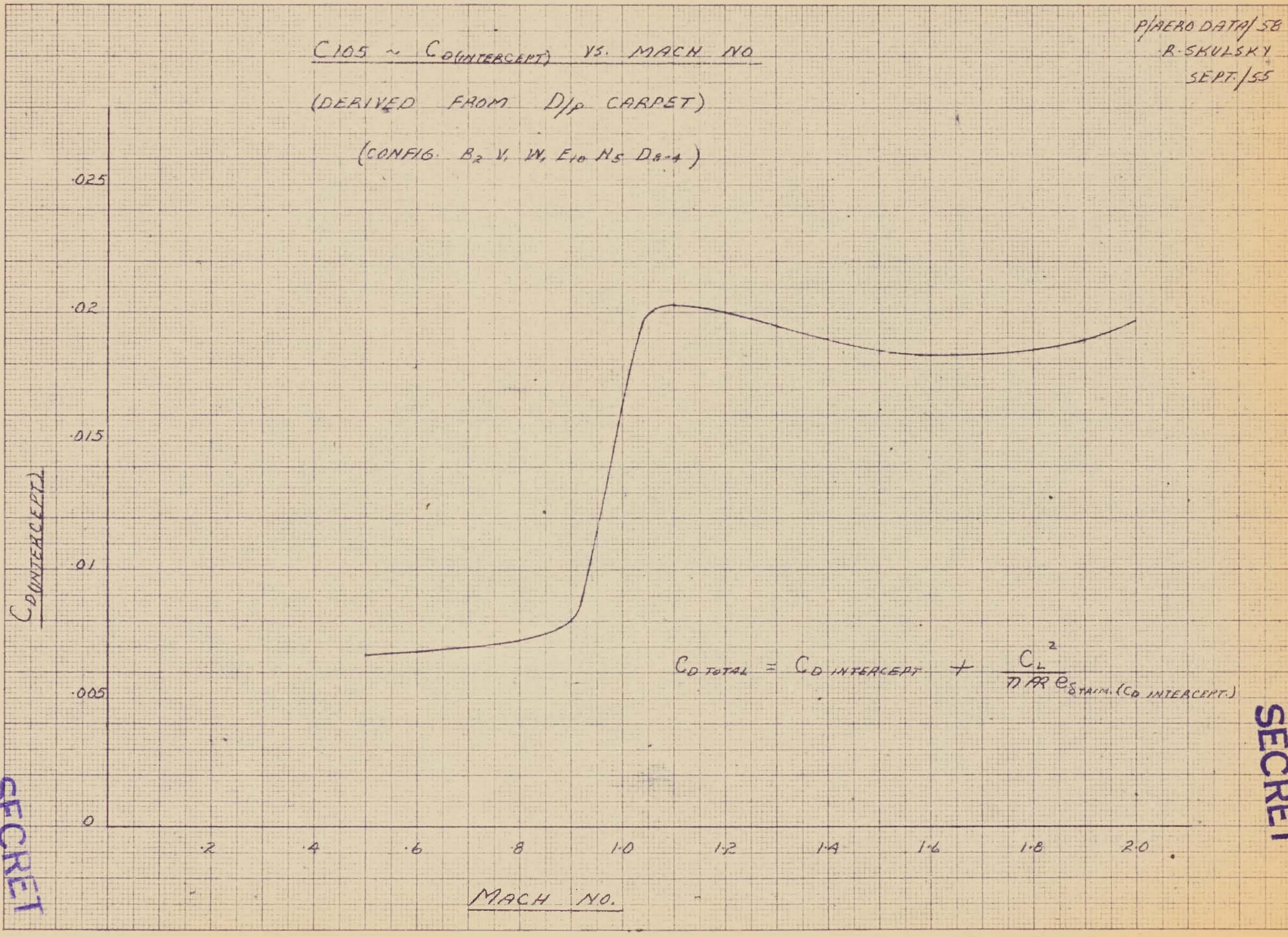
K+E 10 X 10 TO THE 1/2 INCH 359-12
KEUFFEL & ESSER CO.





SECRET
ENGINE

SECRET




SECRET

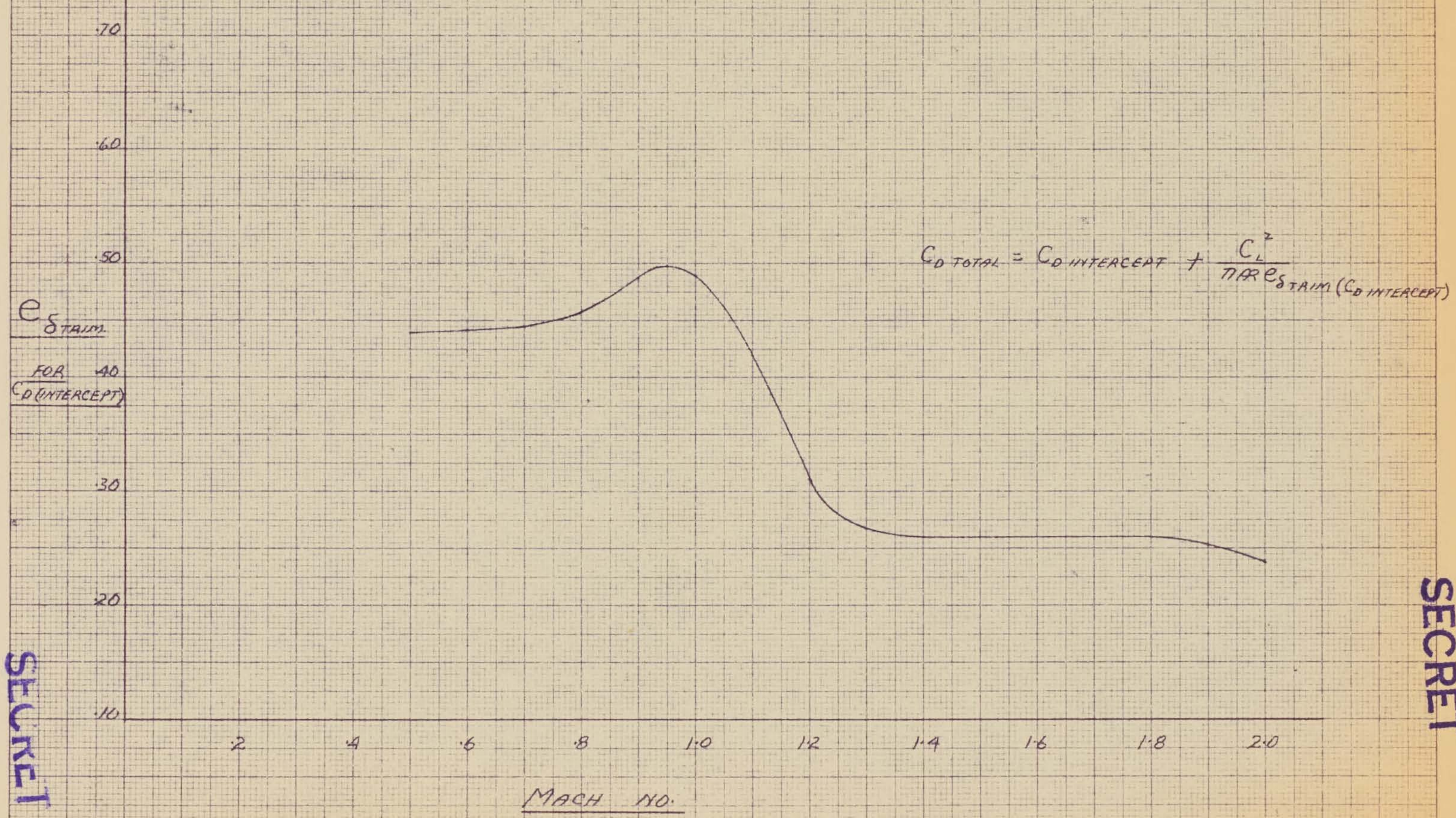
ENGINE

P/AERO DATA/58
R. SKULSKY
SEPT./55

# C105 ~ $e_{\delta_{TRIM}}$ VS. MACH NO FOR $C_{D(INTERCEPT)}$

(DERIVED FROM D/p CARPET)

(CONFIG. $B_2 V_1 W_1 E_{10} N_5 D_{8-4}$)






ENGINE

AVRO AIRCRAFT LIMITED
TECHNICAL DEPARTMENT (Aircraft)

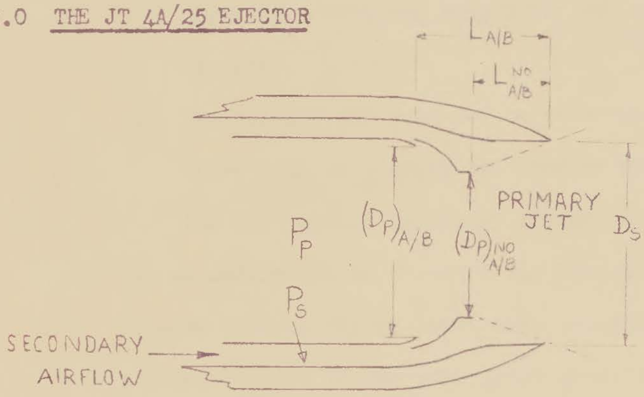
| AIRCRAFT | | REPORT NO. P/POWER/51 SHEET NO. 1 | |
|---|---|---|---|
| CF-105 | METHOD OF ESTIMATING EJECTOR PERFORMANCE | PREPARED BY J.R. Monk | DATE Oct. '55 |
| | | CHECKED BY | DATE |

SECRET

## 1.0 INTRODUCTION

The following report gives a method of using the experimental results obtained by the N.A.C.A. ejector tests to estimate the thrust of the Pratt & Whitney JT 4A/25 fitted with an ejector. It has been found that under certain conditions the ejector gives a significant increase of thrust, particularly at high speeds. The ejector also serves to give a cooling airflow over the engine.

## 2.0 THE JT 4A/25 EJECTOR




The JT 4A/25 has a two position primary nozzle. The nozzle diameter $D_p$ is 34.5 in. for afterburning and 28 in. for no afterburning. On the CF-105, it is proposed to instal a secondary shroud of diameter $D_s$ of 39 in. which projects beyond the end of the jet pipe by a distance L. This secondary shroud together with the jet pipe nozzle make up the ejector. The secondary airflow cushions the expansion of the primary jet within the shroud and under certain conditions it gives an increase of thrust. The increase of thrust is due physically to the action of secondary pressure acting over the annulus area of the ejector.

SECRET

2.1 Theoretically a larger secondary shroud diameter will give a greater gain in thrust but the increase is limited by two effects.

(a) The shroud must run filled at its exit otherwise the secondary pressure will fall to ambient pressure or lower and there will be a loss of thrust. The length of the shroud L, is here an important factor and there is an optimum length for any given shroud diameter.

(b) The primary jet must not be over-expanded otherwise there will be a shock and loss of thrust within or at the exit of the secondary shroud.

2.2 The secondary shroud diameter has been chosen as a compromise between thrust boost when using afterburner and minimum loss of thrust with afterburner off where effect (a), mentioned above, operates. The shroud length L has been chosen to give optimum performance with the particular diameter chosen. Note that too long a shroud ~~with~~ would give rise to unnecessary friction loss.

2.3 It is convenient to express the ejector geometry in non-dimensional form by using diameter ratio $D_s/D_p$ and spacing ratio $L/D_p$. The properties of the ejector are grouped into four parameters, secondary pressure ratio $P_s/p_a$, primary pressure ratio $P_p/p_a$, mass flow ratio corrected

SECRET

for effect of temperature $\frac{m_s \sqrt{T_s}}{m_p \sqrt{T_p}}$ and gross thrust ratio $F_{ej}/F_{jet}$ which we can regard as the ratio of ejector gross thrust to simple convergent nozzle gross thrust.

2.4 The characteristics of the JT 4A-25 ejector have been taken from NACA RM E52 L24 and are presented in carpet form in Figures 1, 2, 3 and 4 as follows:-

| Related Parameters:- | A/B | NO A/B |
|---|---|---|
| $\frac{M_s \sqrt{T_s}}{M_p \sqrt{T_p}}$, $\frac{P_s}{P_a}$, $\frac{P_p}{P_a}$ | Fig. 1<br>$L/D_p$ = 0.58<br>$D_s/D_p$ = 1.13 | Fig. 2<br>0.52<br>1.43 |
| $F_{ej}/F_{jet}$, $P_s/P_a$, $P_p/P_a$ | Fig. 3<br>$L/D_p$ = 0.58<br>$D_s/D_p$ = 1.13 | Fig. 4<br>0.52<br>1.41 |

TABLE 1.

Summary of the graphs of nozzle characteristics used in this report. Note that the above geometrical ratios do not correspond exactly with those now proposed for the JT 4A/25 to be fitted in the CF-105, but the differences are not thought to be significant.

SECRET

3.0 ESTIMATION OF THRUST BOOST GIVEN BY EJECTOR (See Appendix I)



The basic problem is to find the primary and secondary pressures and the primary and secondary airflows which act on the ejector. Once we know these operating conditions, it is a simple matter to find the increase of thrust by comparison with the experimental data given in NACA RM E52L24 and Figures 1-4.

3.1 The gross thrust ratio has been found as outlined above and is plotted in Figure 12 for maximum afterburning and military thrust (maximum non-afterburning). The gross thrust has also been computed for partial afterburning conditions at M = 1.5 and partial non-afterburning at M = 0.92. The calculations are listed in Part II of Report P/Power/51.

We are here concerned primarily with thrust but the calculations have also enabled estimations to be made of secondary mass flow and pressure recovery in duct of engine airflow.

4.0 CONVERSION OF GROSS THRUST RATIO INTO NET THRUST WITH ALLOWANCE FOR SPILLAGE DRAG AND INPUT MOMENTUM DRAG

The starting point of the calculation is the net thrust given in the engine brochure. From this, we find the gross thrust of the basic engine by subtracting the total pressure loss and adding the input momentum. The basic gross thrust is then factored by the gross thrust ratio to give the actual gross thrust with ejector from which we can find the actual net thrust ($F_{Ae}$) by subtracting the input momentum and spillage drag.

The procedure of this part of the estimation is given in Appendix II. The actual net thrust was obtained for maximum after-burning and military conditions over a range of Mach number and altitude and is plotted in Figures 13 and 13a for A/B lit and Figure 14 for no A/B. The thrust has also been estimated for partial afterburning conditions at M = 1.5 and partial thrust, no A/B at M = 0.92.

The thrust is also plotted as the ratio with atmospheric pressure and as its ratio with the basic net thrust with no ejector.

5.0 PROPOSED FUTURE WORK

It is hoped to issue a report in the near future to examine the following topics, some of which will represent a further examination of the somewhat arbitrary assumption used in the present report.

(a) The interdependence of duct inlet area, bypass inlet area and ejector size and their relation to the thrust-drag summation.

(b) A more rigorous determination of bypass inlet total pressure.

(c) A comparison of the ejector thrust-gain with that obtained from a convergent divergent nozzle of similar dimensions using one dimensional theory.

## APPENDIX I



## METHOD OF ESTIMATING THE EQUILIBRIUM BYPASS AIRFLOW AND THE ASSOCIATED PRESSURES AND THRUST FOR GIVEN ENGINE AND FLIGHT CONDITIONS

### INTRODUCTION

An iterative process is employed to establish the duct, engine and ejector operating point. Basically this involves finding the secondary airflow at which the pressure losses in the duct and bypass just account for the difference between the intake pressure and the secondary pressure at the ejector at the same secondary airflow.

We require to know the following variables:-

### Flight Parameters

Aircraft Mach Number

Aircraft altitude and appropriate ambient pressure and temperature

Engine throttle setting (convenient to define as a percentage of ideal brochure thrust)

### Geometry

Duct and bypass inlet areas

Ejector diameter ratio and spacing ratio ) appropriate to afterburner lit or not lit

### Pressure Losses

Pre-entry shock loss (at supersonic speed)

Inlet separation loss (at low subsonic speed)

Duct friction loss

Pressure loss in bypass

METHOD

Page 13 shows a specimen calculation which is examined, column by column in the following description:-

Column

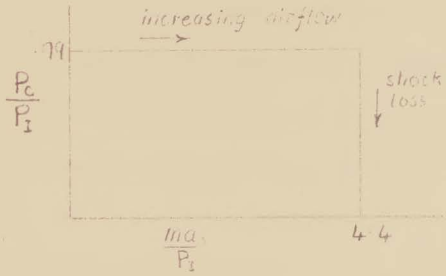
- \- Aircraft Mach number and altitude
- 1\. Aircraft speed in knots
- 2\. Aircraft speed in ft./sec.
- 3\. Ideal engine airflow, read from engine brochure
- 4\. Bypass area communicating with ejector, in sq.ins.
- 5\. Stagnation speed of sound $a_0 = a\left(1+\frac{\gamma-1}{2}M^2\right)^{\frac{1}{2}}$, $\gamma = 1.4$
- 6\. Square root of stagnation temp. $\sqrt{T_0} = \sqrt{T_{amb}}\left(1+\frac{\gamma-1}{2}M^2\right)^{\frac{1}{2}}$ $\gamma = 1.4$
- 7\. Ambient atmospheric pressure $p_a$, lb./sq.in.
- 8\. Stagnation pressure $p_o = p_a\left(1+\frac{\gamma-1}{2}M^2\right)^{\frac{\gamma}{\gamma-1}}$, $\gamma = 1.4$
- 9\. Shock loss ratio, ratio of total pressure behind shock structure to stagnation pressure, $\frac{P_I}{P_o}$, read from Figure 8 or at speeds of less than M = 0.5, there is a separation loss, caused by breakway of the flow round the intake lips, given in Figure 9.
- 10\. Total pressure at inlet $P_I = \left(\frac{P_I}{P_o}\right).P_o$
- 11\. Ratio of total pressure of air entering compressor to inlet total pressure, $\frac{P_c}{P_I}$. Note that when the bypass gills open at approximately M = 0.6, the duct boundary layer is taken into the bypass so that under these conditions, the core of air entering the engine suffers very little friction loss.

METHOD Cont'd.

SECRET

Column

11 cont. When the duct inlet is choked it is possible to get a shock wave in the diffusing part of the duct. Thus it is convenient to define $P_c/P_I$ in terms of the duct choking parameter $ma_o/P_I$ by means of the following diagram:-




$\frac{ma_o}{P_I}$ = 4.4, corresponds to choking mass flow in an inlet area of 5.6 sq.ft. or $\frac{m}{m*}$ = 1.0.

Note:

When the bypass gills are closed use a value of $P_c/P_I$ = .965.

12. Ratio of total pressure at compressor to stagnation pressure, $P_c/P_o$.

$$\frac{P_c}{P_o} = \frac{P_c}{P_I} \cdot \frac{P_I}{P_o}$$ i.e. (Col. 9) x (Col. 11)

13. Actual engine airflow $m_e = m' \cdot P_c/P_o$ i.e. (Col. 3) x Col. 12).

14. Duct inlet choking parameter $\frac{ma_o}{P_I} \cdot \frac{1}{144g}$ where m is total duct airflow.

(i.e. m= engine airflow + bypass airflow + cooler airflow)
(Column 13) (Column 21) (Column 22)

Make an initial guess for $\frac{ma_o}{P_I}$ based on the engine airflow plus five or ten percent depending on the Mach number

SECRET

SECRET

METHOD Cont'd.

Column

14. and then when the secondary airflow is established the initial guess can be checked.

15.

16. The ratio of total pressure of the air entering the bypass to the inlet total pressure, $P_B / P_I$.

Note that the secondary airflow is made up largely of the duct boundary layer. The total pressure of this flow has been assumed to lie half-way between the mean total pressure and the static pressure calculated from the mean duct Mach number at the compressor face. This assumption is justifiable where the mean duct Mach number is moderate (for JT 4A/25 $M_{Duct} \not> .6$)

$P_B / P_I$ has been plotted in Figure 10.

17. The total pressure at the bypass $P_B = (P_B / P_I) . P_I$.

18. The bypass airflow parameter $\frac{m_s \sqrt{T_o}}{AP_B g}$ (where A = 74 sq.ins.).

This has a maximum value of .0123 when the inlet gills are choked. The quantity refers to the air going to the ejector only and not to the cooler airflow. A suitable initial guess is .01. It is then adjusted in subsequent lines until the pressure losses through the bypass just makes up the difference between the bypass inlet pressure $P_B$ and the ejector secondary pressure $P_S$.

SECRET

METHOD Cont'd.

SECRET

Column

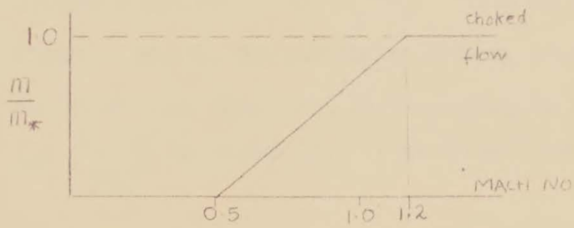
18. The pressure loss through the bypass is $P_B - P_4$ so $\frac{m_s \sqrt{T_o}}{AP_B}$ must be adjusted until $P_4 = P_S$, (i.e. $\frac{P_4}{P_a} = \frac{P_S}{P_a}$ or column 20 = column 26)

19. The ratio of the static pressure in the rear engine bay to the bypass inlet total pressure, $P_4 / P_B$. Note that the cross sectional area of the rear engine bay is very large, consequently the velocity is small and it may be assumed that $p_4 = P_4$. Figure 11 gives the ratio $p_4 / P_B$ in terms of the parameter $\frac{m_s \sqrt{T}}{AP_B}$

20. Using column 7 and 17 convert $\frac{P_4}{P_B}$ into $\frac{P_4}{P_a}$.

21. Secondary airflow through the bypass, lb./sec. Find from columns 4, 6, 17 and 18.

22. Oil cooler airflow, lb./sec. The inlet to the oil cooler is assumed to be choked above a Mach number of 1.2, falling linearly to zero flow at M = 0.5. Thus the cooler airflow can be represented in the following diagram:-




Arbitrary diagram of oil cooler airflow.

SECRET

METHOD Cont'd.

Column

23. The primary jet pressure ratio $P_p/p_a$ is calculated from the Pratt & Whitney Brochure (Curve 17227 for no afterburner, letter of March 24 1955. for afterburner). For convenience these have been replotted in Figures 5 and 6. To get the actual $P_p/p_a$, the ideal $P_p/p_a$ from the brochure is factored by the duct loss ratio $P_c/P_o$.

24. The temperature correction factor $\sqrt{\frac{T_S}{T_p}}$ to be applied to the secondary and primary airflow.

$T_S$ has been defined as $T_S = T_o + \Delta T$ where $\Delta T$ is given in the following table:-

TABLE II

| Mach Number | | 0 - 1.3 | 1.5 - 2.0 |
|---|---|---|---|
| $\Delta T$ | A/B Lit | 75°C | 50°C |
| | No A/B | 100°C | 75°C |

25. Corrected mass flow ratio $\frac{m_s}{m_p} \cdot \sqrt{\frac{T_s}{T_p}}$

26. Secondary pressure ratio $P_s/p_a$. This is found from Figure 1 or 2 at the particular primary pressure ratio and corrected mass flow ratio corresponding to the value chosen for $\frac{m_s \sqrt{T_o}}{AP_B}$ in column 18. As mentioned under column (18)

SECRET

METHOD Cont'd.

Column

26. the value of the secondary airflow parameter, $\frac{m_s \sqrt{T}}{AP}$, is improved by iteration in subsequent lines until $\frac{P_s}{p_a} = \frac{P_4}{p_a}$

27. Gross thrust ratio $\frac{F_{ej}}{F_{Jet}}$. This is read from Figure 12 for the value of primary pressure ratio and the value of secondary pressure established in Column 26 above.

Part II of the Appendix details the procedure for finding the value of the thrust increase from the gross thrust ratio.

SECRET

Case A/B lit    Area of bypass going to cooler 24 □"

M = 1·8

$p/p_0$ = ·1741  $T/T_0$ = ·6068  $\frac{a}{a_0}$ = ·7790    · Estimate

| | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 |
|---|---|---|---|---|---|---|---|---|---|---|
| | knots | ft/sec | ideal engine airflow | A □" | $a_0$ | $\sqrt{T_0}$ | $p_a$ | $p_0$ | Shock loss $P_I/p_0$ | $P_I$ |
| 25,000 | 1084 | 1830 | 334 | 74 | 1308 | 19·8 | 5·452 | 31·3 | ·925 | 29 |
| | | | | | | | | | | |
| | | | | | | | | | | |
| | | | | | | | | | | |
| 35,000 | 1038 | 1751 | 251 | | 1250 | 18·9 | 3·458 | 19·9 | ·925 | 18·4 |
| | | | | | | | | | | |
| | | | | | | | | | | |
| | | | | | | | | | | |
| | | | | | | | | | | |
| 45,000 | 1037·5 | 1751 | 154 | | 1249 | 18·96 | 2·143 | 12·32 | ·927 | 11·42 |
| | | | | | | | | | | |
| | | | | | | | | | | |
| | | | | | | | | | | |
| | | | | | | | | | | |
| | | | | | | | | | | |

FORM 1544

...s going to cooler 24□"

790

## Example

**Estimation of Ejector Gross thrust ratio.**

AVR...

AIRC...

WEIG...

C. G...

| 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 | 14 | 15 | 16 | 17 | 18 | 19 | 20 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| [illegible] | $p_a$ | $p_0$ | Shock loss $P_I/P_0$ | $P_I$ | $P_c/P_I$ | $P_c/p_0$ | Engine airflow lb/sec | $\frac{m a_0}{P_I g}$ | | $P_B/P_I$ | $P_B$ | $\frac{m_s\sqrt{T_0}}{A P_B g}$ | $p_4/P_B$ | $p_4/p_a$ |
| ...8 | 5.452 | 31.3 | .925 | 29 | .99 | .915 | 306 | 3.75<br>3.42 | | .90 | 26.1 | .01 | .725 | 3.47... |
| | | | | | | | | | | | | .01114 | .606 | 2.9... |
| | | | | | | | | 3.45 | | .92 | 26.6 | .0112 | .6 | (2.94...) |
| | | | | | | | | | | | | | | |
| ...9 | 3.458 | 19.9 | .925 | 18.4 | .99 | .915 | 230 | 3.95<br>3.8 | | .884 | 16.25 | .0107 | .66 | 3.1... |
| | | | | | | | | | | | | .01034 | .694 | 3.26... |
| | | | | | | | | 3.83 | | .893 | 16.43 | .0105 | .682 | 3.24... |
| | | | | | | | | | | | | .01045 | .686 | (3.26...) |
| | | | | | | | | | | | | | | |
| ...96 | 2.143 | 12.32 | .927 | 11.42 | .99 | .916 | 141 | 3.77<br>3.8 | | .892 | 10.2 | .0106 | .67 | (3.19) |
| | | | | | | | | ↑ | | | | | Notice the... | |

Note that the initial guess of $\frac{m a_0}{P_I}$ must be checked back before the iteration is concluded.

## APPENDIX II

SECRET

### CONVERSION OF GROSS THRUST RATIO TO NET THRUST

1.0 NOTATION

$F_n'$ Ideal net thrust from brochure.

$F_{Ae}$ Actual net thrust with ejector.

$F_{Ao}$ Actual net thrust no ejector.

$F_{ej}$ Gross thrust with ejector.

$F_{Jet}$ Gross thrust with plain convergent nozzle.

Note:- $F_{ej} / F_{Jet}$ is called the gross thrust ratio and $F_{Ae} / F_{Ao}$ the net thrust ratio.

$\Delta F_D$ Loss of thrust due to duct loss.

$\Delta F_S$ Loss of thrust due "spillage drag".

$C_D$ Duct loss coefficient from engine brochure.

$m$ Total duct airflow )
$V$ Aircraft velocity ) gives $\frac{mV}{g}$ input momentum

$A_\infty$ Free stream area of air entering duct.

X,Y Spillage drag parameters (see table overleaf).

2. An example of the method used to find the actual net thrust with ejector is given on page 16. The method is based on the following formulae:-

$$F_{Ao} = F_n' - (\Delta F_D + \Delta F_S)$$

$$F_{Ae} = \left(F_n' - \Delta F_D + \frac{mV}{g}\right) \cdot \frac{F_{ej}}{F_{Jet}} - \left(\frac{mV}{g} + \Delta F_S\right)$$

Where - $\Delta F_D = F_n' P$

$$\Delta F_S = P_\infty \cdot 144 \, (X - A_\infty Y)$$

$$A = \frac{m}{a_\infty \rho_\infty g M_\infty}$$

SECRET

Case A/B lit

35,000 ft

$\delta = .425$

EXAMPLE

Conversion of gross thrust rati

$$F_{A_o}' = F_n' - (\Delta F_D + \Delta F_s')$$

$$F_{A_e}' = \left(F_n' - \Delta F_D + \frac{mV}{g}\right)\frac{F_{ej}}{F_{jet}} - \left(\frac{mV}{g} + \Delta F_s'\right)$$

| | | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 |
|---|---|---|---|---|---|---|---|---|---|---|---|
| knots | Mach No | $F_n'$ | $\frac{F_n'}{\delta_{amb}}$ | $(1+C_d)$ | $\frac{P_c}{P_I}\cdot\frac{P_I}{P_o}$ | $1-\frac{P_c}{P_o}$ | $\Delta F_D$ | $m_e$ lb/sec | $m_s$ | $m_c$ | $m = m_e+m_s+m_c$ |
| 432 | ·75 | 9200 | 39,140 | 1·33 | | | | | | | |
| 530 | ·92 | 10,600 | 45,106 | 1·288 | ·928 | ·072 | 983·0 | 107·66 | 4·65 | 2·9 | 115·2 |
| 663 | 1·15 | 12,900 | 54,900 | 1·235 | ·934 | ·066 | 1051·5 | 134·78 | 6·75 | 3·7 | 145·2 |
| 749 | 1·3 | 14,700 | 62,553 | 1·21 | ·938 | ·062 | 1102·8 | 155·7 | 8·75 | 4·3 | 168·7 |
| 865 | 1·5 | 16,900 | 71,900 | 1·18 | ·954 | ·046 | 917·3 | 189·4 | 12·6 | 5·5 | 207·5 |
| 922 | 1·6 | 18,000 | 76,600 | 1·17 | ·953 | ·047 | 989·8 | 205 | 15·2 | 6·4 | 226·6 |
| 1038 | 1·8 | 19,900 | 84,600 | 1·155 | ·918 | ·082 | 1884·7 | 230 | 21·7 | 8·2 | 259·9 |
| 1152 | 2·0 | 21,400 | 91,000 | 1·144 | ·85 | ·15 | 3672·2 | 244 | 29·1 | 10·5 | 283·6 |

FORM 1344

EXAMPLE

...version of gross thrust ratio to net thrust

from P&W brochure curve 16805.

$$\Delta F_D = F_n'\left(1 - \frac{P_c}{P_0}\right)(1 + C_D)$$

$$- \left(\frac{mV}{g} + \Delta F_s'\right) \qquad \Delta F_S = P_\infty (X - A_\infty Y) 144. \qquad A_\infty = \frac{m}{a_\infty \rho_\infty g M_\infty}$$

AVRO A...

AIRCRAFT

WEIGHT

C. G. POSIT...

| 7 | 8 | 9 | 10 | 11 | 12 | 13 | 14 | 15 | 16 | 17 | 18 | 19 | 20 | 21 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| $m_e$ .../sec | $m_s$ | $m_c$ | $m = m_e + m_s + m_c$ | V ft/sec | $\frac{mV}{g}$ | $F_n' - \Delta F_D + \frac{mV}{g}$ | $\frac{F_{ej}}{F_{jet}}$ | (13)×(14) | $a_\infty$ ft/sec | $P_\infty$ | $\rho_\infty$ | $A_\infty$ | Y | $A_\infty$... |
| | | | | | | | 1.016 | | 973 | 3.458 | .000737 | – | | |
| ...07.66 | 4.65 | 2.9 | 115.2 | 895 | 3202 | 12,819 | 1.028 | 13178 | | | | – | | |
| ...34.78 | 6.75 | 3.7 | 145.2 | 1120 | 5050.4 | 16,899 | 1.05 | 17,744 | | | | 5.468 | .375 | 2.05... |
| ...55.7 | 8.75 | 4.3 | 168.7 | 1265 | 6627.5 | 20,225. | 1.0615 | 21,469 | | | | 5.62 | .8049 | 4.51...<br>4.55... |
| ...89.4 | 12.6 | 5.5 | 207.5 | 1460 | 9407 | 25,389 | 1.0735 | 27,256 | | | | 5.99 | 1.459 | 8.362...<br>8.739... |
| ...205 | 15.2 | 6.4 | 226.6 | 1557 | 10,958 | 27,968 | 1.0785 | 30,163 | | | | 6.134 | 1.822<br>1.338 | 10.27...<br>8.20... |
| ...230 | 21.7 | 8.2 | 257.9 | 1752 | 14,14.1 | 32,156 | 1.0845 | 34,873 | | | | 6.253 | 2.237 | 13.99... |
| ...244 | 29.1 | 10.5 | 283.6 | 1946 | 17,13.8 | 34,866 | 1.088 | 37,934 | | | | 6.140 | 3.163 | 19.42... |

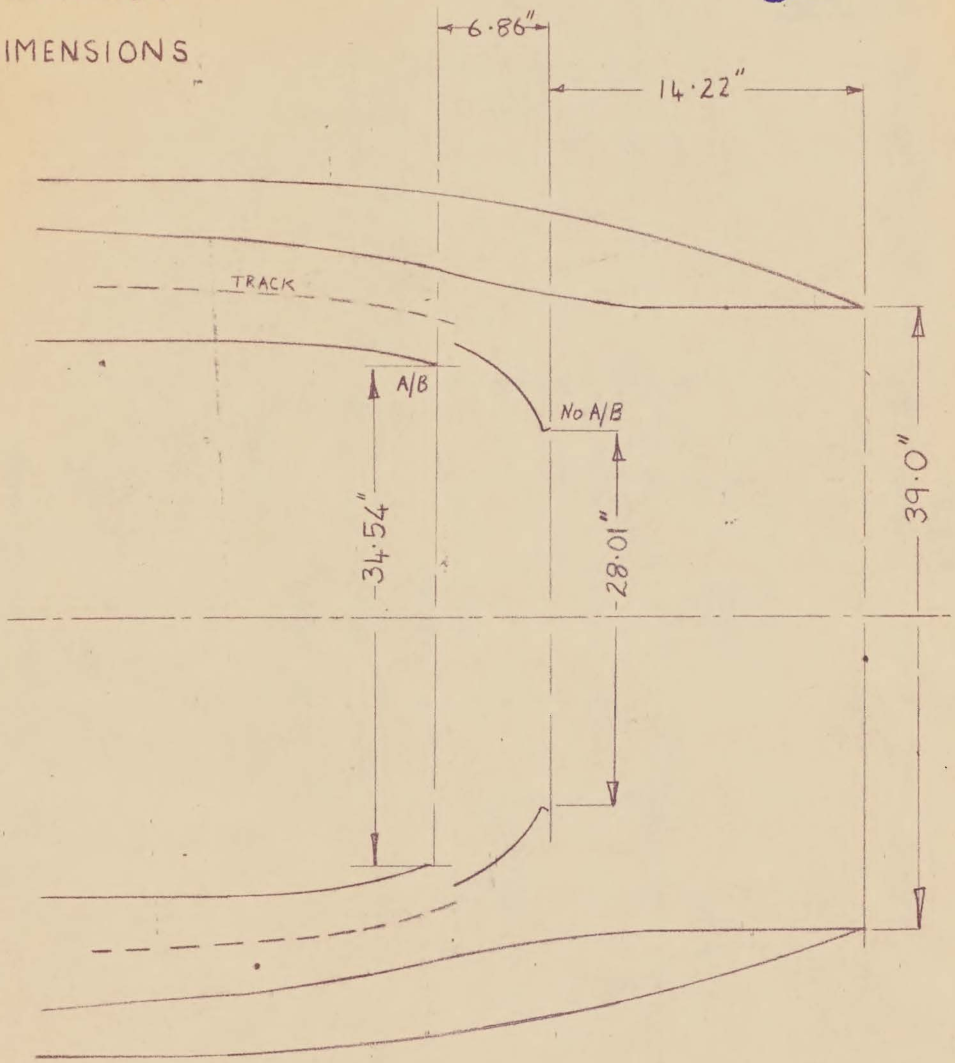
A. V. ROE CANADA LIMITED
MALTON - ONTARIO
**TECHNICAL DEPARTMENT (Aircraft)**

REPORT NO. P. Power 51.
SHEET NO. 17

| AIRCRAFT: | | PREPARED BY | DATE |
|---|---|---|---|
| CF 105 | J 75 EJECTOR | J MONK | 31 AUG 55 |
| | | CHECKED BY | DATE |
| | | | |

SECRET

FINALISED
DIMENSIONS




| | A/B ON | A/B OFF |
|---|---|---|
| SPACING RATIO | 0·612 | 0·507 |
| DIA. RATIO | 1·13 | 1·39 |

SECRET

FORM 1319 A

A. V. ROE CANADA LIMITED
MALTON · ONTARIO
**TECHNICAL DEPARTMENT (Aircraft)**

REPORT NO. P/POWER/51
SHEET NO. 18

| PREPARED BY | DATE |
|---|---|
| J.R. Monk | Oct. '55 |
| CHECKED BY | DATE |
| | |

AIRCRAFT:
CF-105
JT 4A-25

SECRET

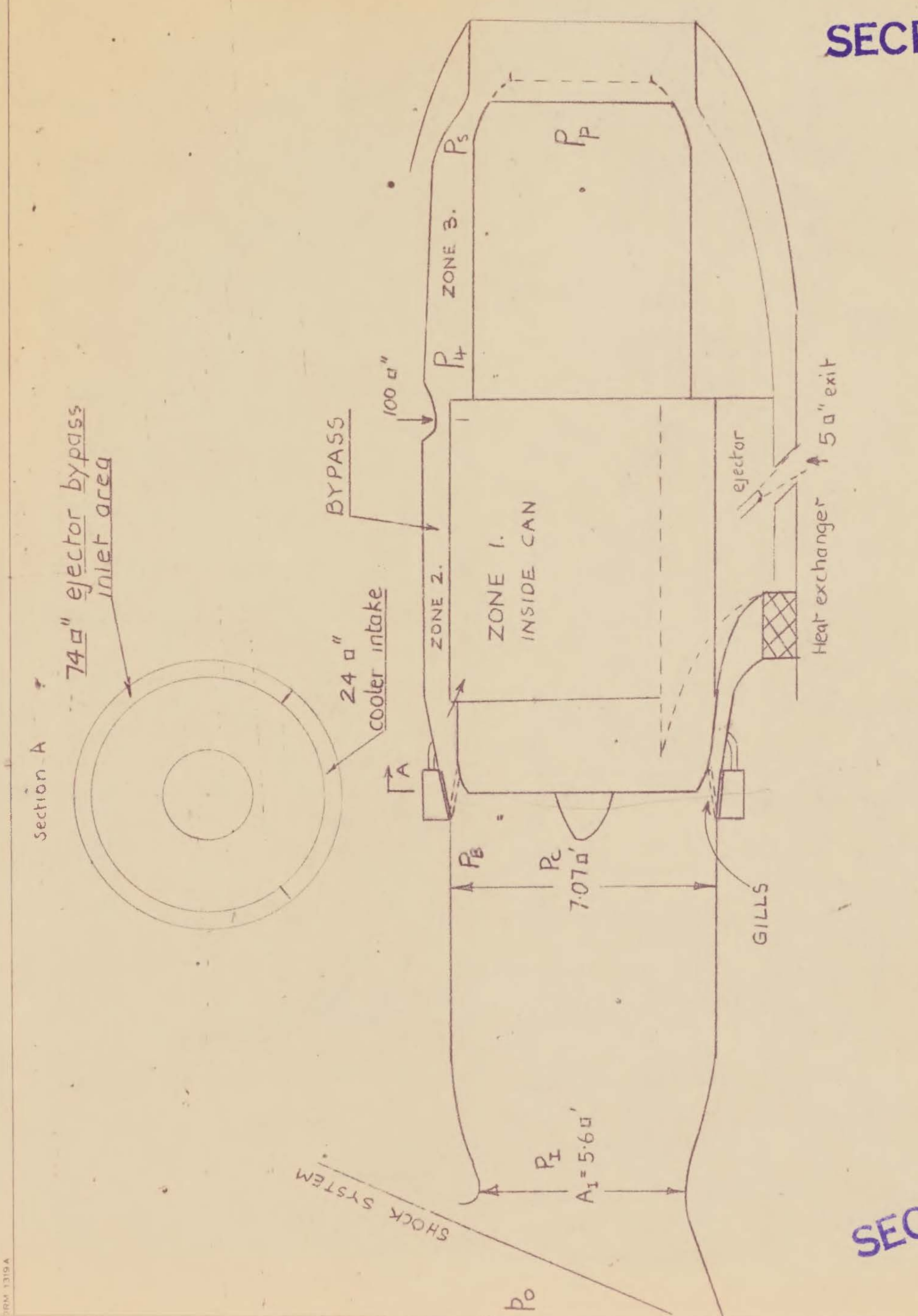


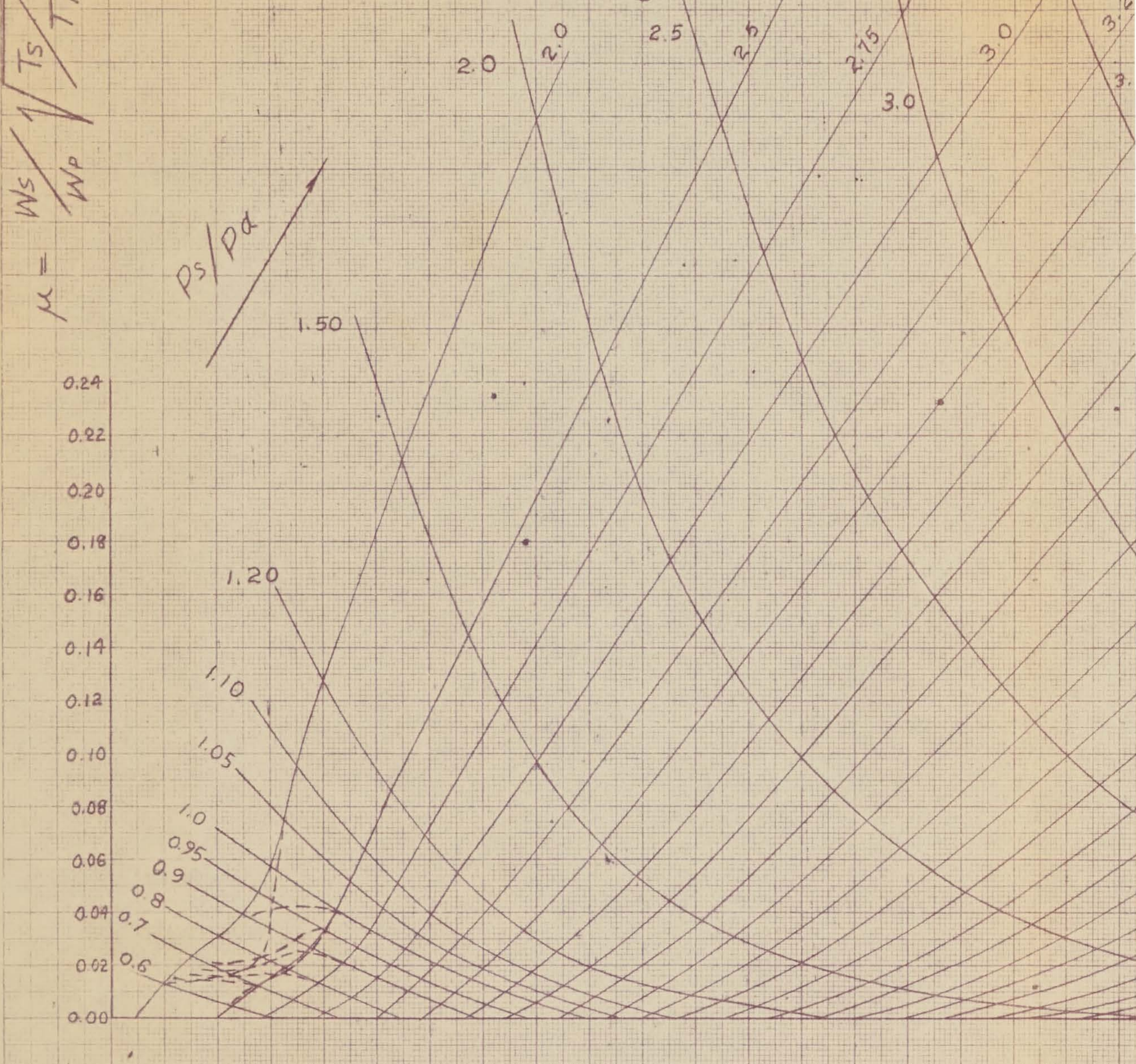

DIAGRAM OF DUCT, ENGINE BYPASS AND EJECTOR SYSTEM.

SECRET

ORM 1319A

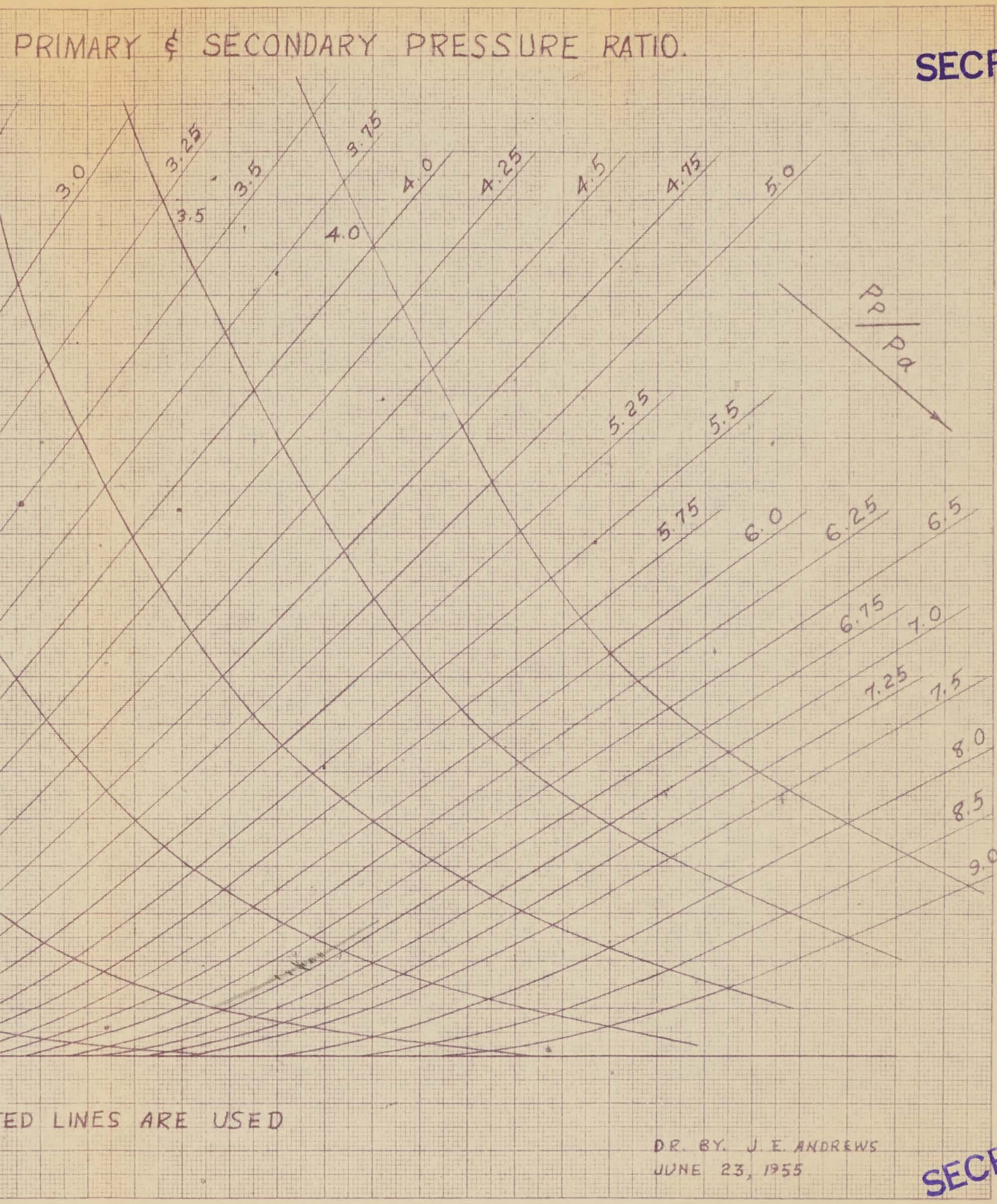
FIG 1. MASS FLOW RATIO AGAINST PRIMARY &

FOR DIAMETER RATIO = 1.13
SPACING RATIO = 0.58.




NOTE FOR SPACING RATIO 0.52 DOTTED LINES ARE

SECRET




SECRET

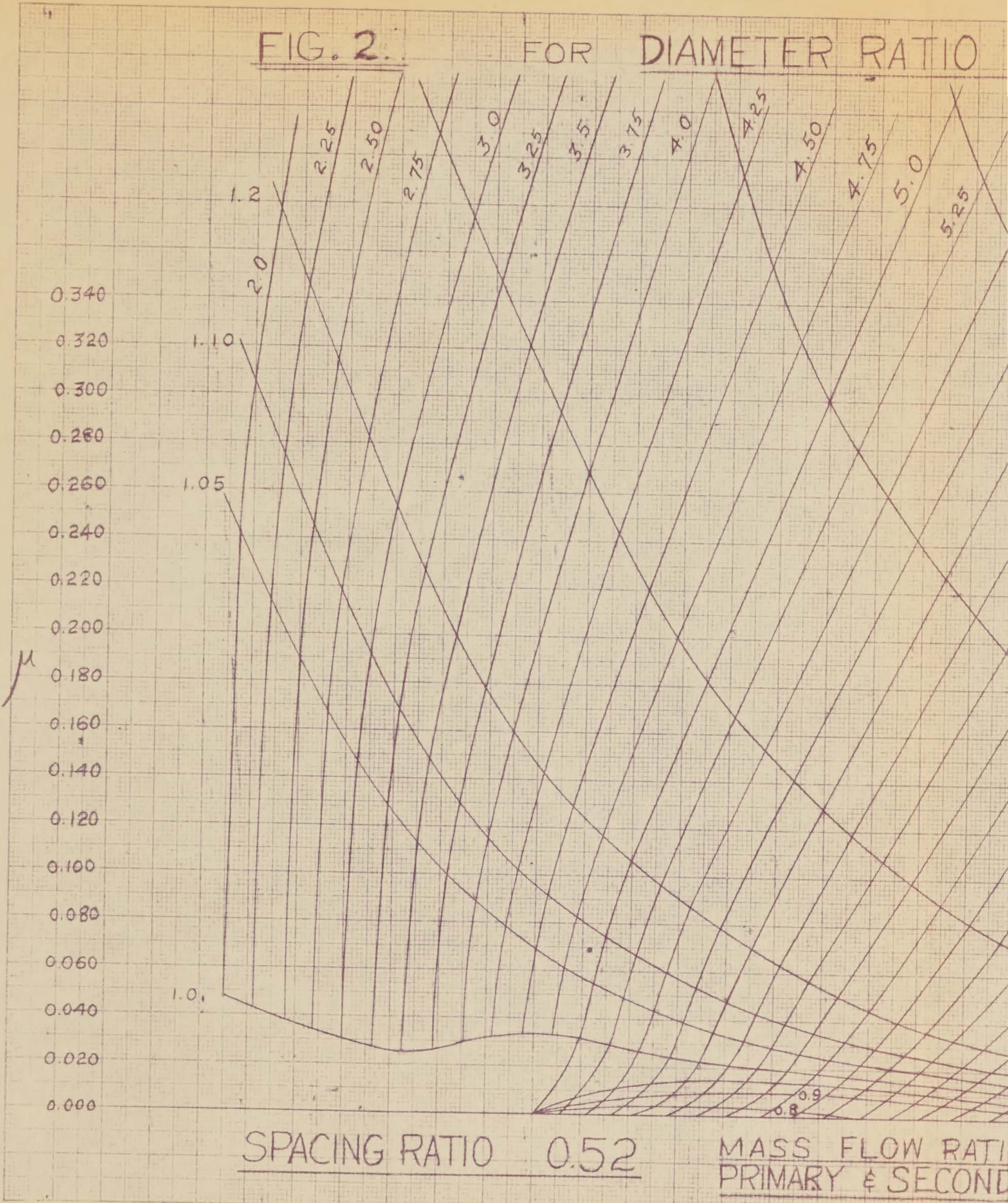
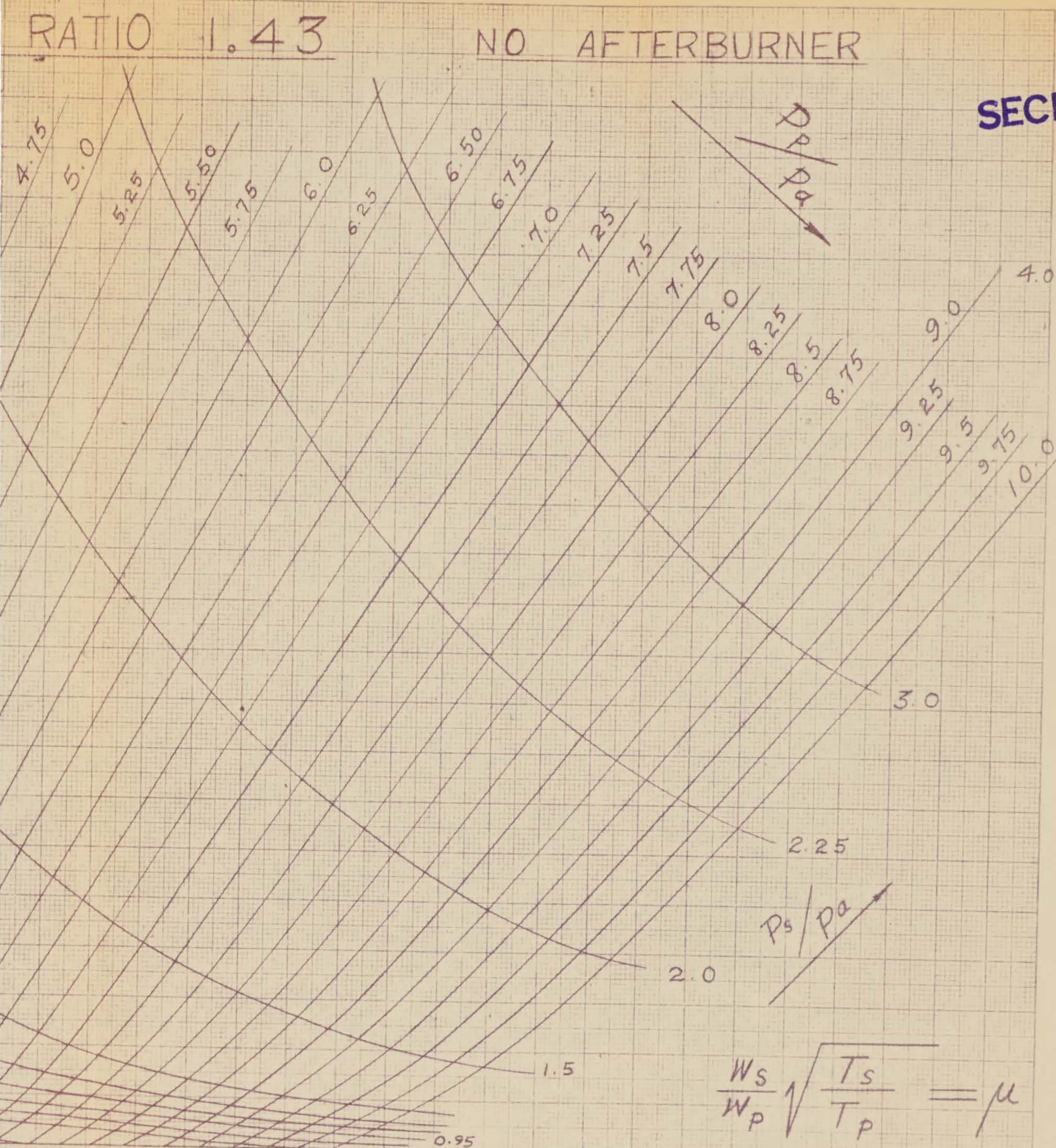


FIG. 2.
FOR DIAMETER RATIO
2.25
2.50
2.75
3.0
3.25
3.5
3.75
4.0
4.25
4.50
4.75
5.0
5.25
1.2
2.0
1.10
1.05
1.0
0.9
0.8
μ
0.340
0.320
0.300
0.280
0.260
0.240
0.220
0.200
0.180
0.160
0.140
0.120
0.100
0.080
0.060
0.040
0.020
0.000
SPACING RATIO 0.52
MASS FLOW RATI
PRIMARY & SECOND




FLOW RATIO AGAINST
Y & SECONDARY PRESSURE RATIO

MAY 31, 1955
DR. BY J. E. ANDREWS

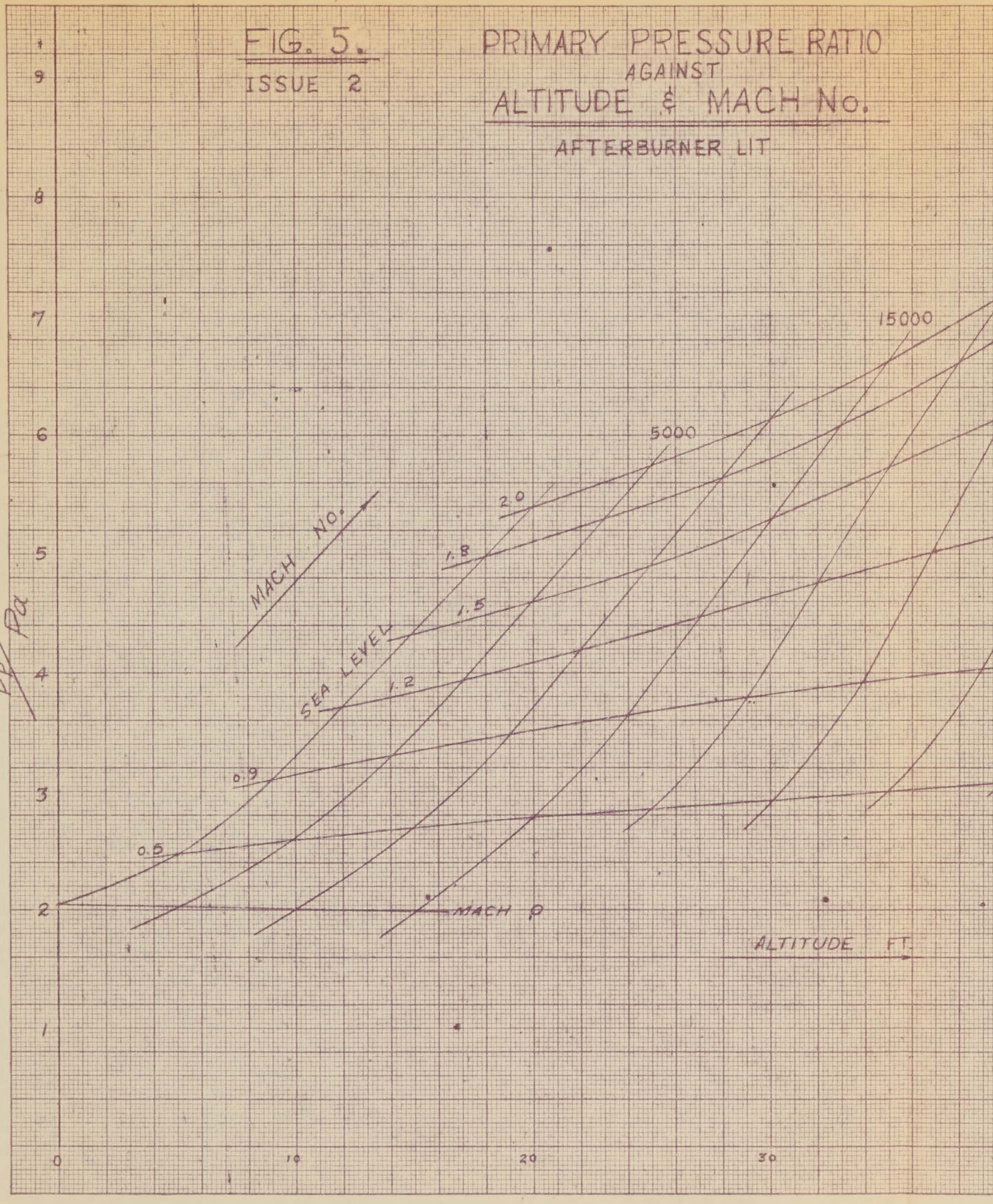



FIG. 5.
ISSUE 2

PRIMARY PRESSURE RATIO AGAINST ALTITUDE & MACH No.

AFTERBURNER LIT

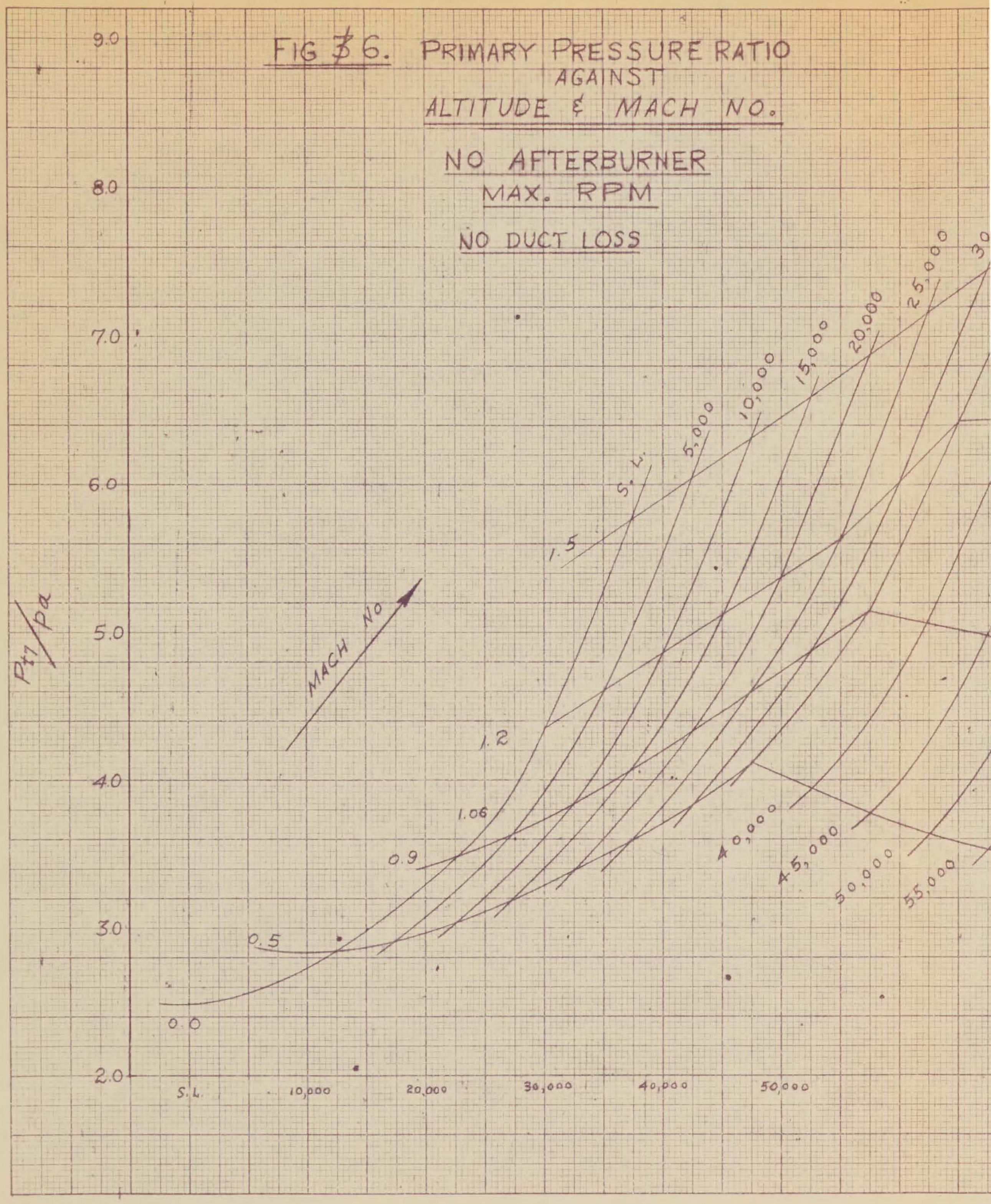


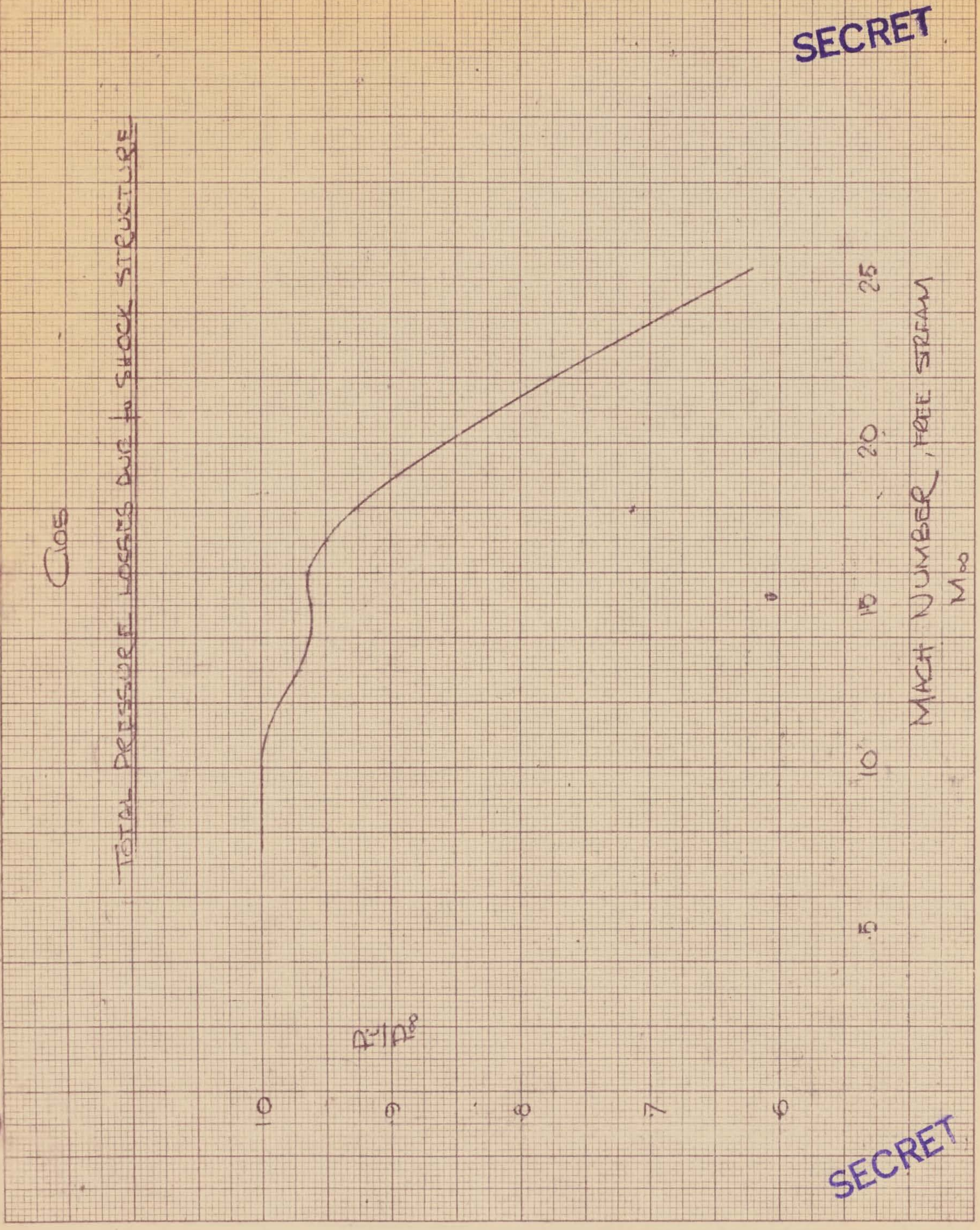
FIG 36. PRIMARY PRESSURE RATIO
AGAINST
ALTITUDE & MACH NO.
NO AFTERBURNER
MAX. RPM
NO DUCT LOSS
Pt7/pa
MACH NO
9.0
8.0
7.0
6.0
5.0
4.0
3.0
2.0
S.L.
10,000
20,000
30,000
40,000
50,000
0.0
0.5
0.9
1.06
1.2
1.5
S.L.
5,000
10,000
15,000
20,000
25,000
40,000
45,000
50,000
55,000

AIRCRAFT Gos | COMPONENT PERFORMANCE | SHEET No. 2 | REPORT No. 2

A. U. W. | DATE Jan. '58. | PREP. BY [illegible]

FIG. 8.

SECRET




SECRET



FORM 1746

5.5.55. J MONK.

CF 105 5.6 □' INTAKE. J75

SECRET

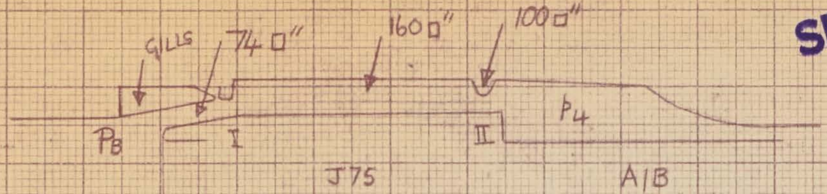
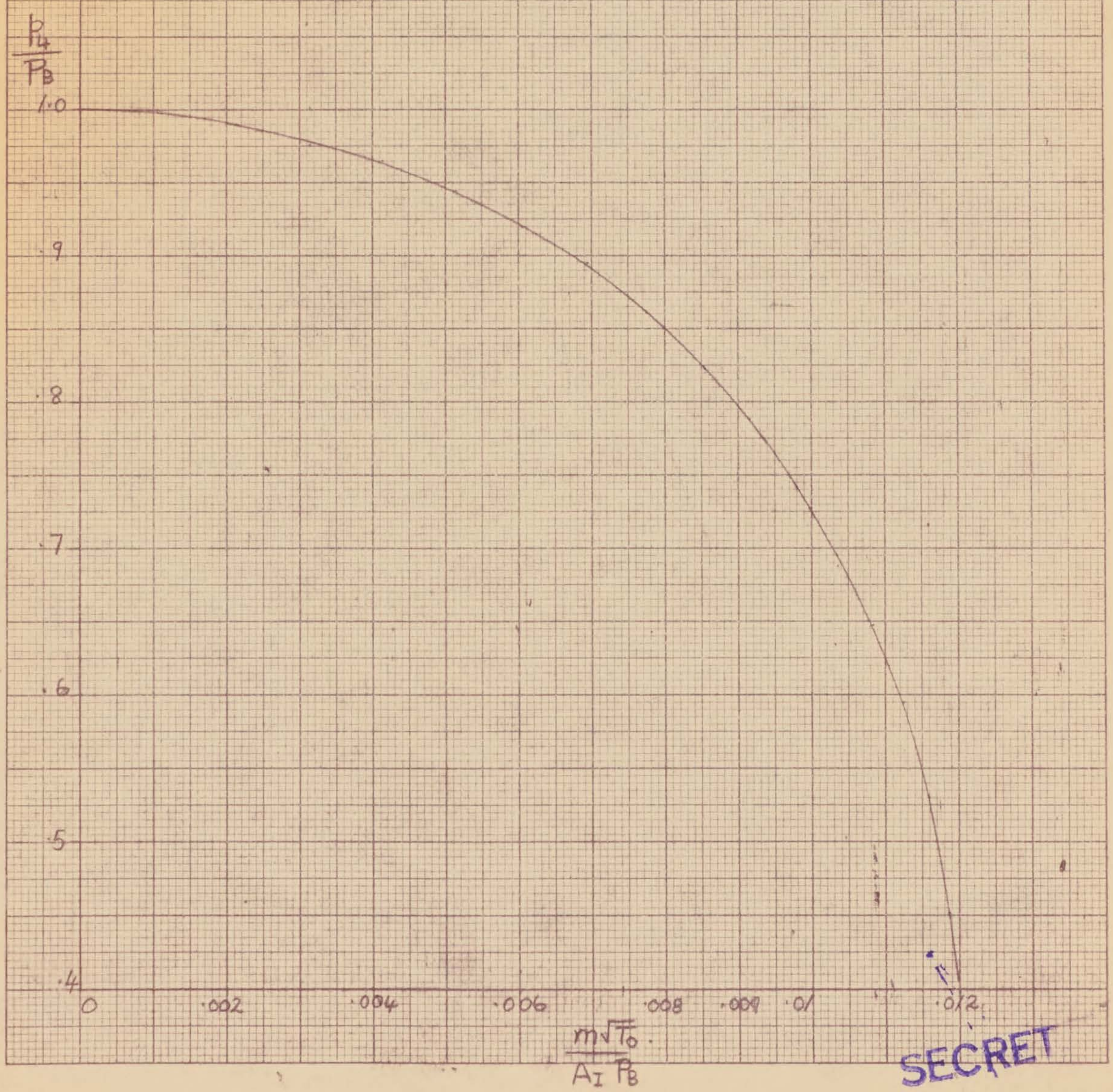



Fig 11. Relationship of $p_4$ and $P_B$ at various Bypass flow parameter $\frac{m\sqrt{T}}{A_I P_B}$ assuming no diffusion after gills at (I) and 100□" restriction (II)

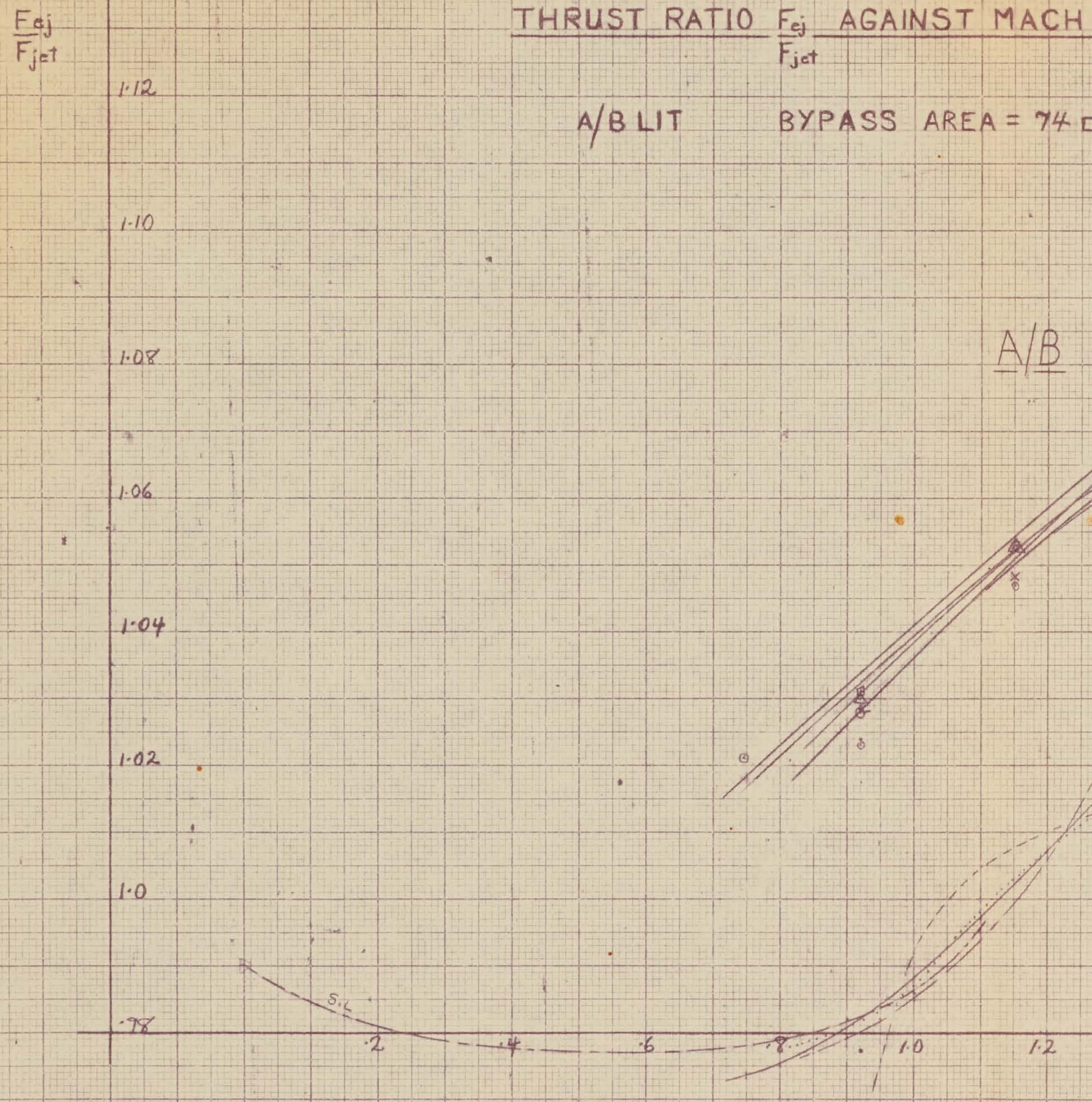



SECRET

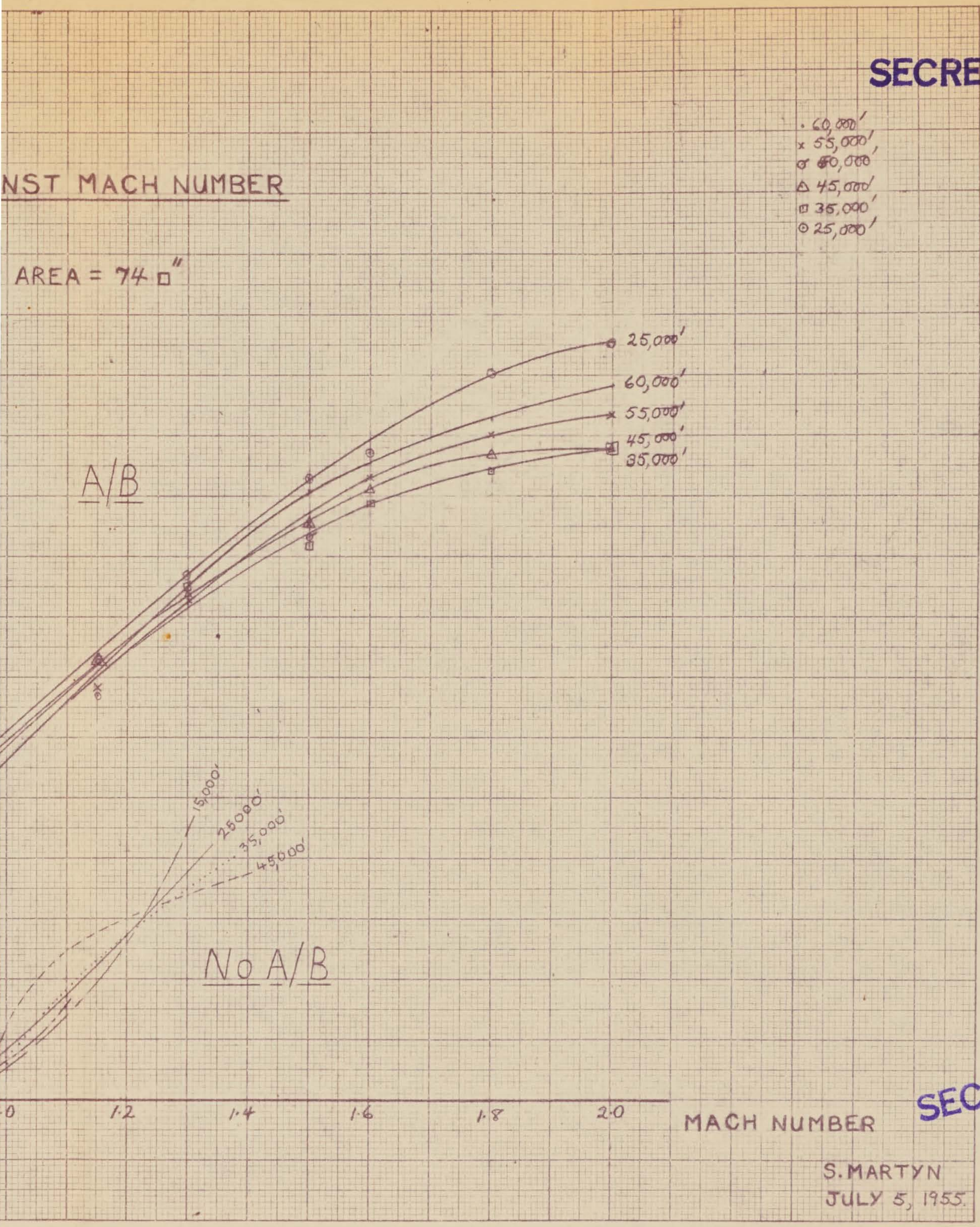
FIG 12.

THRUST RATIO $\frac{F_{ej}}{F_{jet}}$ AGAINST MACH

A/B LIT BYPASS AREA = 74



SECRET

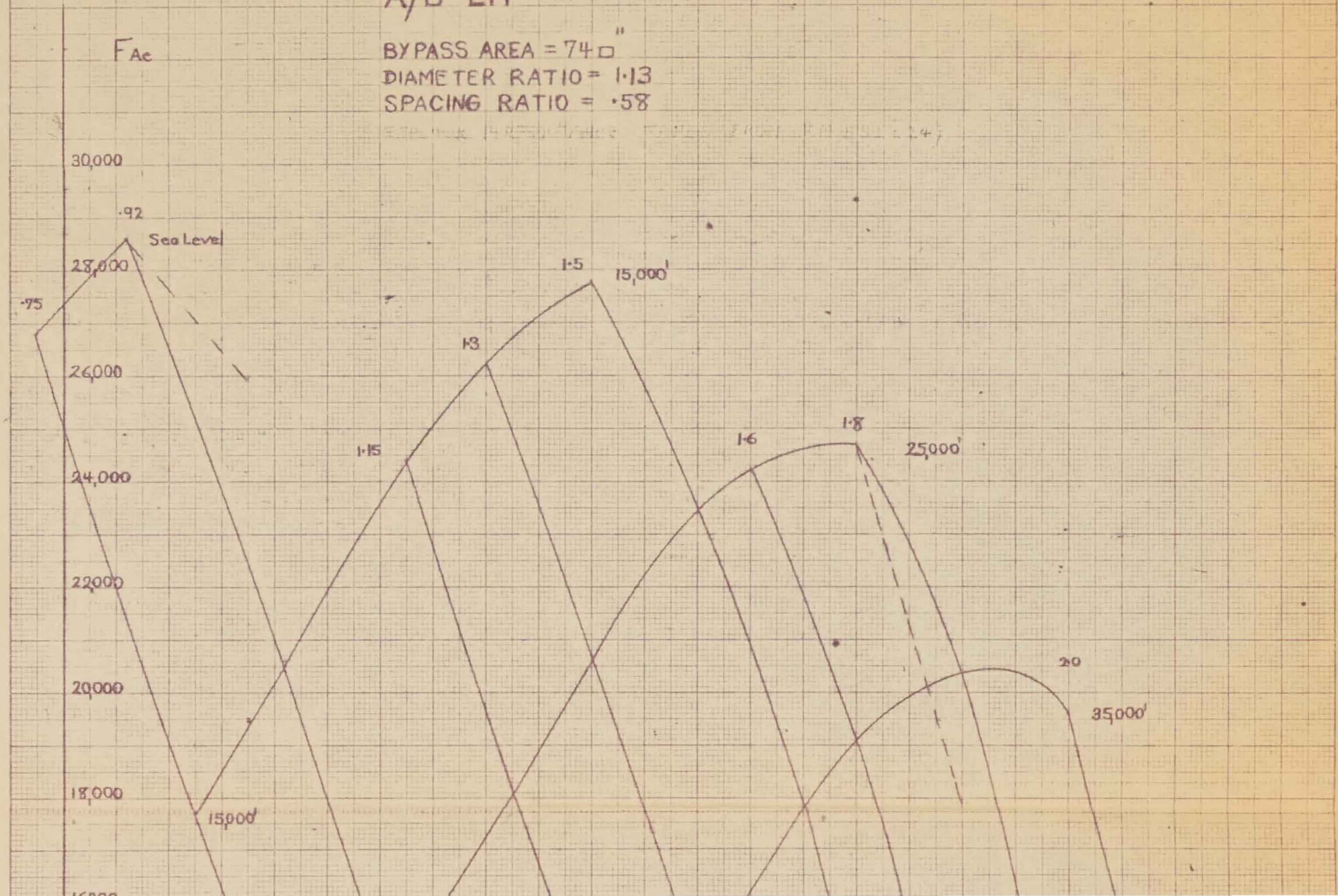



SECRET

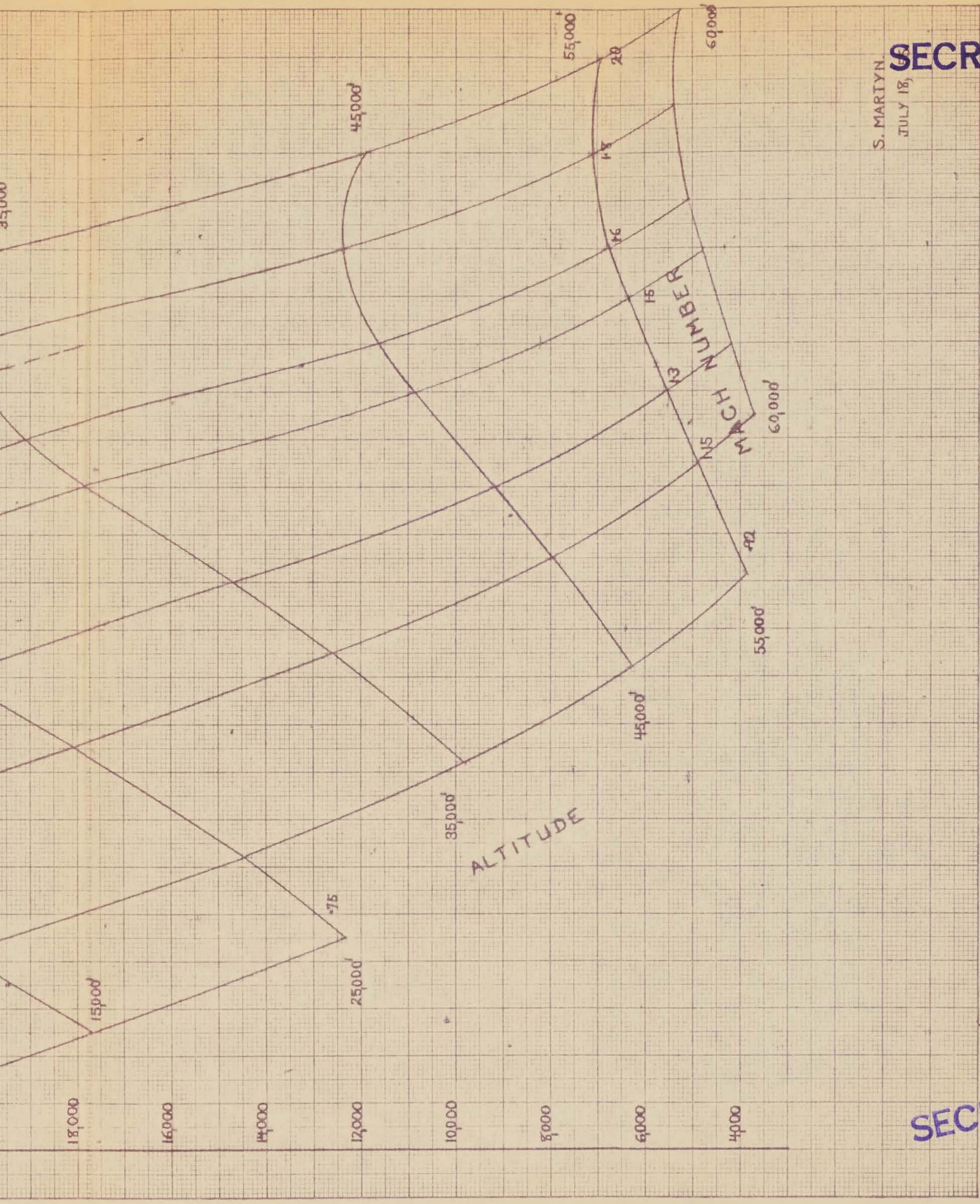
FIG. 13.

## J75 NET THRUST $F_{Ae}$ AGAINST MACH NUMBER

A/B LIT



SECRET




SECRET

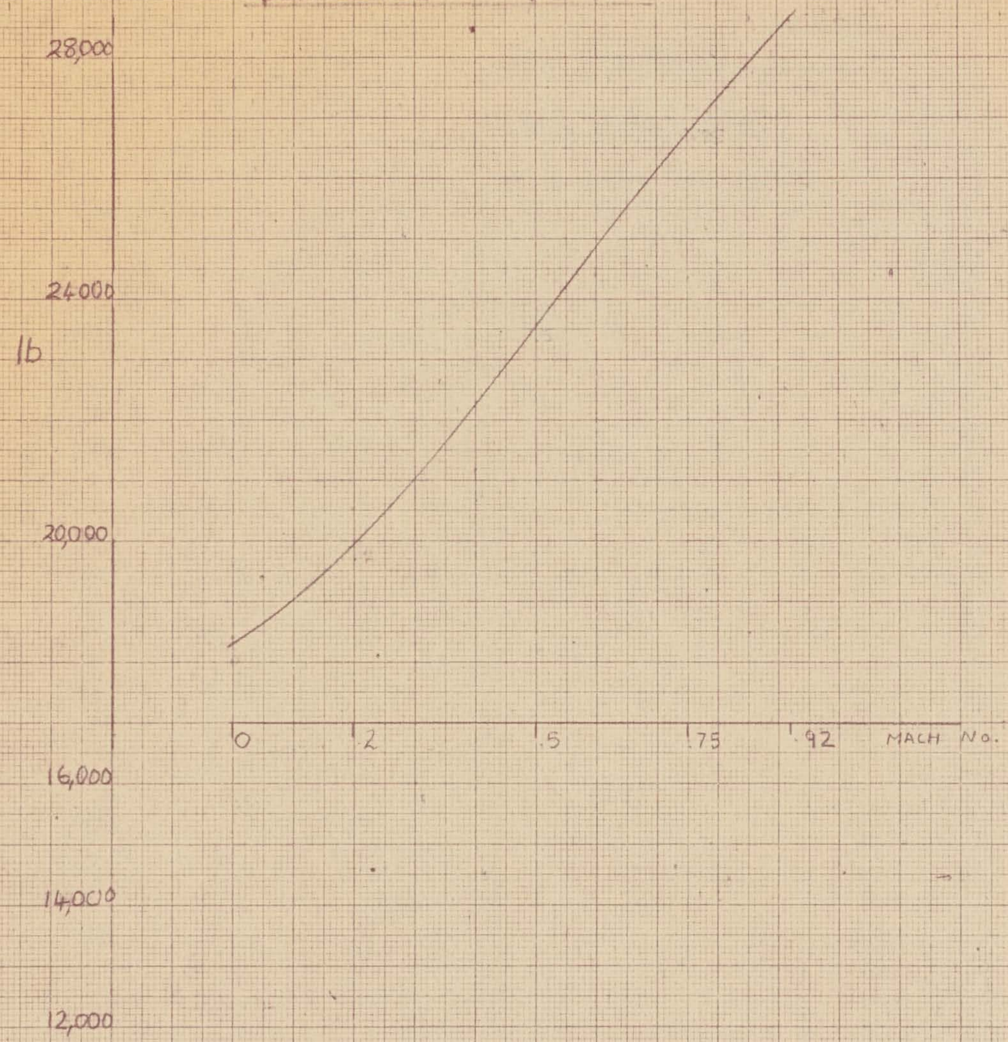
SECRET

FIG. 13a.

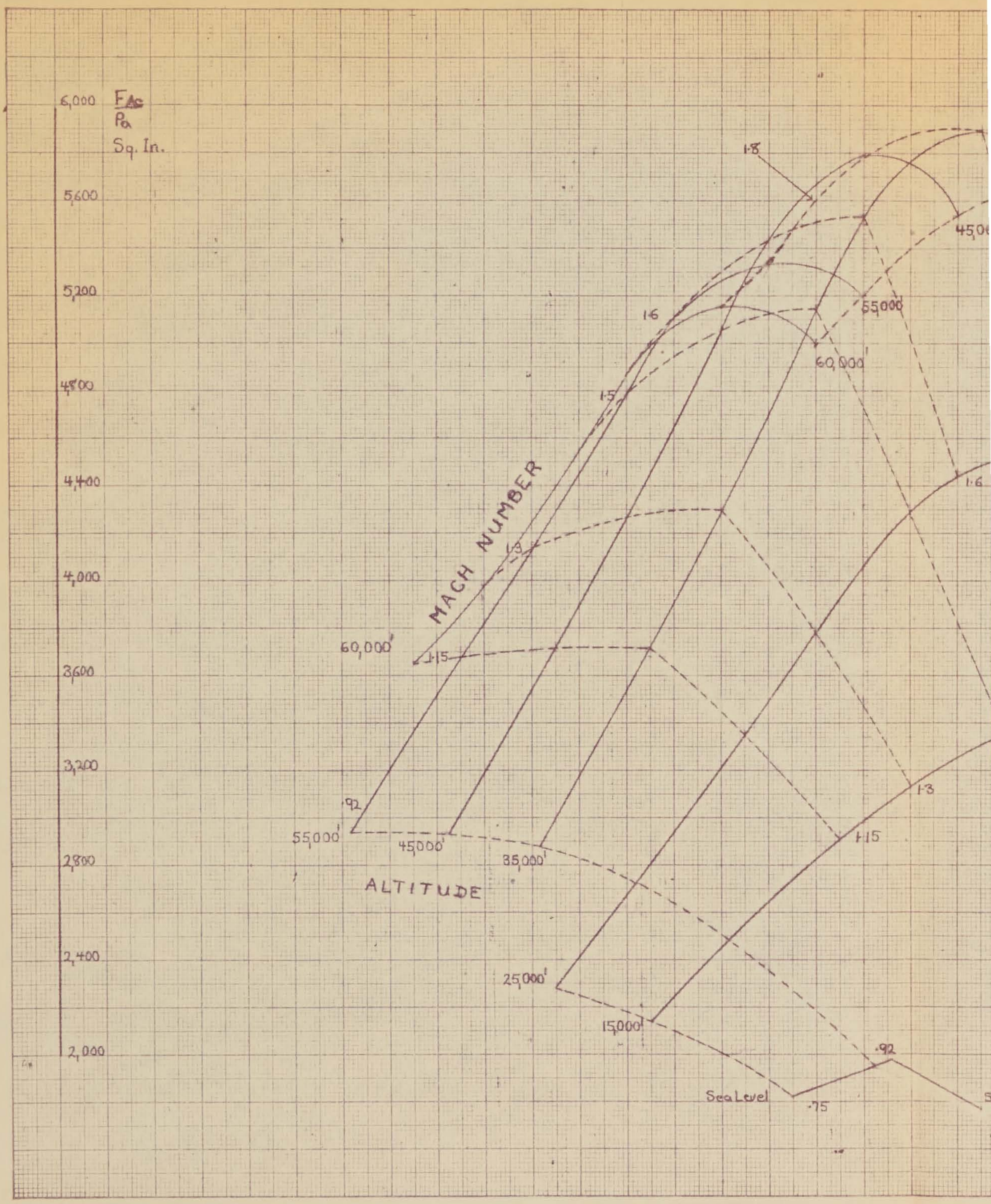
J75:- NET THRUST

SEA LEVEL A/B LIT

Ejector
Dia ratio 1·13
Spacing ratio 0·82




SECRET

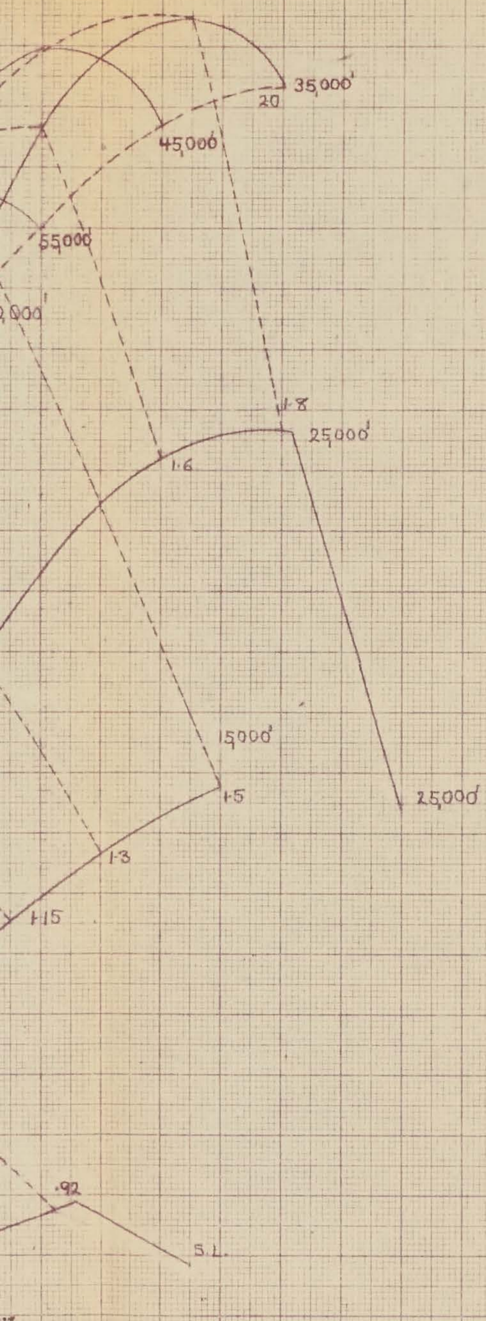


$F_{Ac}$ / $P_a$ Sq. In.
6,000
5,600
5,200
4,800
4,400
4,000
3,600
3,200
2,800
2,400
2,000
MACH NUMBER
ALTITUDE
1.8
1.6
1.5
1.3
1.15
.92
.75
60,000'
55,000'
45,000'
35,000'
25,000'
15,000'
Sea Level

SECRET

*Fig 13b*

## J 75 :- THRUST / ATMOSPHERIC PRESSURE AGAINST MACH NUMBER & ALTITUDE




EJECTOR EFFECT TAKEN FROM RM E52L24

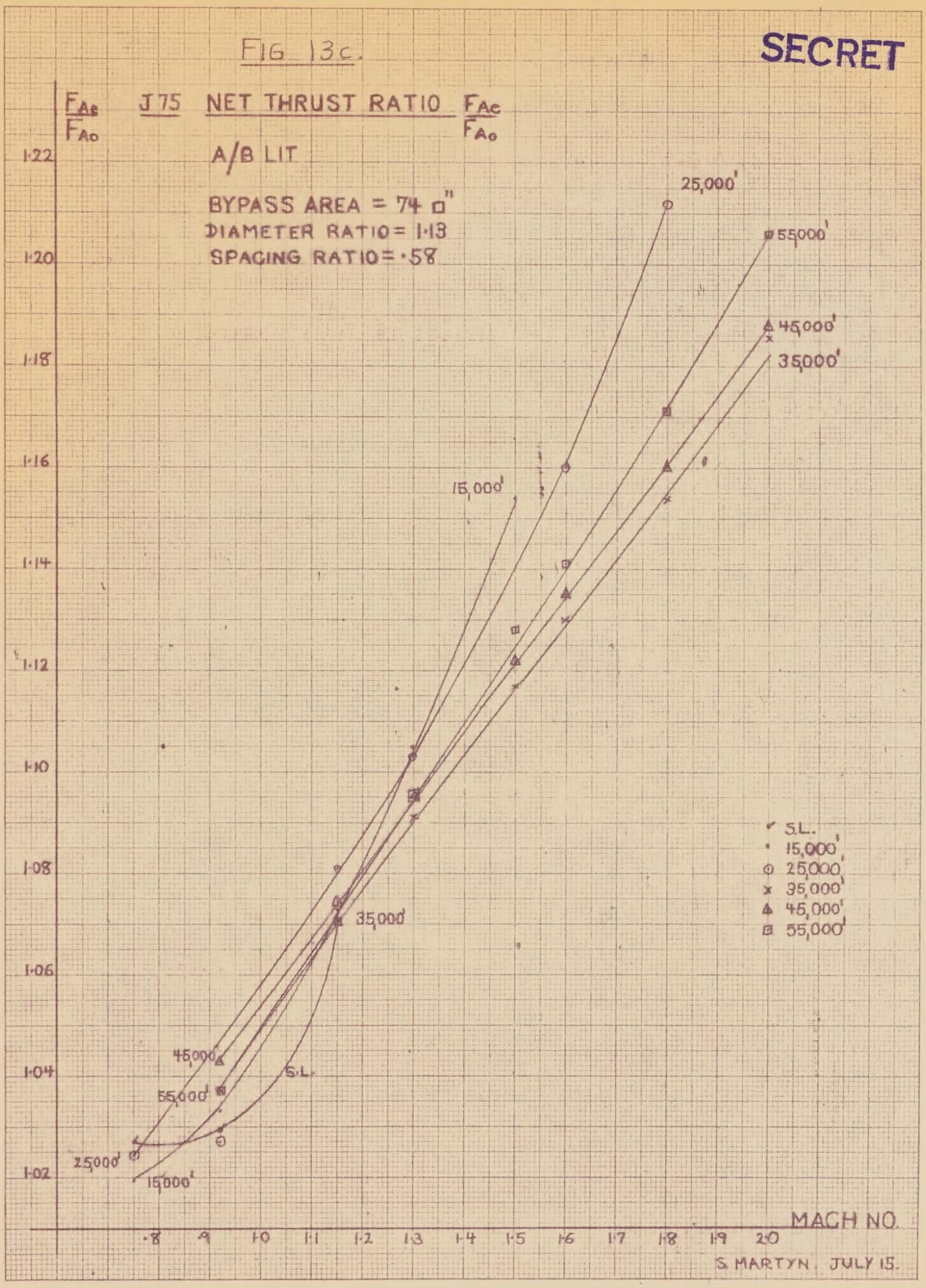
A/B LIT.

BYPASS AREA = 74 □"
EJECTOR :- SPACING RATIO = 1·13
DIAMETER RATIO = ·58

S. MARTYN.
JULY 19, 1955.

SECRET

SECRET



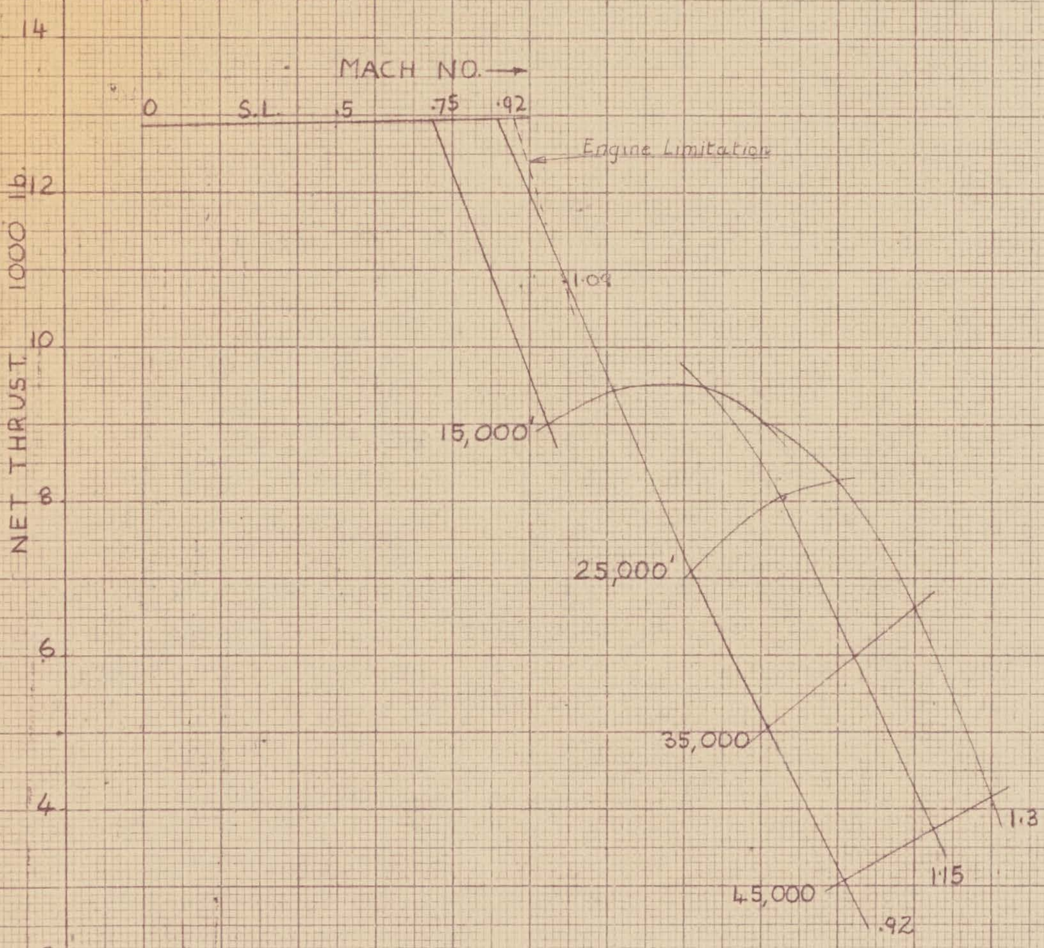

SECRET

SECRET

FIG. 14.

J75 (JT4A/25) NO A/B. MAX. RPM.

NET THRUST v MACH No. & ALTITUDE.



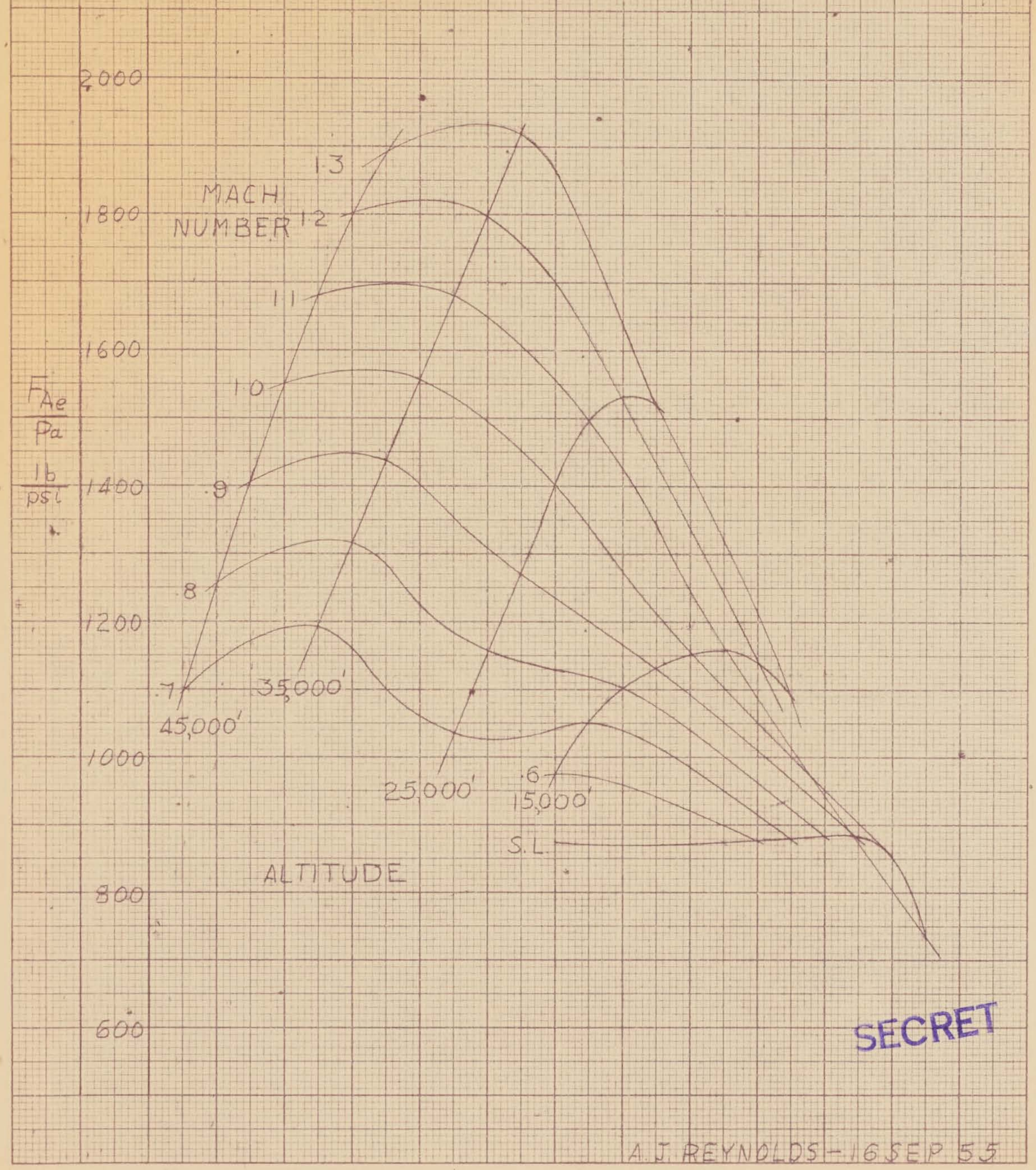

Issue 2
JRMonk.
Sept 14.

SECRET

J75 (JT4A-25) FIG. 14a.

# $\frac{F_{Ae}}{P_a}$ VS. MACH NUMBER & ALTITUDE

NO A/B MAX. RPM





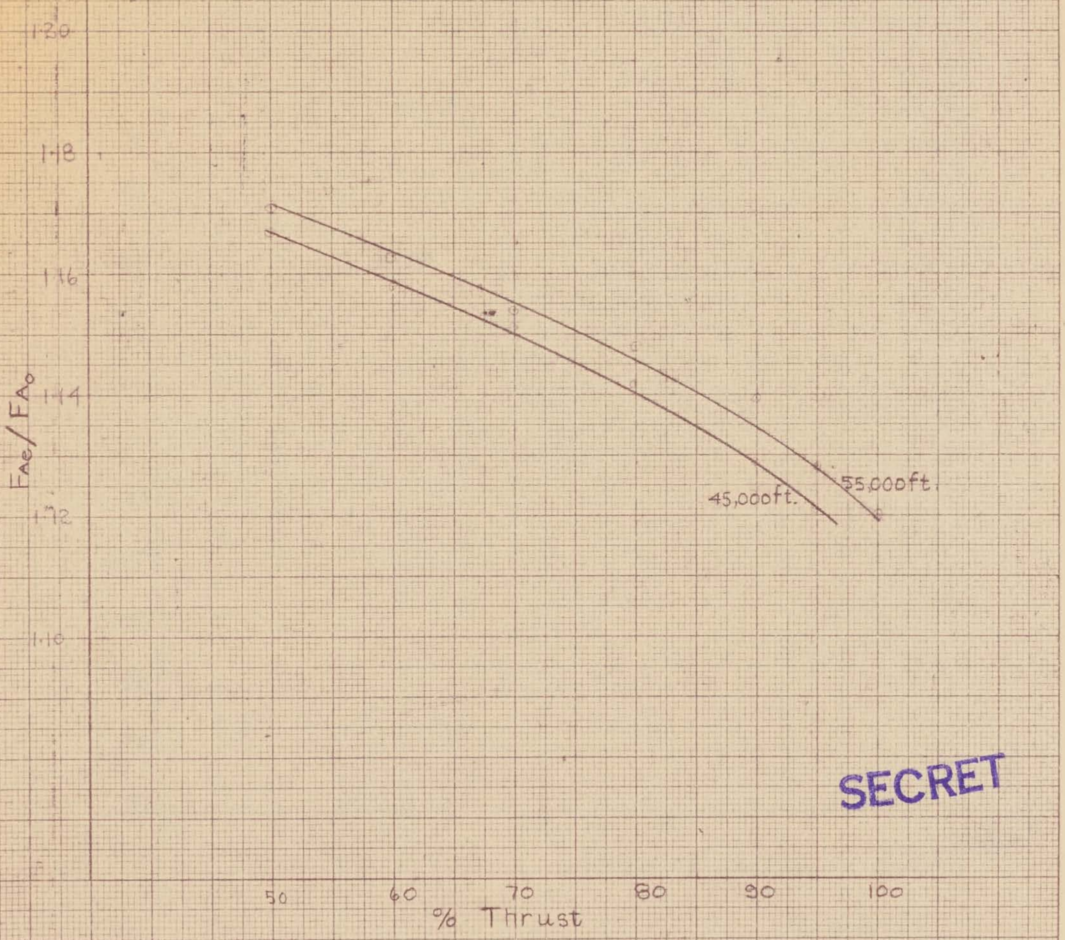

A.J. REYNOLDS - 16 SEP 55

SECRET

# FIG 15.

## Thrust Ratio vs % Thrust

Partial A/B M=1·5

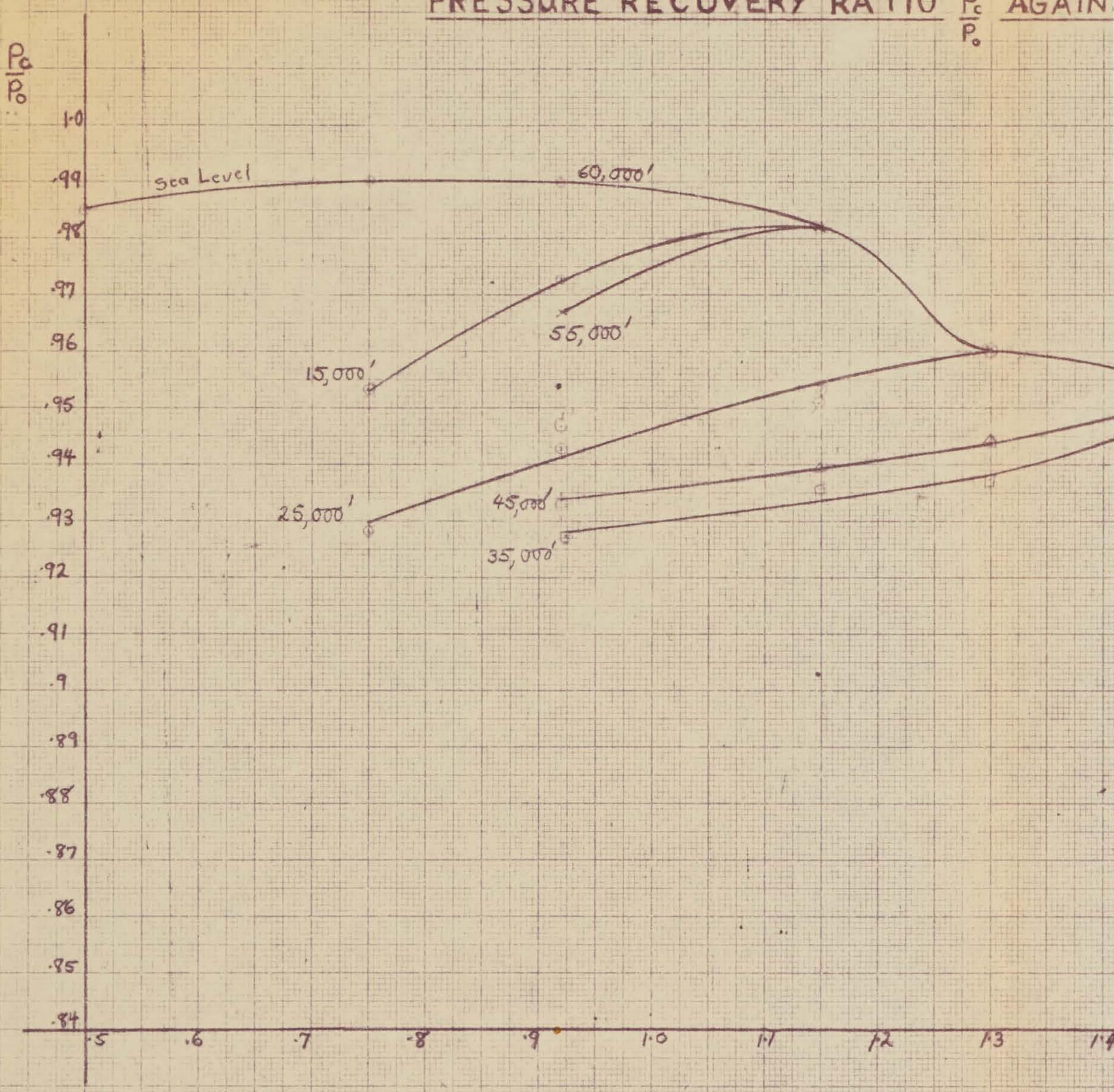



W. Grover
July 1955

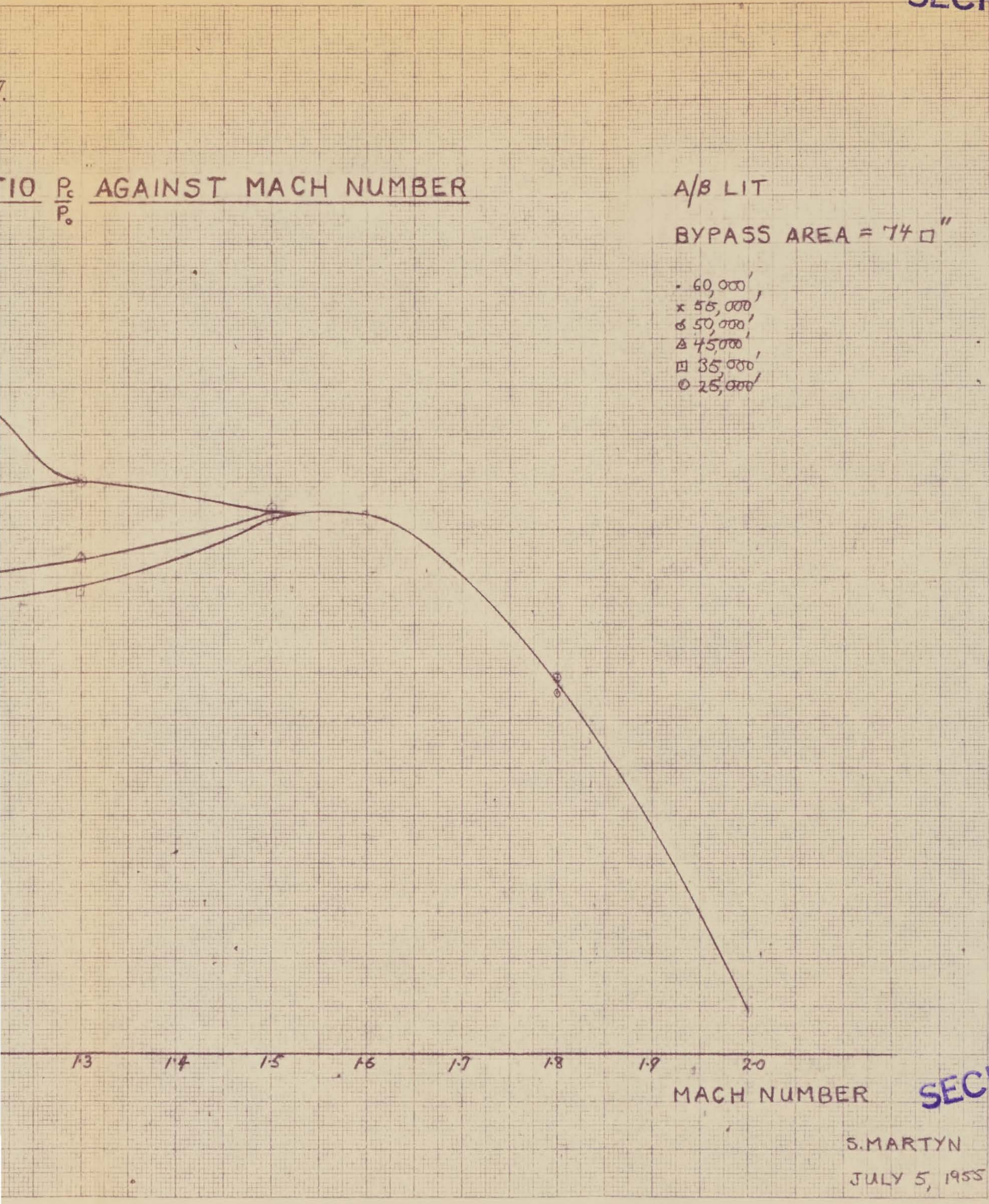
SECRET

FIG. 17.

PRESSURE RECOVERY RATIO $\frac{P_c}{P_o}$ AGAIN



SECRET



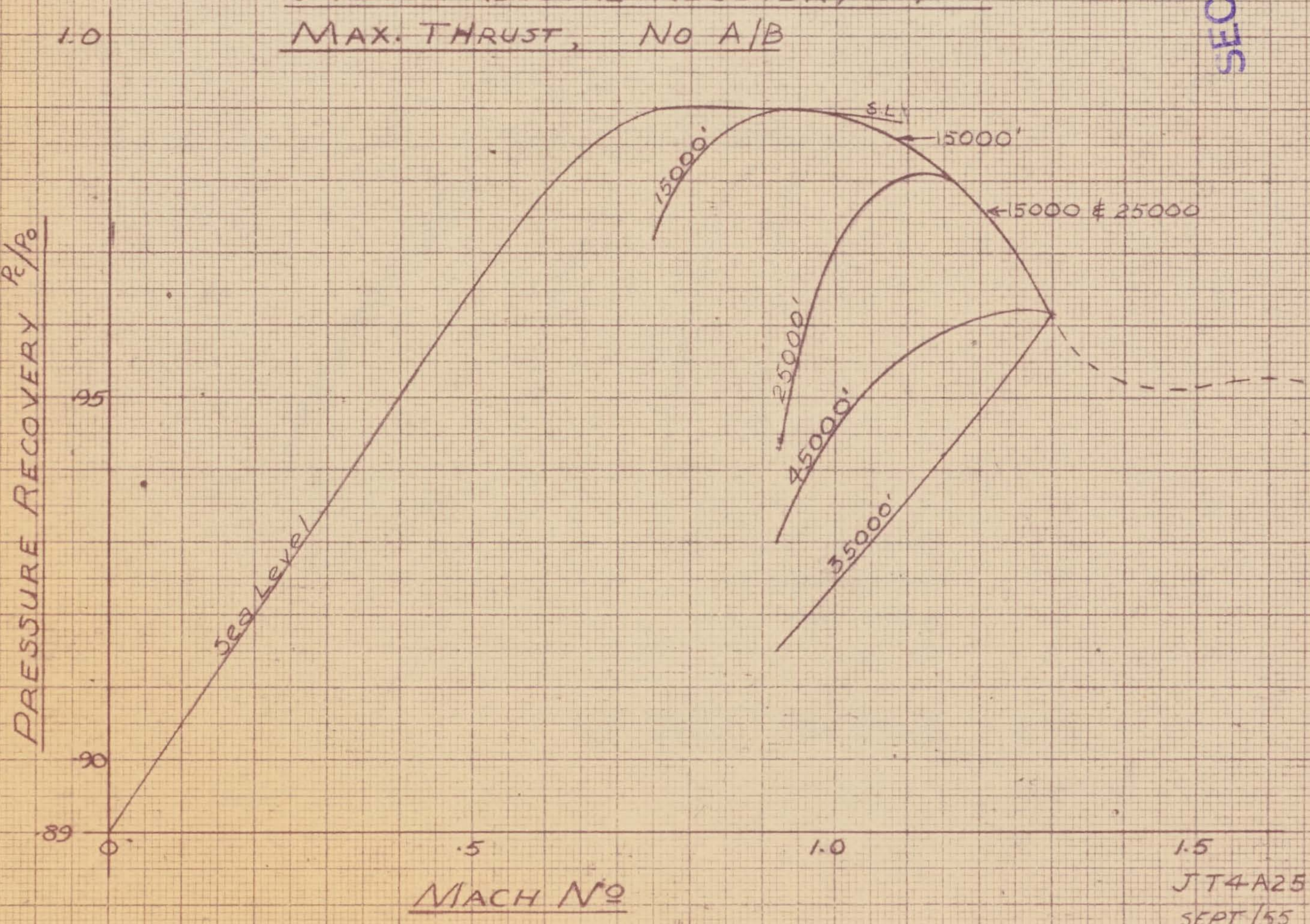

SECRET

SECRET

FIG 18

J75:- PRESSURE RECOVERY $P_c/P_o$

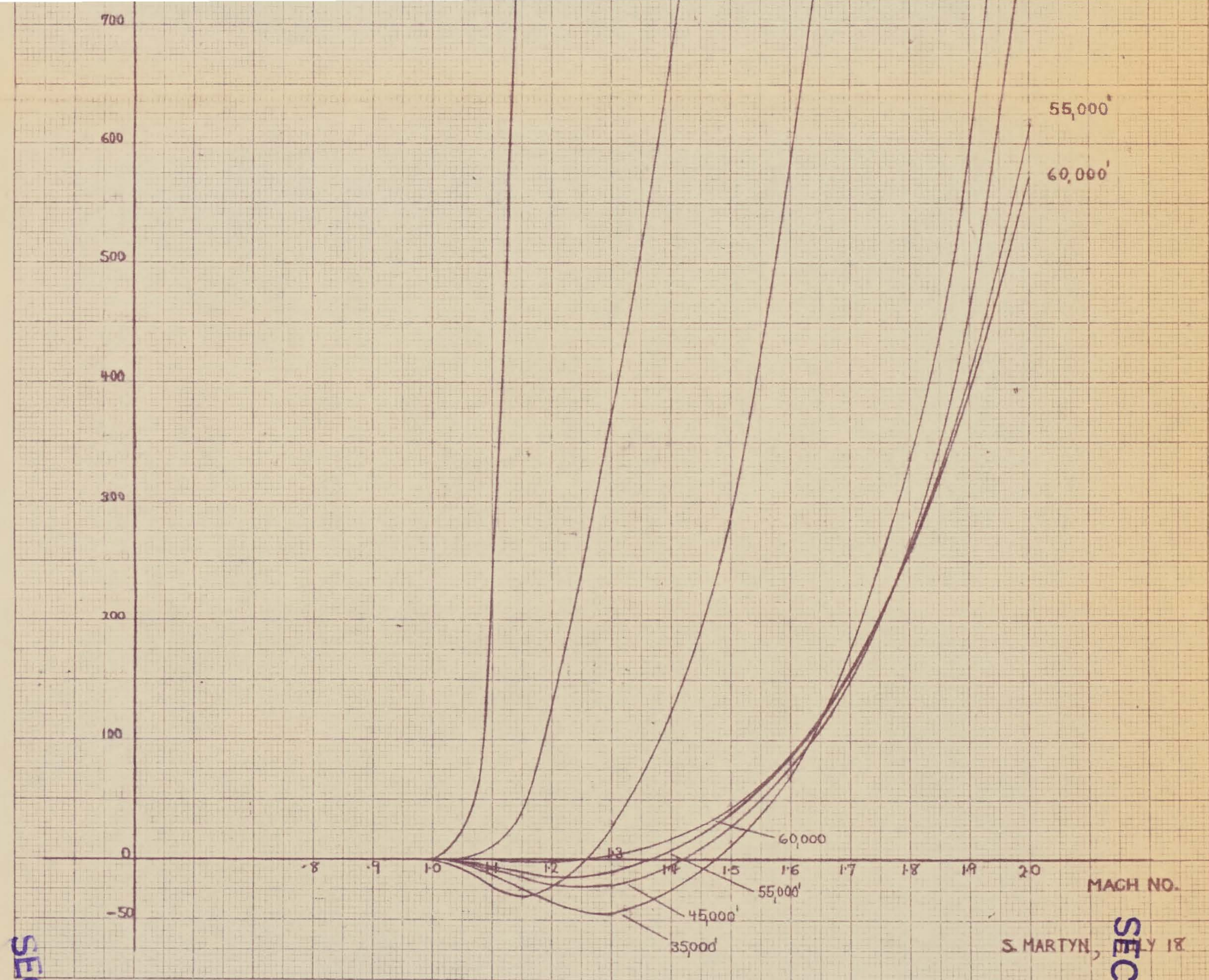
MAX. THRUST, NO A/B




JT4A25
SEPT/55
A SOKOL

SECRET

FIG 18.


700
600
500
400
300
200
100
0
-50
·8
·9
1·0
1·1
1·2
1·3
1·4
1·5
1·6
1·7
1·8
1·9
2·0
MACH NO.
55,000'
60,000'
60,000
55,000'
45,000'
35,000'
S. MARTYN,