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BY Rosemary L. Rovenstein 25 Jul 60

SYSTEM 606A

VTOL FIGHTER - BOMBER PROJECT

NOTES ON THE

# PRELIMINARY PERFORMANCE CALCULATIONS

AVRO/SPG/TR 276

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SPECIAL PROJECTS GROUP
Technical Report No. 276

SYSTEM 606A

# NOTES ON THE PRELIMINARY PERFORMANCE CALCULATIONS VTOL FIGHTER-BOMBER PROJECT

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Rosemary J. Lowenstine 25 Jul 60
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The number of pages in this report, including Security Warning, Title Page, and Table of Contents is 33.

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## 1.0 INTRODUCTION

These notes give the preliminary performance estimates that have been made for the System 606A V.T.O.L. fighter-bomber project and also a brief explanation of the drag, weight and engine data on which the estimates are based. (A pre-phase I proposal for wind tunnel tests and study has already been presented and the technical report, AVRO/SPG/TR255, on this weapon system study due under Contract AF33(30161) is being prepared.

A general arrangement drawing of the project is shown in Figure 21. It will be seen that it has a circular planform with a wing section which is symmetrical about the vertical centre line. The basic propulsion unit is a reverse flow turbojet engine in which the moving parts rotate about a vertical axis. The main units of the turbojet are a single sided, single entry centrifugal compressor, twelve radially spaced combustion cans and an axial flow tip turbine. The exhaust from the turbine is ducted radially outwards to a propulsive nozzle system situated around the wing periphery. This nozzle system is also used for control purposes, and to produce an annular jet downwards for vertical take-off. Six ramjet units are provided as a secondary propulsion system to improve the supersonic performance of the aeroplane. These units are situated in the fairing behind the main engine intake and are fed from a separate air intake which is partially closed in subsonic flight.

## 2.0 PRINCIPAL DIMENSIONS AND WEIGHTS

| | | |
|---|---|---|
| Span (diameter) | = | 28' 10" |
| Wing Area | = | 653 sq. ft. |
| Thickness-chord ratio at centre line | = | 3.47% |
| T.O.W. | = | 20,000 lb. |
| Fuel Weight | = | 7,500 lb. |

## 3.0 DERIVATION OF THE AEROPLANE DRAG

### (a) Subsonic and Transonic Minimum Drag ($C_{D_O}$)

The subsonic and transonic drag was derived from model test data reported in AVRO/SPG/TR186. Two models were tested, a double intake version (Model No. 1) and a single intake version (Model No. 2). The drag measurements of Model No. 2 were used as a starting point for the drag analysis since this model was roughly similar in configuration to the present Project. Figure 1 shows the results of these tests where -

Curve A is the minimum drag with a 'straight through duct' intake

Curve B is the minimum drag less the measured internal drag of the intake

Curve C is curve B plus a measured afterbody drag of an intake fairing.

In the present scheme six ramjets are situated in the fairing behind the main engine intake and when these are operating the afterbody drag will be considerably reduced. It is reasonable to expect therefore, that Curve B approximates most closely to the drag of the present Project.

However it will be noticed that this drag is about the same, or slightly less, than that measured for the basic wing (see Figure 2) over most of the Mach number range.

It is concluded that the actual drag curve (model scale) will be some where between curve A and curve B and a guess was made as to its probable variation with Mach number using the test results as a guide. A correction was then made for Reynolds Number effects to give values which apply to the full size aircraft at an altitude of 36,000 ft. and the curve was further modified to allow for the increase in drag due to the ramjet intake system. It was assumed that this increase would be of the order of 10% to give the final estimated drag curve shown in Figure 4.

### (b) Supersonic Minimum Drag ($C_{D_O}$)

The supersonic part of the drag curve given in Figure 4 was derived from model tests reported in AVRO/SPG/TR168. Figure 3 shows the $C_{D_O}$ versus M for the basic wing which has been obtained from the results of these tests with a correction for Reynolds Number to correspond to the full size aircraft at 36,000 ft. It was assumed that the total aeroplane $C_{D_O}$ followed the general shape of this curve so a constant drag increment was added to these values to account for the drag of the intakes etc. This increment was arranged so that the final drag curve 'tied in' with the transonic values.

## 3.0 DERIVATION OF THE AEROPLANE DRAG (continued)

### (c) Drag Due to Lift ($C_{Di}$)

From the test results on a subsonic model (see AVRO/SPG/TR12) it was possible to derive an approximate relation for $C_{Di}$ in terms of $C_L^2$ as follows -

$$C_{Di} = 0.312\ C_L^2$$

which was found to apply for a fairly wide range of jet coefficients.

The transonic and supersonic variation of $C_{Di}/C_L^2$ versus M is shown in Figure 5 which has been taken from AVRO/SPG/TR168. (A slight modification was made to these values to apply to a typical jet coefficient $C_j = 0.1$ for use in performance calculations).

## 4.0 ENGINE DATA

The propulsion unit is essentially a dual system consisting of two sections, the principal features of which are as follows -

### (a) The Primary Propulsion Unit

Is a simple centrifugal flow turbo-jet engine with a pressure ratio of 4:1 and a maximum cycle temperature of 1150°K (1611°F). The design mass flow is 308 lb./sec., giving a sea-level static thrust of 17,000 lb.

### (b) The Secondary Propulsion

Consists of six ramjets with a total combustion chamber inlet area of 12 sq. ft. and a nozzle outlet area of 8 sq. ft. The propelling nozzles are assumed of the fixed area convergent type and the operating combustion temperature is 1750°K (2691°F).

The estimated performance of the primary propulsion unit is shown in Figures 7 and 8 where net thrust/ambient static pressure and fuel flow/ambient static pressure versus Mach Number are given for three engine ratings. Figure 9 shows gross thrust at sea-level static conditions versus combustion chamber temperature. The estimated performance of the secondary propulsion unit is shown in Figure 10 and 11 where net thrust/ambient static pressure and fuel flow/ambient static pressure versus Mach Numbers are given for 1750°K combustion chamber temperature.

4.0 ENGINE DATA (continued)

In estimating the performance of the propulsion system it was assumed that,

(i) Since the primary propulsion unit is a simple centrifugal flow turbojet engine, an accurate estimate of performance could be obtained by scaling a typical turbojet engine of the same pressure ratio and maximum cycle temperature, the thrust and mass flow being related only to the square of the linear dimension. For this purpose data on the Rolls-Royce "Nene" was used in conjunction with the intake pressure recovery curves shown in Figure 6.

The higher pressure recovery is somewhat better than that achieved in tests of a 1/12 scale model reported in AVRO/SPC/TR137 (D.D. No. 58RDZ-16142). The improvement assumed is expected to result from eliminating the sweepback at the sides (Fig. 45 in the above report), particularly in view of other tests results of similar configurations, e.g. NACA RM L.52.J.02 and L.5[illegible].J.07.

The lower pressure recovery assumed a pitot type intake. Total thrust for the dual system is shown in Figure 13 for both assumptions. Since the ramjet system is providing the majority of the net thrust for supersonic operation the simpler pitot intake is probably preferable for the centrifugal engine and is,found to cost about 5000 ft. of ceiling and some 7% supersonic range. Performance given is however based on the higher assumption.

For the internal loss in turning into the central sink, the initial diffusion is about 35%. Assuming sonic flow in the mouth which represents the worst condition and the loss factor of 0.2q shown in the tests reported in AVRO/SPC/TR177 (E.D. No. 58RDZ-16143), a loss of 3½% of total head results. 4% has been used to allow for the initial diffusion. In the static case the intake is assumed to be relieved to prevent a greater loss and (depending on the design) the assumption may prove too severe for this case. Suitable areas for preliminary design were determined from a static design point calculation as follows:

TABLE I ASSUMED EFFICIENCIES

| | |
|---|---|
| Air Intake Pressure Recovery $P_2/P_1$ (internal) | 0.96 |
| Combustion Chamber Pressure Recovery $P_4/P_3$ | 0.96 |
| Outlet Duct and Nozzle Recovery $P_6/P_5$ | 0.95 |
| Compressor Efficiency | 0.78 |
| Turbine Efficiency | 0.85 |
| Compressor Power Input Factor | 1.035 |
| Compressor Slip Factor | 0.935 |

## 4.0 ENGINE DATA (continued)

TABLE II DESIGN VALUES

| | | |
|---|---|---|
| Design Air Mass Flow | 307 | lb./sec. |
| Design Compressor Pressure Ratio | 4:1 | |
| Temperature Rise | 179.3 | °C |
| Compressor Tip Speed | 1465 | ft./sec. |
| Axial Mach No. (Compressor Inlet) | 0.485 | |
| Inlet Annulus Area | 1266 | sq. in. |
| Radial Outlet Mach No. | 0.1485 | |
| Compressor Outlet Annulus Area | 1160 | sq. in. |
| Combustion Chamber Temperature | 1150 | °K |
| Turbine Temperature Drop | 179.3 | °C |
| Turbine Pressure Ratio | 2.015 | |
| Turbine Radial Outlet Mach No. | 0.5245 | |
| Total Wing Duct Area | 1230 | sq. in. |
| Wing Duct Mach No. | 0.524 | |
| Effective Nozzle Outlet Area | 976.3 | |
| Nozzle Pressure Ratio | 1.74 | |

It will be noted from Figure 9 that blade cooling, for which the design is well suited, could permit an increase in combustion chamber temperature and thus increase the thrust for take-off of the primary unit.

(ii) The pressure loss factor for the secondary combustion chamber was given by,

$$\frac{P_2 - P_3}{q_2} = 3.0 + 1.5\left[\frac{T_3}{T_2} - 1\right]$$

where $P_2$ and $T_2$ are the inlet total pressure and temperature respectively, $P_3$ and $T_3$, are the outlet total pressure and temperature respectively, and $q_2$ is the dynamic pressure at the inlet to the combustion chamber.

This assumption results in the total pressure loss varying from 87% to 84% and entry Mach No. varying from 0.126 to 0.200 as shown by the graph of Figure 12, and is believed to be realistic for a ramjet system in view of NACA RM E51 D11 and NGTE M144 for example.

Combustion chamber Mach No. was iterated to obtain matched conditions with fixed outlet area and a variable throat inlet also capable of the higher pressure recovery of Figure 6 was assumed, together with an internal diffuser loss of 3% of the total pressure.

## 5.0 WEIGHT ESTIMATE

Details of the estimated component weights are given in the following groups.

(1) Structure
(2) Landing Gear
(3) Power Plant
(4) Fixed Services
(5) Useful Load

| | | |
|---|---|---|
| **Group 1 - Structure** | | |
| Skins and Stiffening | | 1,540 |
| Ribs | | 453 |
| Upper structure | | 733 |
| Tip manifold and vanes | | 219 |
| Cockpit | | 311 |
| Contingency | | 144 |
| | TOTAL | 3,470 |
| **Group 2 - Landing Gear** | | |
| Total Landing Gear | | 250 |
| **Group 3 - Power Plant** | | |
| Fuel System | | 700 |
| Insulation | | 200 |
| Centrifugal engine & controls | | 3,060 |
| Ramjets | | 515 |
| Jet pipes | | 365 |
| | TOTAL | 4,840 |
| **Group 4 - Fixed Services** | | |
| Flying controls | | 199 |
| Pneumatics | | 67 |
| Fire protection | | 72 |
| Instruments | | 56 |
| Electrics | | 580 |
| Furnishing air conditioning & emergency | | 480 |
| Fire control navigation & communication | | 936 |
| Bomb release | | 150 |
| | TOTAL | 2,540 |
| **Group 5 - Useful Load** | | |
| Pilot | | 220 |
| Fuel | | 7,500 |
| Payload (armament) | | 1,000 |
| Oil, residual fuel etc. | | 180 |
| | TOTAL | 8,900 |

Total weights may be summarized as follows:

| | | |
|---|---|---|
| Airframe Structure | | 3,470 |
| Landing Gear | | 250 |
| Power Plant | | 4,840 |
| Fixed Services | | 2,540 |
| | WEIGHT EMPTY | 11,100 |

5.0 WEIGHT ESTIMATE (continued)

| | |
|---|---|
| Useful Load | 8,900 |
| Gross Weight | 20,000 |
| Useful Fuel | 7,500 |
| Gross Weight less Useful Fuel | 12,500 |
| Armament | 1,000 |
| Gross Weight less Useful Fuel & Armament | 11,500 |

The basic material for the aircraft Structure is assumed to be steel although some use may be made of titanium and light alloys wherever possible. The simple, radial structure and lack of protuberances presents no unusual design problems.

The most critical area is expected to be the mechanical design of the turbine blades, blade attachments, and rotor spokes. A preliminary investigation of this area has been performed in AVRO/SPG/TR275 and is submitted in conjunction with this report. The investigation shows that, although care must be exercised in the proportioning of the various components, sufficient latitude exists within the stressing limitations of the materials available for a sound design without exceeding the weight allowance specified above.

In estimating the above weights a synthetic estimate was first made making wide use of .031" steel for skins and similar associated structure and bringing out a weight of 3,226 lb. for the complete aircraft. This was compared with the outer wing of PV.704 for which the estimate of 3,631 lb. given in AVRO/SPG/TR71, Issue 3 (I.D. No. 58RDZ-11196) represented a complete detail design for an all steel structure. The weight/sq.ft. of this structure was 6.35 lb. However it included a large amount of detail structure for the outer wing combustion system. A 15% improvement on this to a specific figure of 5.35 lb./sq.ft. is considered not unreasonable and this involved adding a 7.5% contingency to the synthetic figure.

With regard to the power plant the synthetic estimate brought out a figure of 2,640 lb. for the centrifugal engine, with an all steel rotor weight of 900 lb. estimated on the basis of the preliminary stressing reported in TR275. This was considered low and was increased by a 16% contingency to 3,060 lb. to give a specific weight of 0.18 which was considered reasonable in view of the evident weight saving from the airframe engine integration.

Fixed services at 2,540 lb. is not very much greater than on PV.704 (2,310 lb.) largely due to the very large weight saving on controls, for which the present estimate was based on the Avrocar; the relevant weights being adjusted to the new design and perimeter. In electrics a basic

5.0 WEIGHT ESTIMATE (continued)

power supply of two 12 KVA alternators was assumed adequate to drive the radar and all other services and the system synthesised having regard to percentage weights for similar systems. Fire control, navigation and communication was based on the General Electric Company's 'Bantam' system.

6.0 PERFORMANCE ESTIMATES

The drag versus Mach No. has been plotted in 'non-dimensional' form in Figure 13 together with the thrust curves. The maximum speed versus altitude as derived from these plots is shown in Figure 14 for an aeroplane weight of 15,250 lb. Two basic range profiles have been calculated and are summarized in Table I (which is for a supersonic cruise at approximately 80,000 ft.) and Table II (Which is for a subsonic cruise at 50,000 ft.).

Plots of acceleration and climb performance are shown in Figures 15 to 19.

Finally, Figure 20 illustrates some typical mission profiles.

7.0 LAMINAR FLOW WING

The all wing design and layout of the propulsion system of the Project make it suitable for attempts at boundary layer control by surface suction. It is shown in AVRO/SPG/TR269, that if a fully laminar boundary layer could be achieved, the maximum L/D would be of the order of 20. A bombing mission profile assuming that an L/D of this magnitude is possible is shown in Figure 20.

TABLE I

Range Summary (Supersonic Cruise)

T.O.W. 20,000 lb. Fuel available = 7,500 lb.

| Phase | Engine Conditions | Fuel Used lb. | Time Taken min. | Distance Gone N.A.M. |
|---|---|---|---|---|
| Take-off and Transition | Max. rpm | 290 | 1.0 | 0 |
| S.L. Acceleration to M = 0.9 | Max. rpm No Ramjet | 230 | 0.7 | 5 |
| Subsonic Climb at M = 0.9 Sea Level to 36,090 ft. | Max. rpm No Ramjet | 400 | 1.7 | 14 |
| Transonic Supersonic Acc'tn. M = 0.9 to 2.3 | Max. rpm With Ramjet | 1118 | 2.1 | 30 |
| L/W = 2 turn into Supersonic Climb M = 2.3 to M = 2.5 completing turn at 45,000 ft. | Max. rpm With Ramjet | 430 | 0.4 | 7 |
| Supersonic Climb at M = 2.5 45,000 ft. to 80,000 ft. | Max. rpm With Ramjet | 525 | 1.7 | 38.0 |
| Cruise at M = 2.5 | 95%Max. rpm With Ramjet | 3757 | 29.8 | 712 |
| Descent and Landing | — | 750 | — | — |
| | TOTALS | 7500 | 37.4 | 806 |

TABLE II

Range Summary (Subsonic Cruise)

T.O.W. 20,000 lb. Fuel available = 7,500 lb.

| Phase | Engine Conditions | Fuel Used lb. | Time Taken min. | Distance Gone N.A.M. |
|---|---|---|---|---|
| Take-off and Transition | Max. rpm | 290 | 1.0 | 0 |
| S.L. Acceleration to M = 0.9 | Max. rpm No Ramjet | 230 | 0.7 | 5 |
| Subsonic Climb at M = 0.9 S.L. to 50,000 ft. | Max. rpm No Ramjet | 620 | 4.0 | 32 |
| Subsonic Cruise at M = 0.92 | 90%Max. rpm No Ramjet | 5610 | 152 | 1338 |
| Descent and Landing (10% reserve fuel) | — | 750 | — | — |
| | TOTALS | 7500 | 157.7 | 1375 |

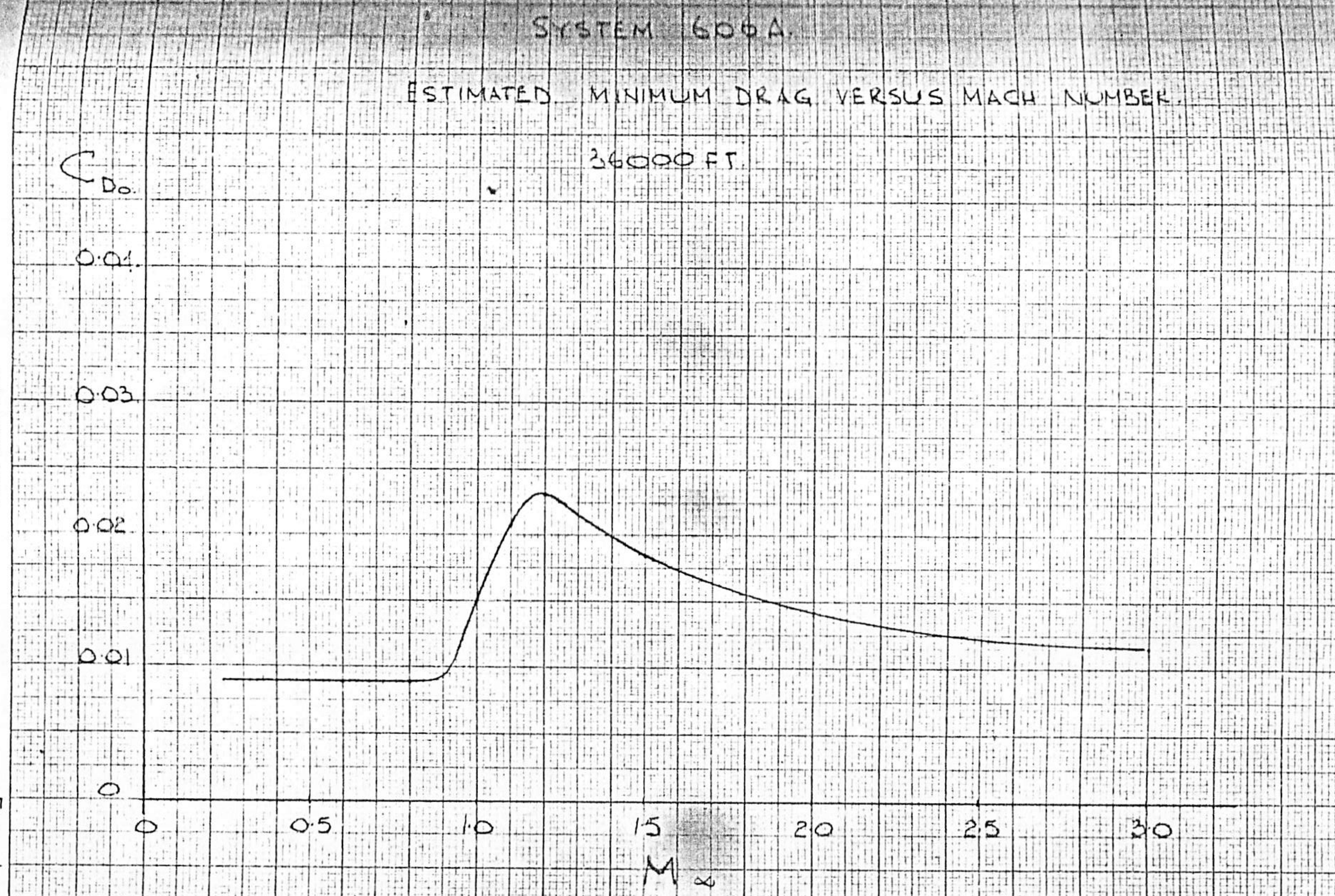


Fig 4.

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AVRO/SPG/TR 276

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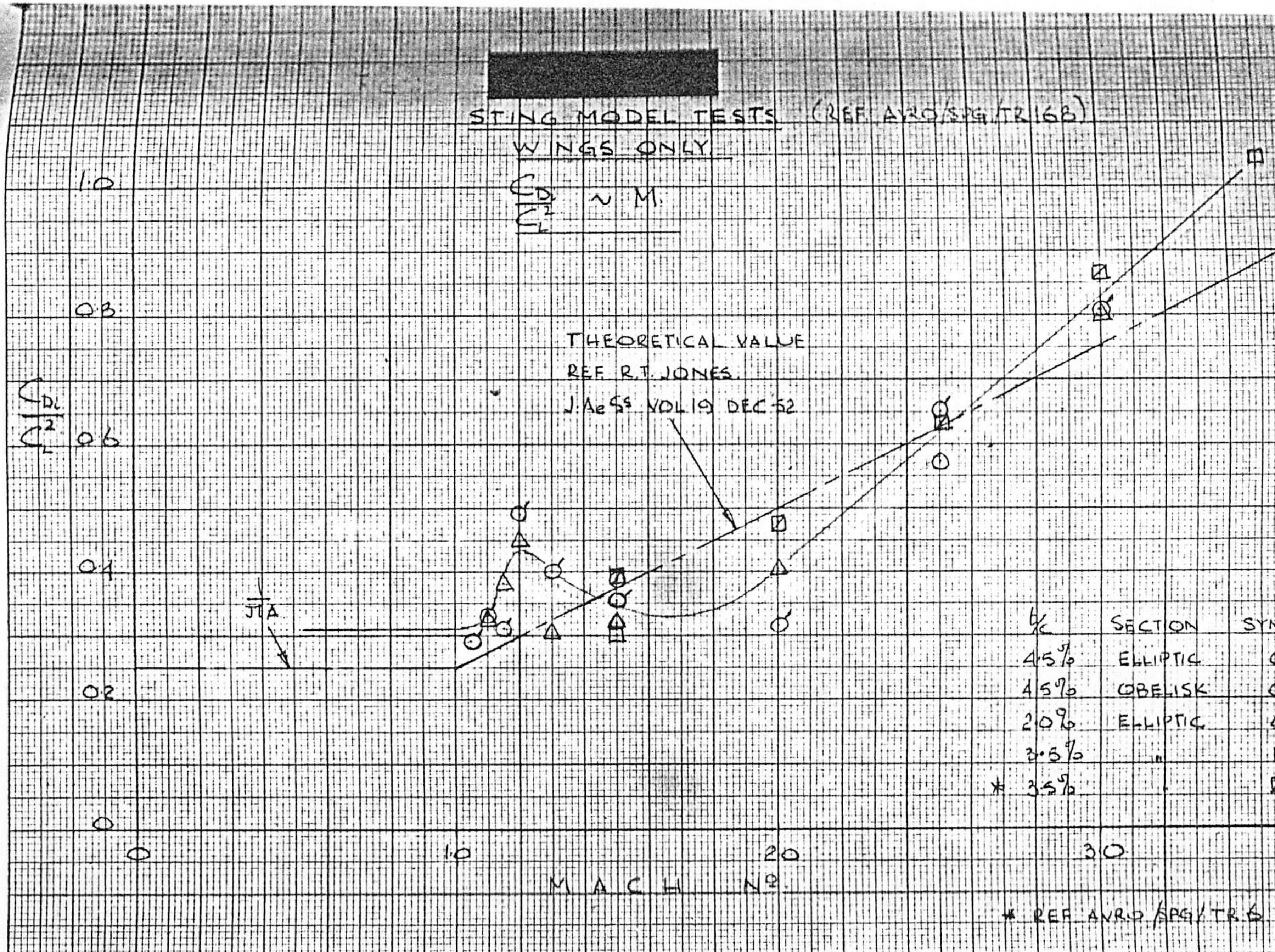




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FIG 5

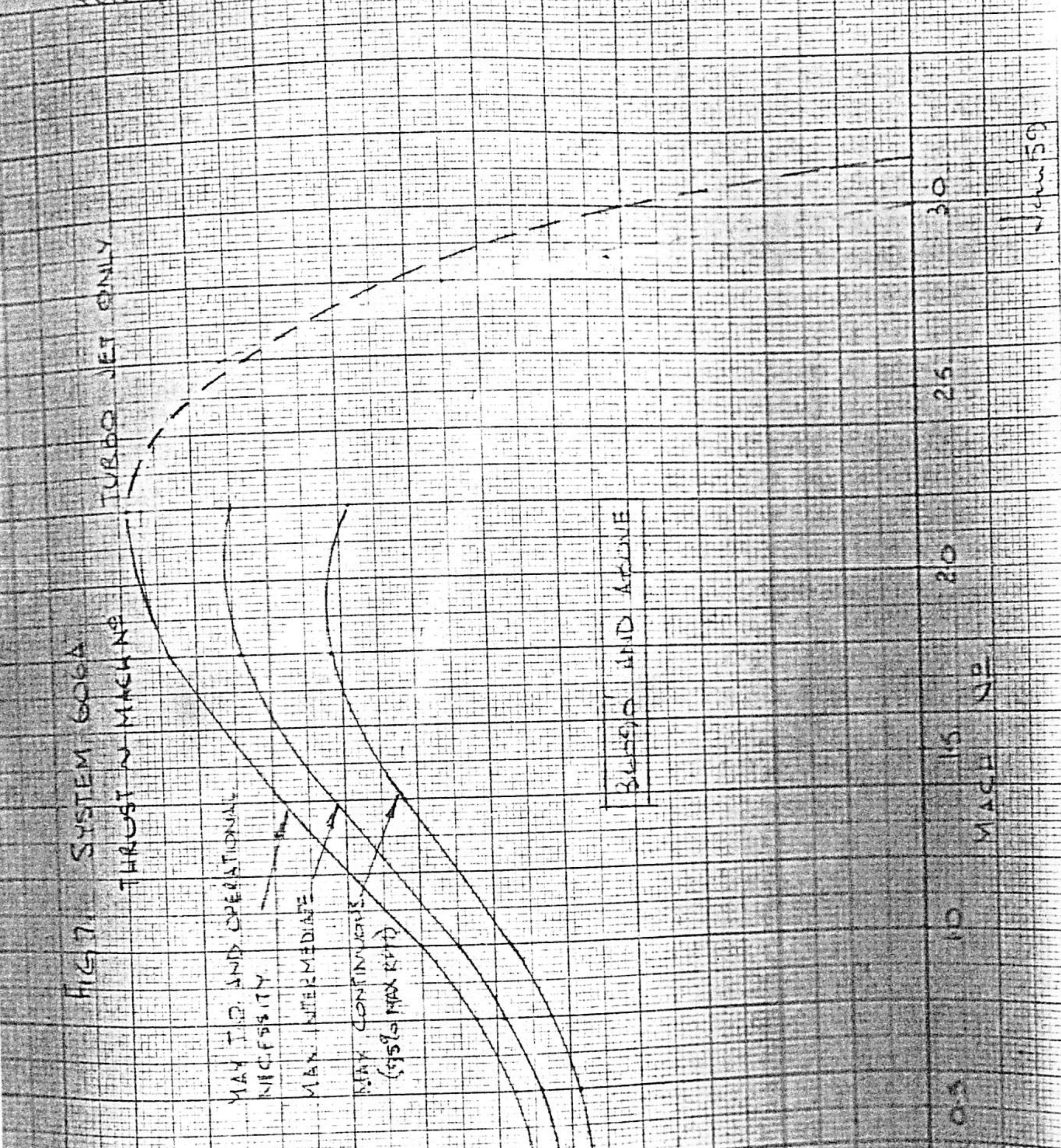
FIG 7. SYSTEM 606A
THRUST ~ MACH No TURBO JET ONLY
MAX T.O. AND OPERATIONAL NECESSITY
MAX INTERMEDIATE
MAX CONTINUOUS (95% MAX RPM)
BLEED AND ACTIVE
$X_N / P_A$
lb/sq ft
20
15
10
5
0
0.5
10
15
20
25
30
MACH No

Fig 7

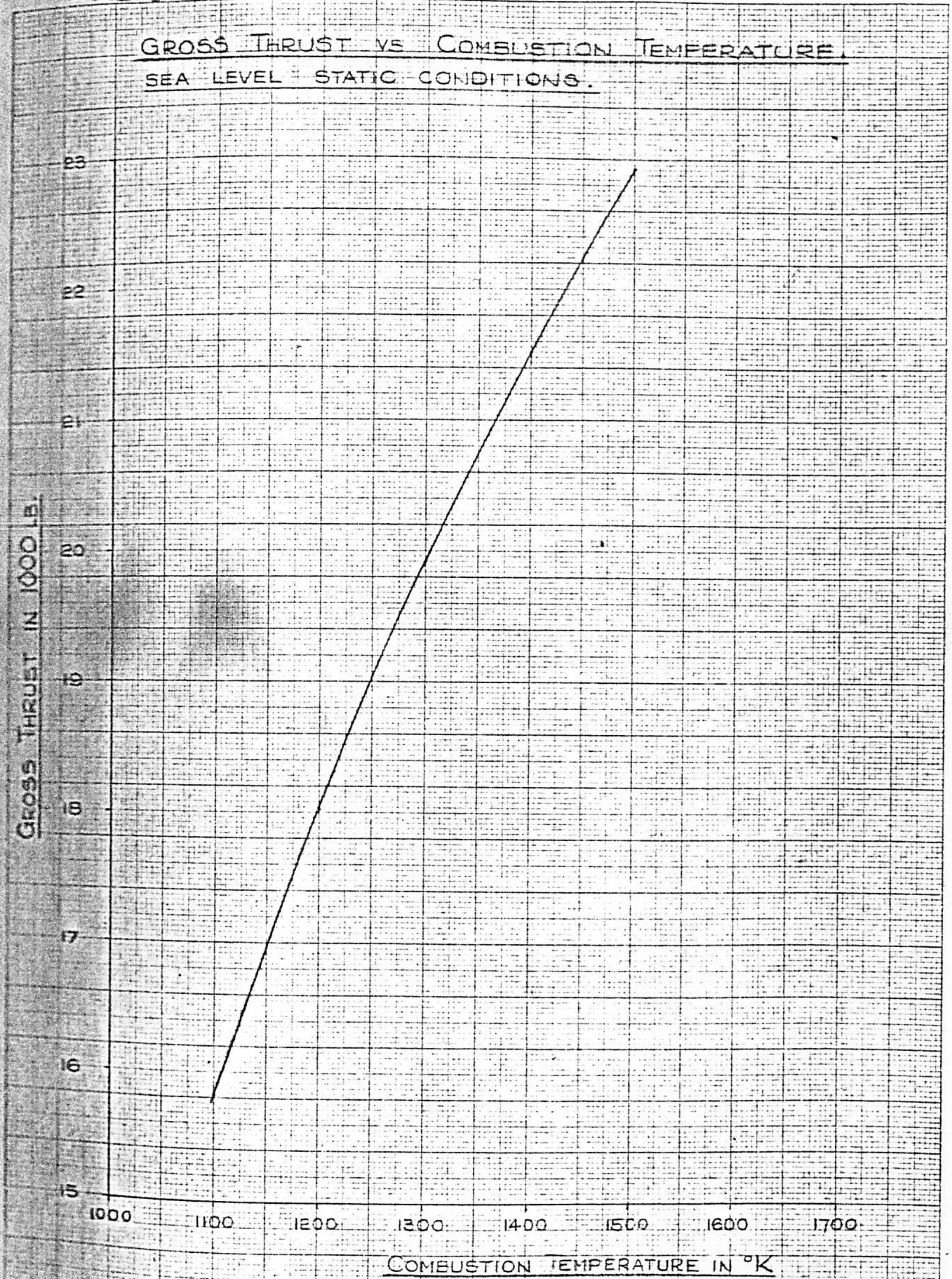
GROSS THRUST VS COMBUSTION TEMPERATURE.
SEA LEVEL STATIC CONDITIONS.
GROSS THRUST IN 1000 LB.
15
16
17
18
19
20
21
22
23
1000
1100
1200
1300
1400
1500
1600
1700
COMBUSTION TEMPERATURE IN °K

FIG. 9

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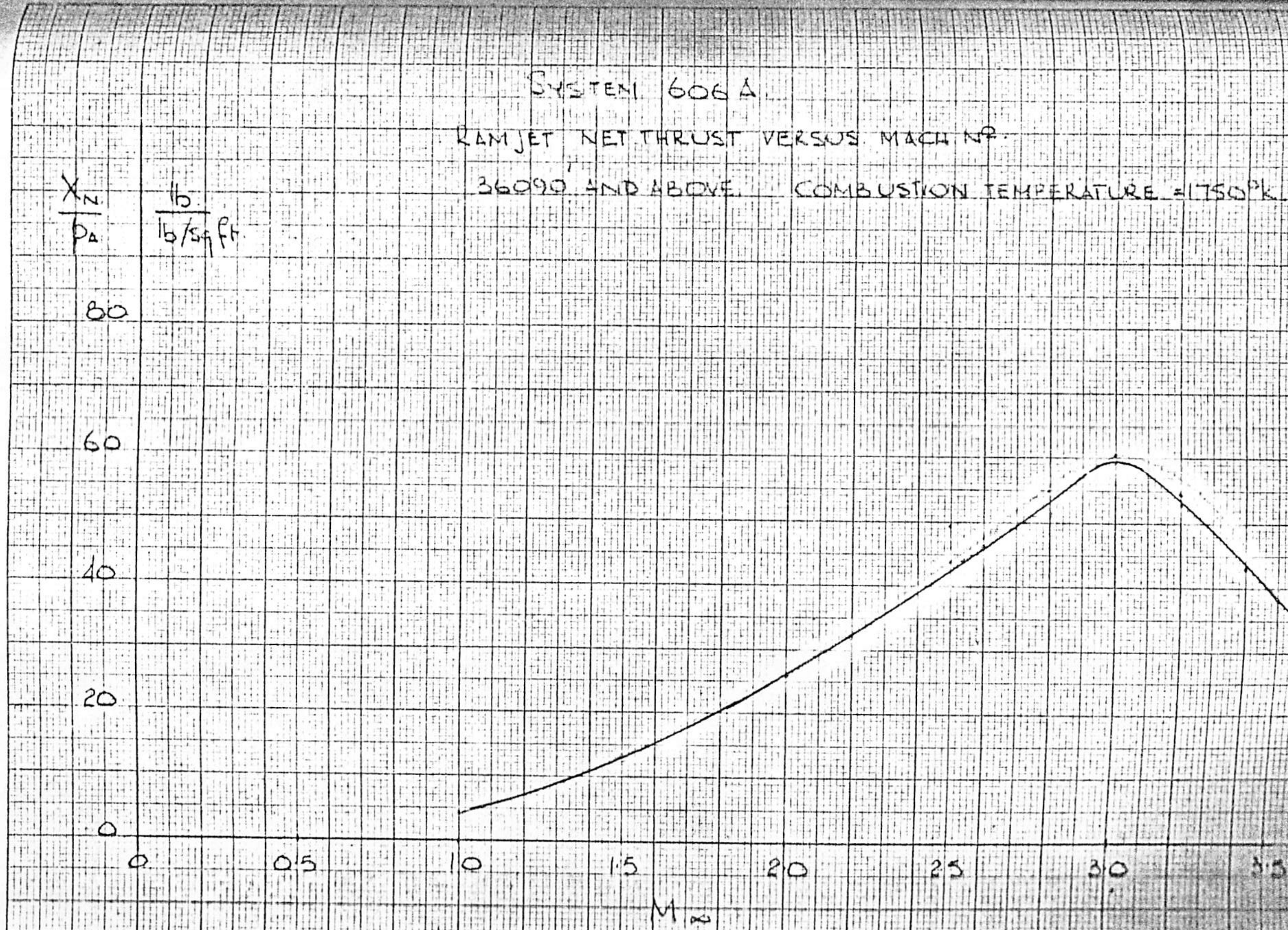


FIG 10



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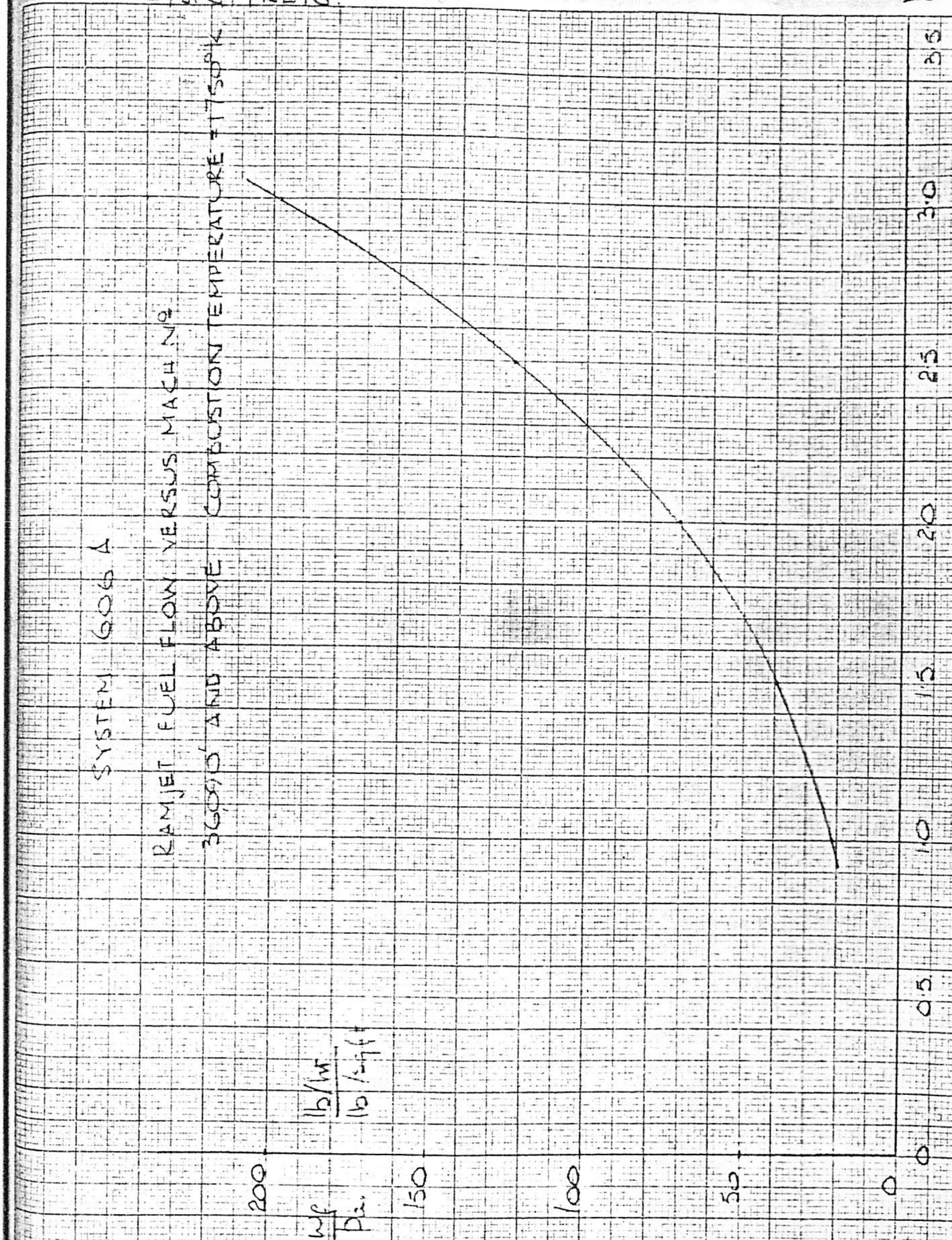
SYSTEM 606 A
RAMJET FUEL FLOW VERSUS MACH No
36000' AND ABOVE COMBUSTION TEMPERATURE = 1750°K
Wf/D2
lb/hr / lb/sq ft
200
150
100
50
0
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5



FIG 11.

AVRO/SPG/TR 276

## SYSTEM 606A

ALTITUDE VERSUS MACH N°

W = 16250 lbs

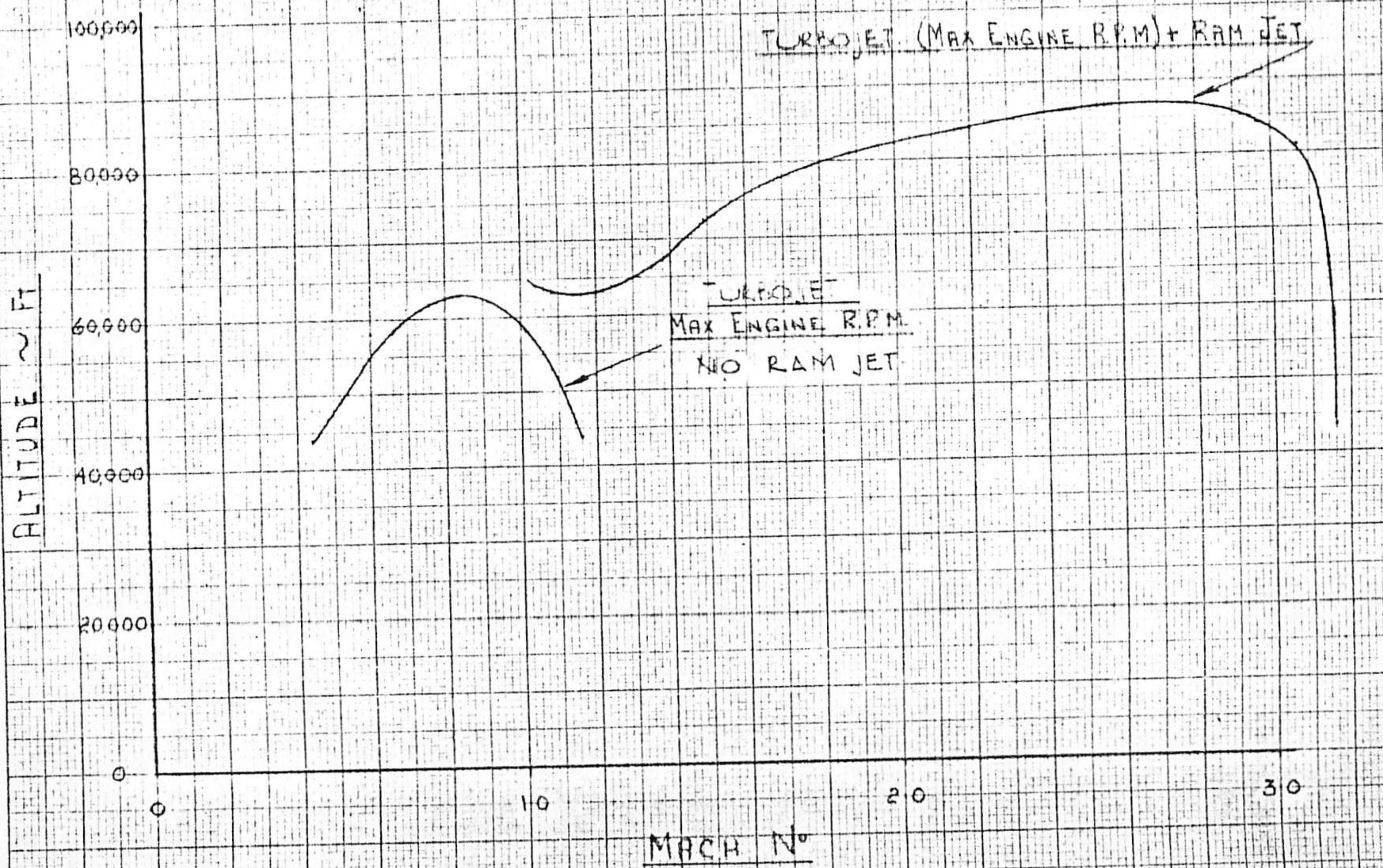


FIG 14

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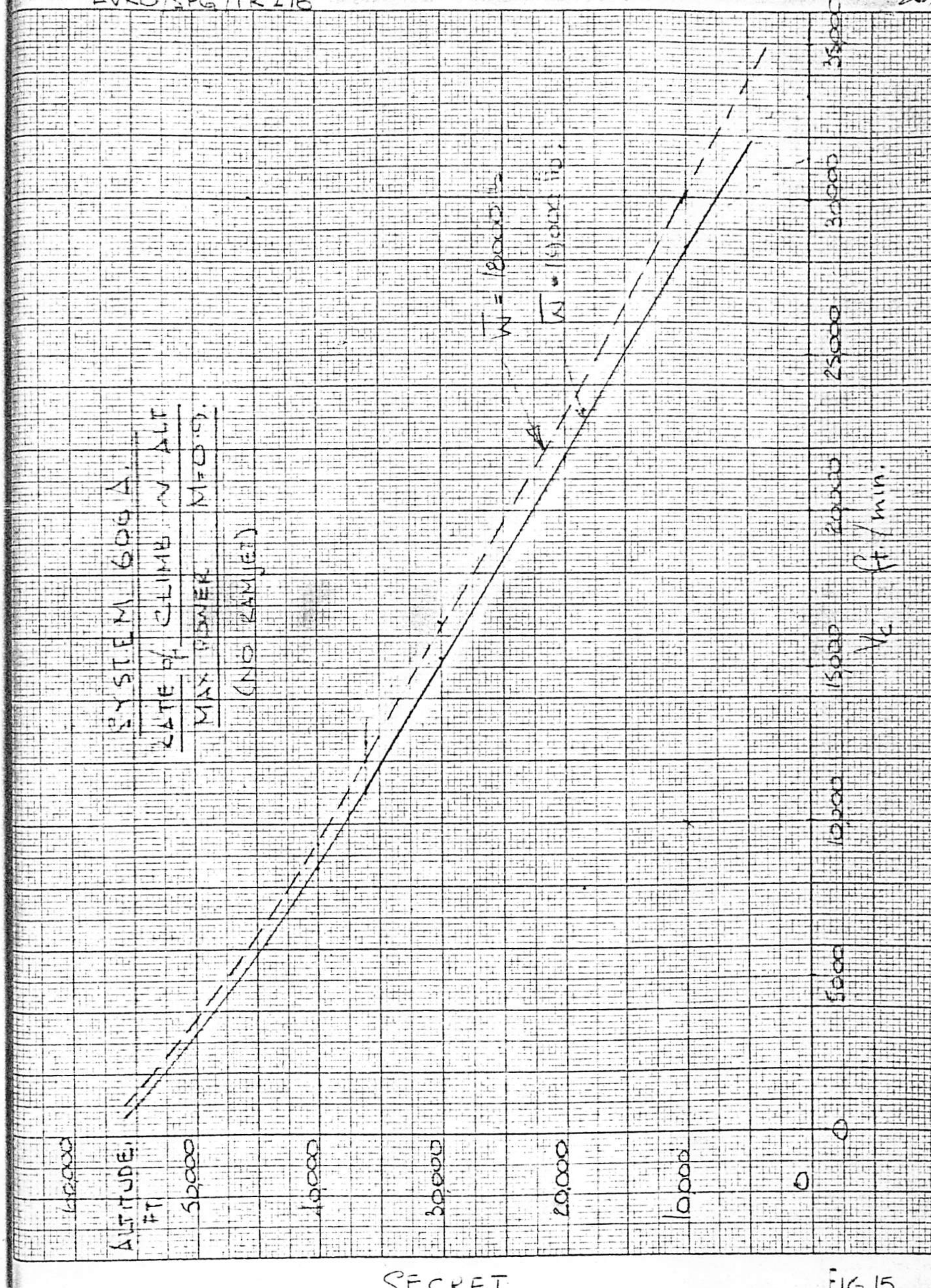
SYSTEM 606 A.
RATE OF CLIMB ~ ALT.
MAX POWER M=0.9.
(NO RAMJET)
W = 18000 lb
W = 19000 lb.
ALTITUDE. FT
60,000
50,000
40,000
30,000
20,000
10,000
0
0
5000
10,000
15000
20000
25000
30000
35000
Vc ft/min.



FIG 15

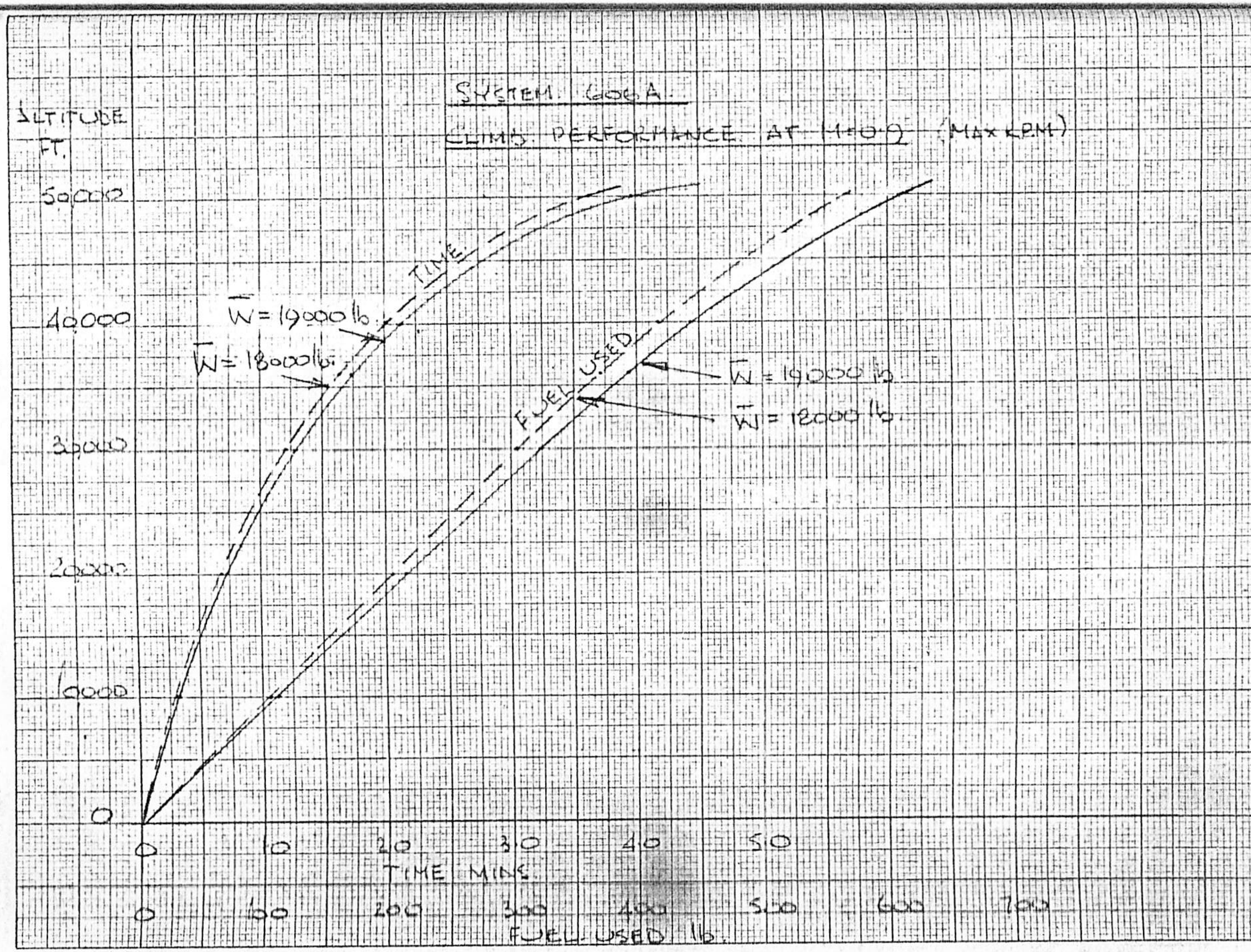
SYSTEM 606A
CLIMB PERFORMANCE AT M=0.9 (MAX R.P.M)
ALTITUDE FT.
50,000
40,000
30,000
20,000
10,000
0
TIME
$\overline{W}$ = 19000 lb.
$\overline{W}$ = 18000 lb.
FUEL USED
$\overline{W}$ = 19000 lb
$\overline{W}$ = 18000 lb.
0
10
20
30
40
50
TIME MINS.
0
100
200
300
400
500
600
700
FUEL USED lb.

Fig. 16

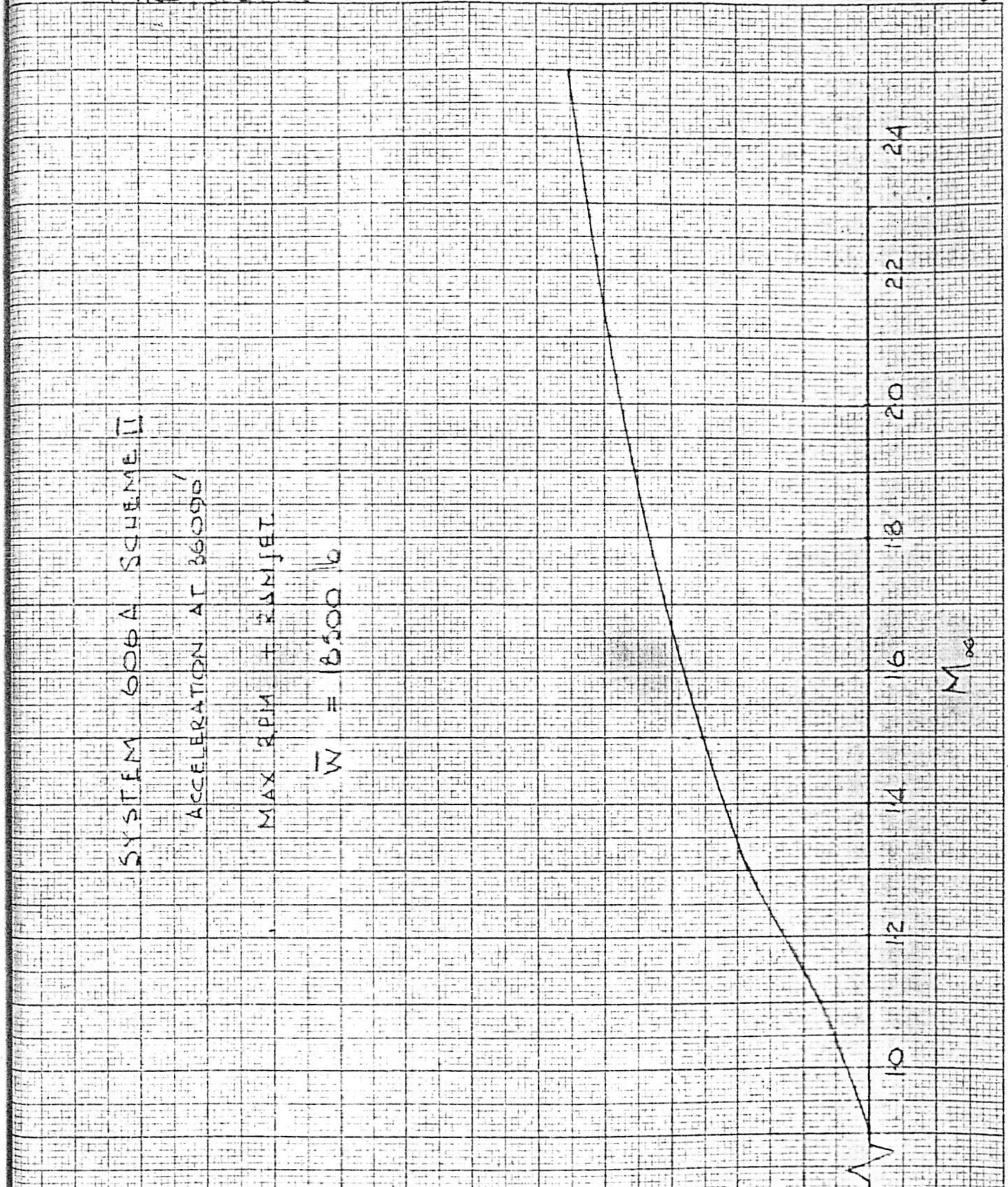
SYSTEM 606A SCHEME II
ACCELERATION AT 36090'
MAX RPM + RAMJET
$\bar{W}$ = 8500 lb
TIME MINS
0
10
20
30
40
$M_\infty$
10
12
14
16
18
20
22
24



FIG. 17.

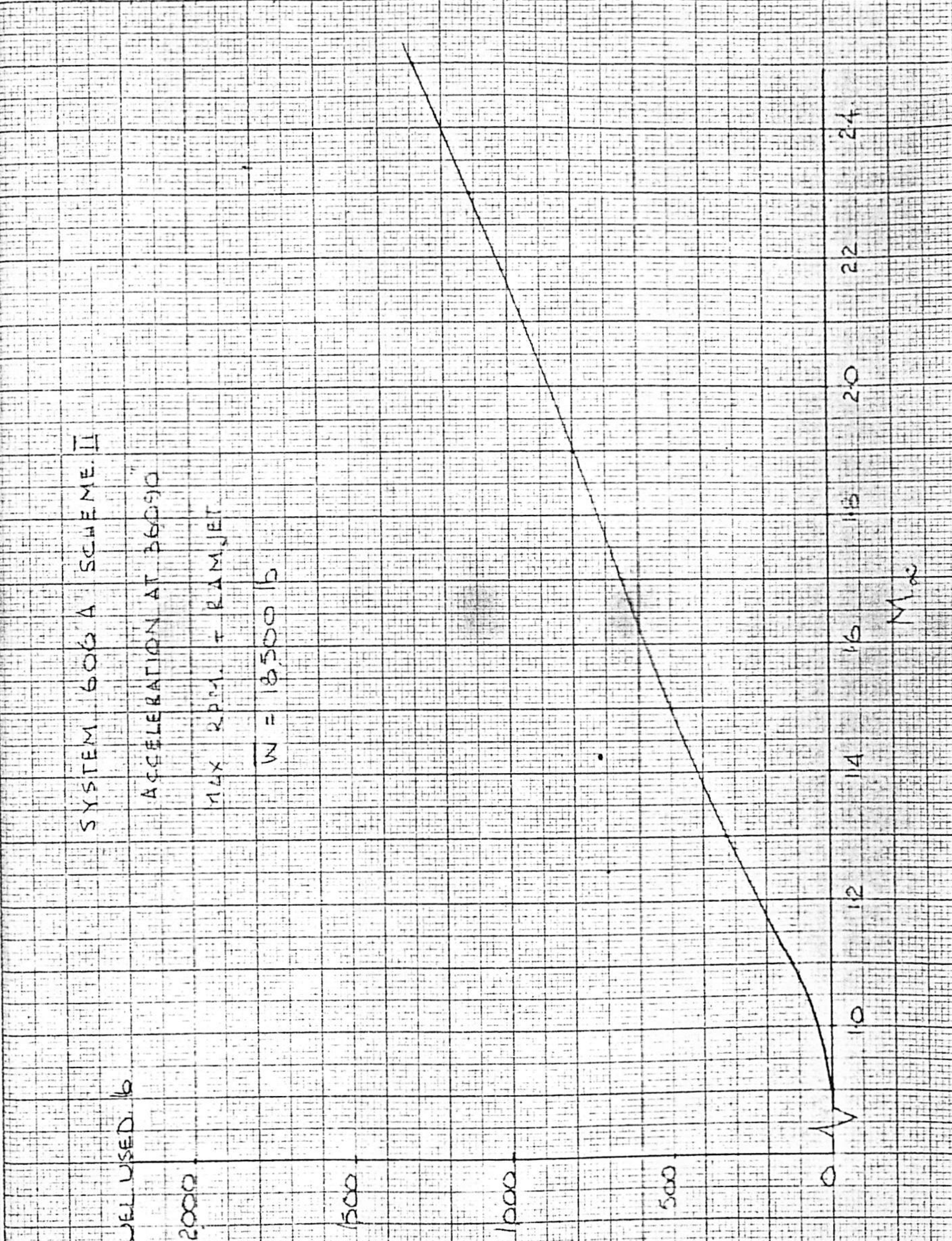
SYSTEM 606A SCHEME II
ACCELERATION AT 36090'
MAX RPM + RAMJET
$\overline{W}$ = 18500 lb
FUEL USED lb
2000
1500
1000
500
0
10
12
14
16
18
20
22
24
$M_\infty$

FIG 18

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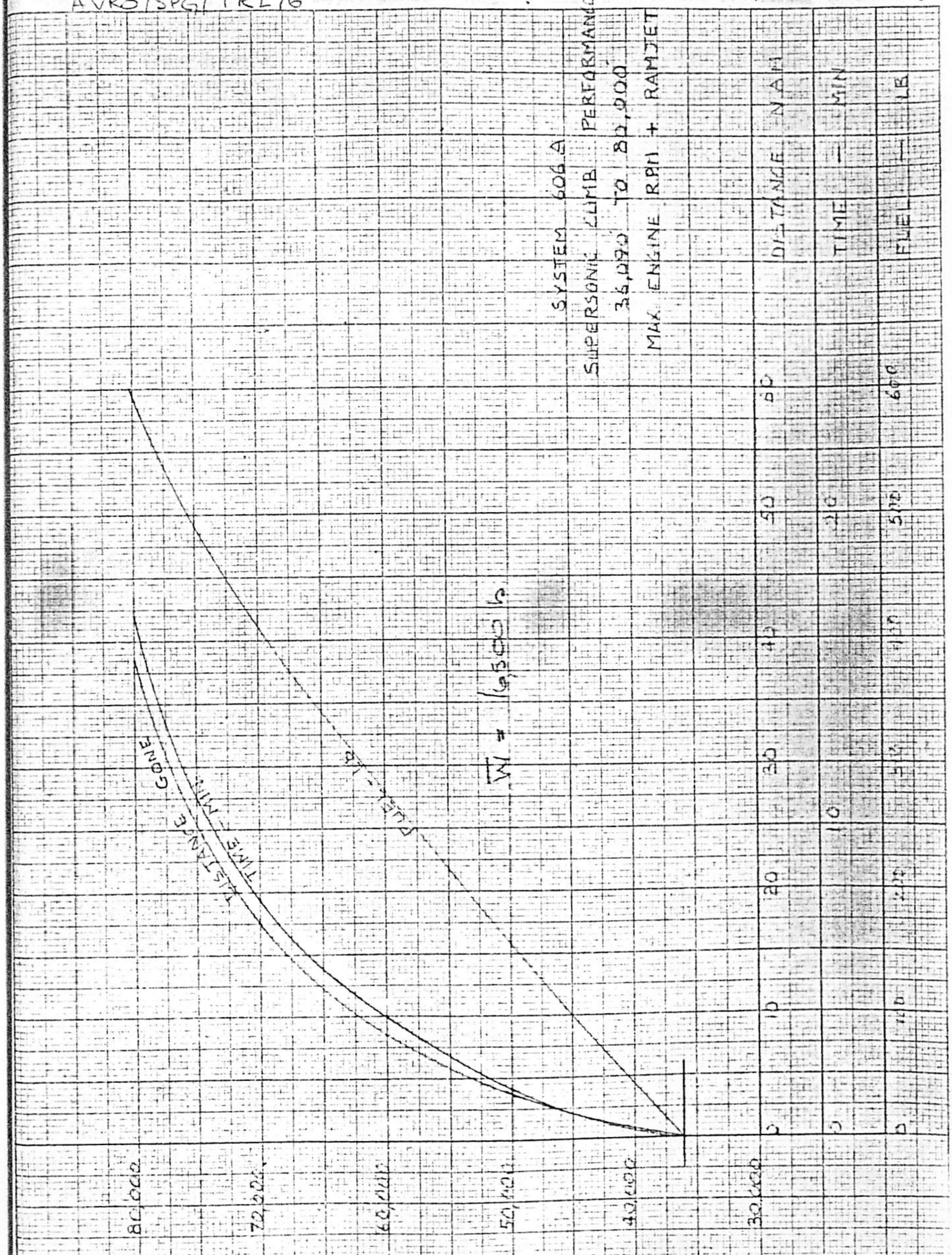




FIG. 19.

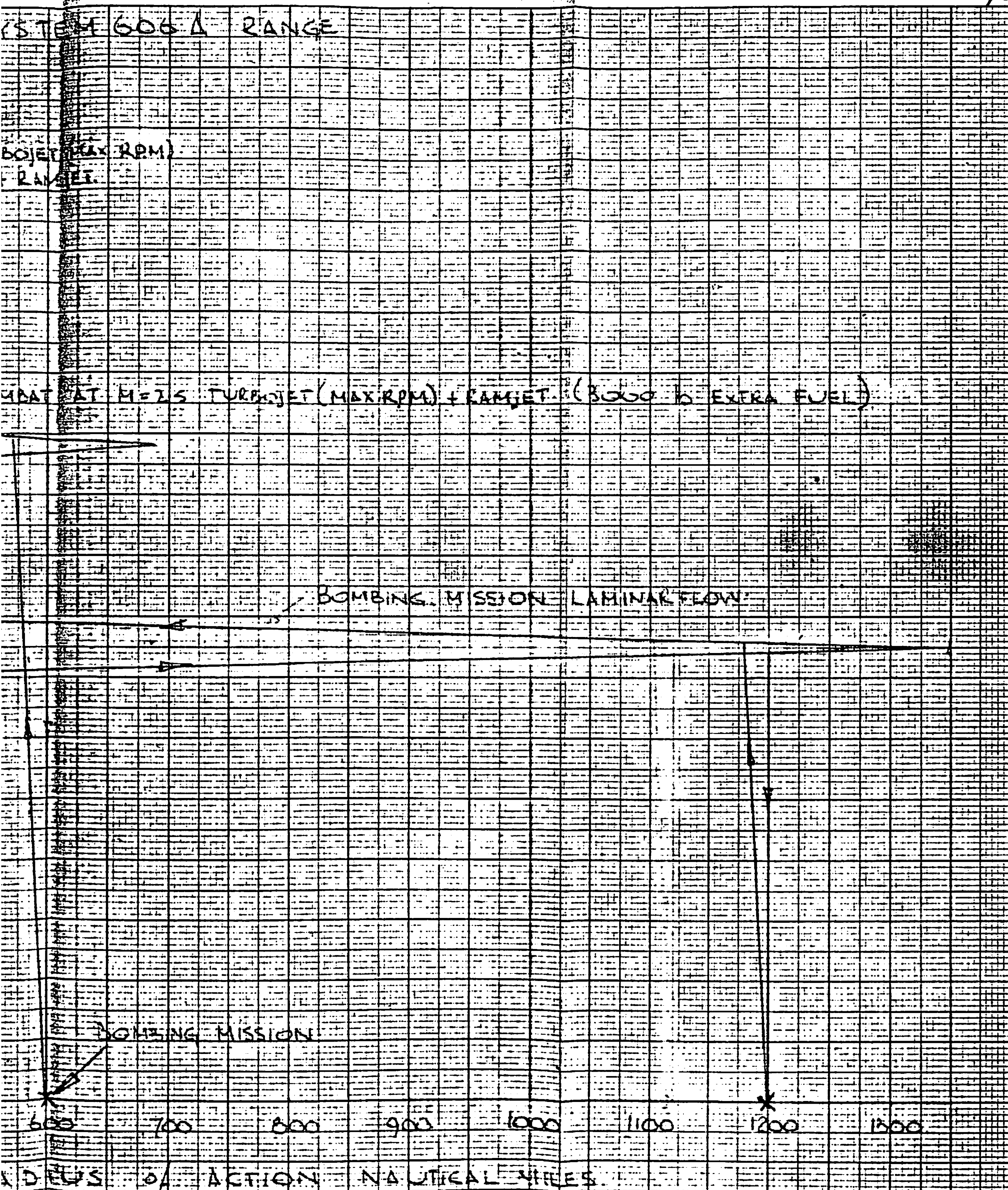
(STEM 606 A RANGE
BOJET (MAX RPM)
RAMJET.
MBAT AT M=2.5 TURBOJET (MAX RPM) + RAMJET (3000 lb EXTRA FUEL)
BOMBING MISSION LAMINAR FLOW
BOMBING MISSION
600
700
800
900
1000
1100
1200
1300
ADIUS OF ACTION NAUTICAL MILES.



FIG 20

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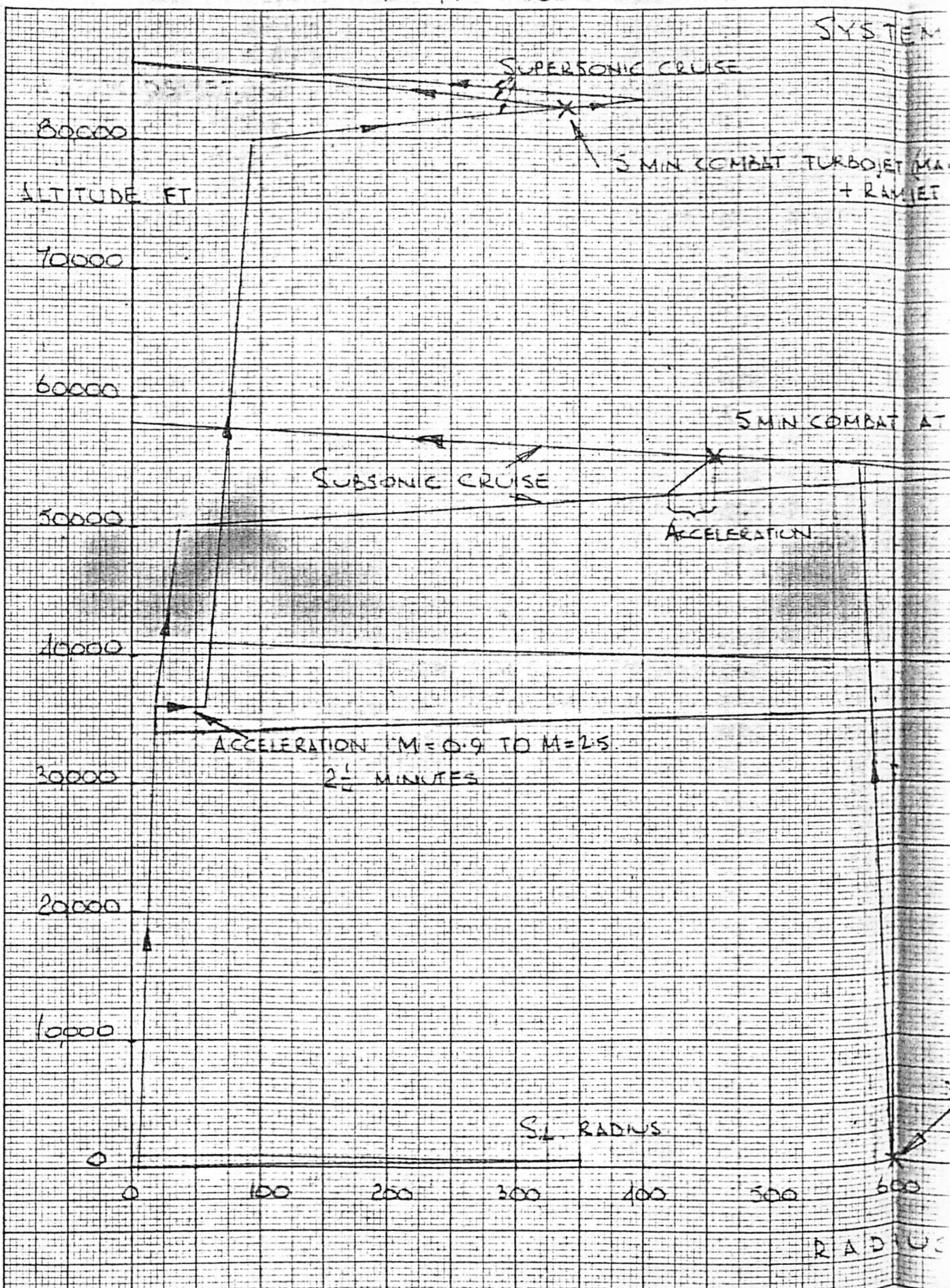
SEC




10 X 10 to the ½ inch, 5th lines accented
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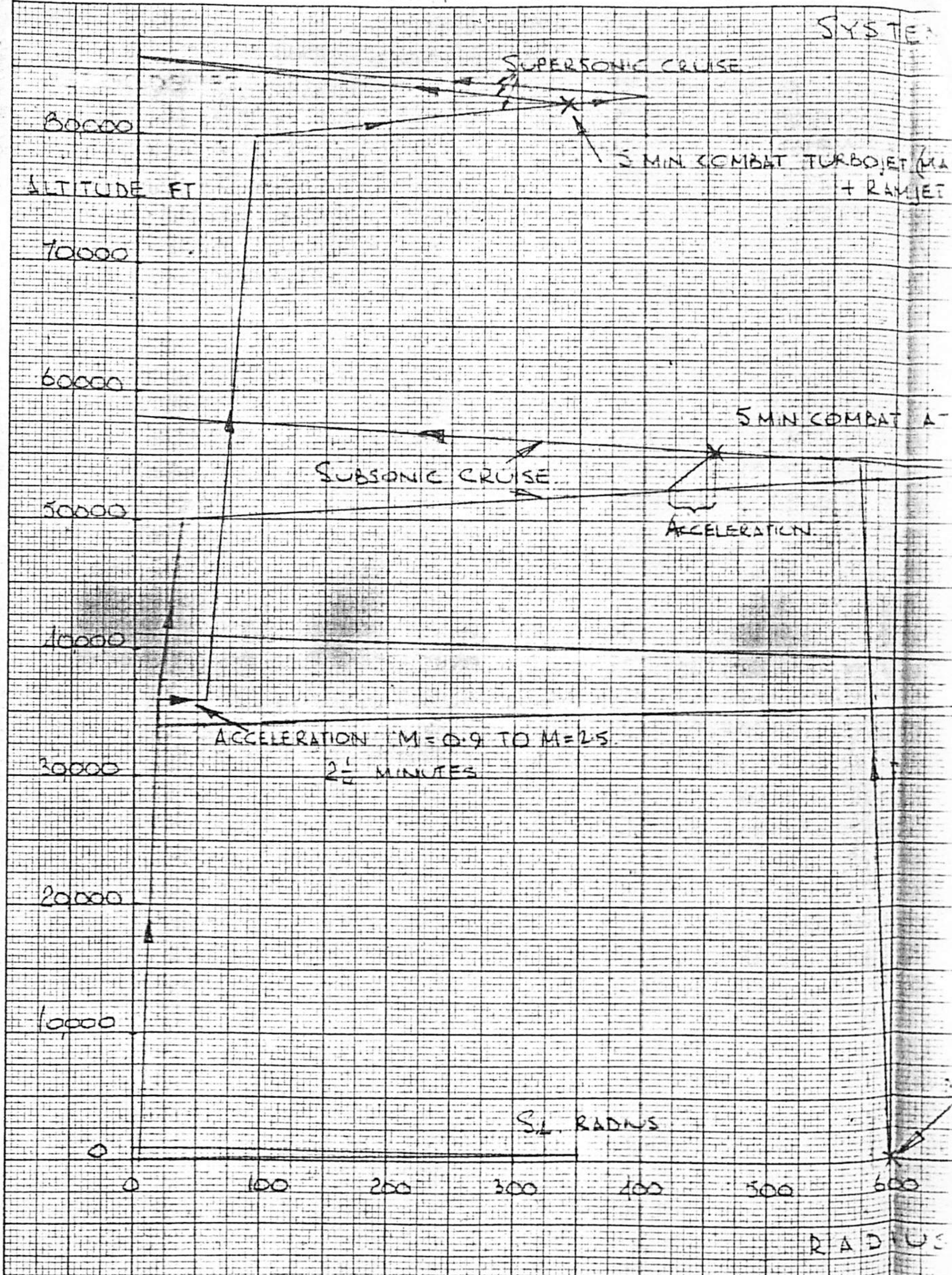


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10 X 10 to the ½ inch, 5th lines accented
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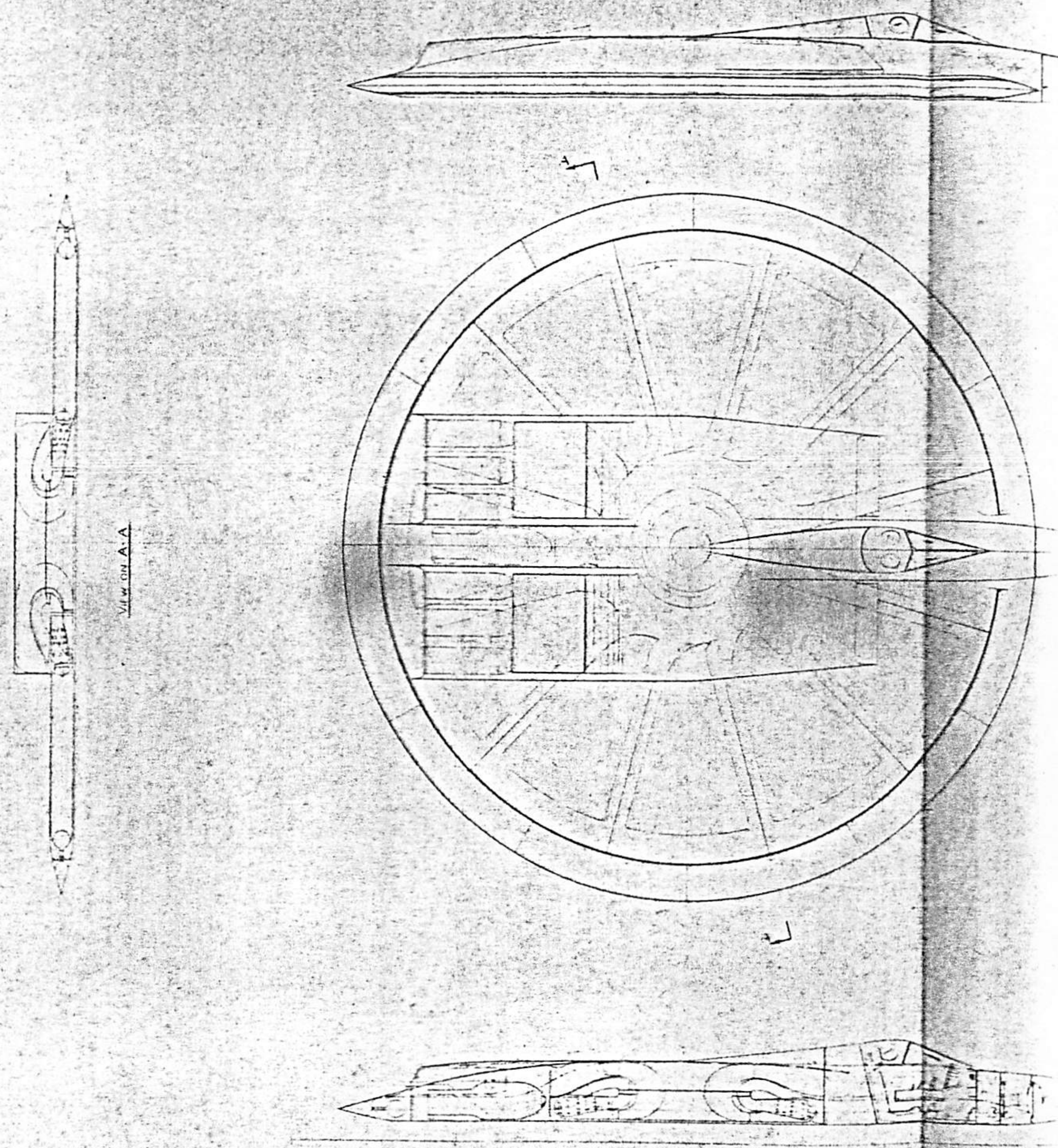
AVRO/SPG/TR 27.6

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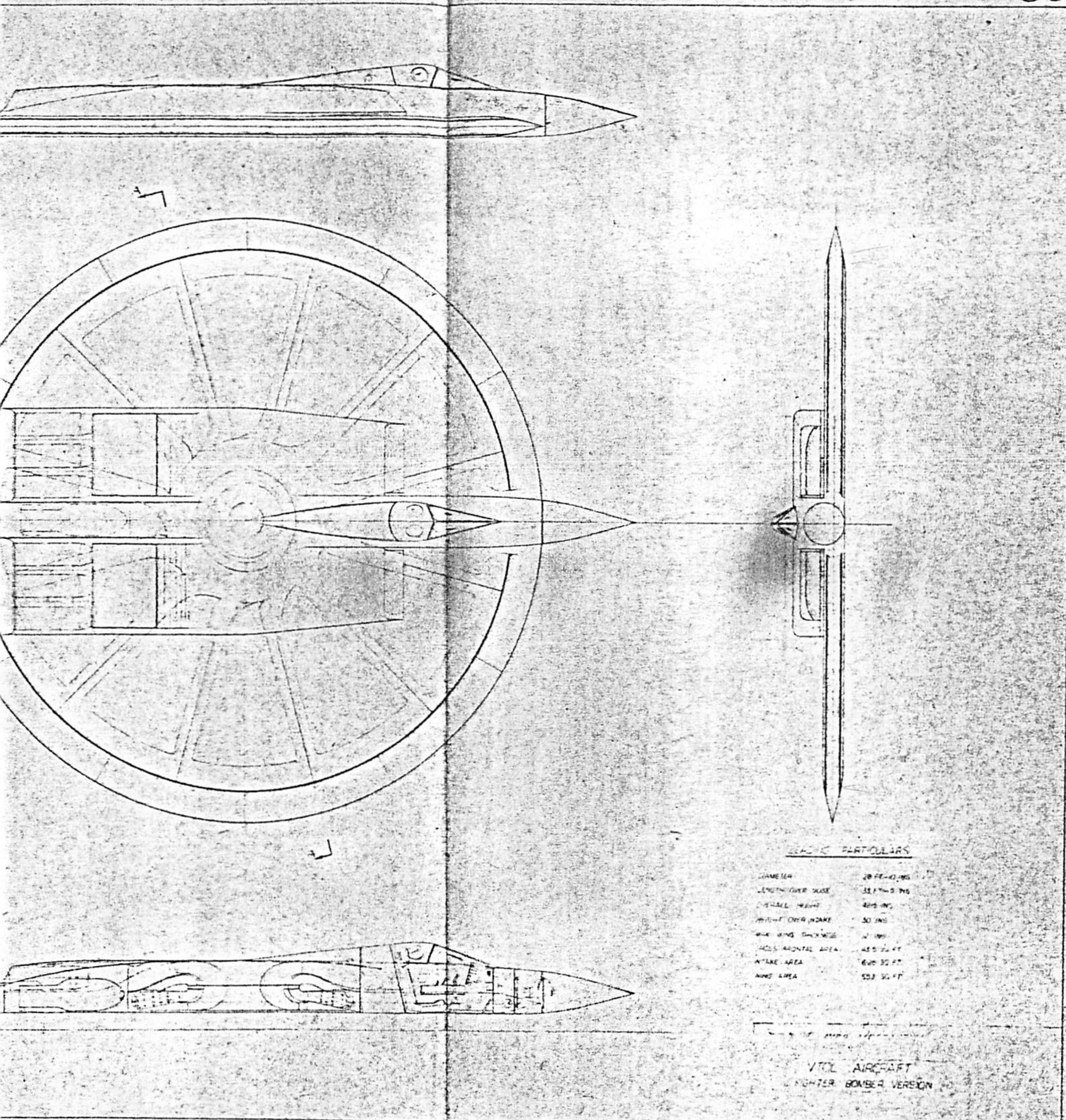


FIG. 21