"Preliminary Design and Analysis of
a Ram-Jet Missile"

Project M-679 GA
Project Wizard

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INTRODUCTION

The purpose of this report is to present a preliminary analysis and design of a missile which fulfills the requirements specified by Project MX -794 as outlined in the University of Michigan report, UMR-8 for the UMA -1 Missile, and which utilizes ram-jet propulsion for the major portion of its thrust. Final dimensional and weight data are to be compared for equal end condition of altitude, horizontal distance and time, to model UMA -1 liquid-rocket missile.

Particular attention has been paid in this analysis to the structural design from a high temperature and high acceleration standpoint, in order to insure that the missile conforms closely to the structural requirements of an actual missile. Drag coefficients have been revised upward to give decidedly more conservative performance results than heretofore. Analyses of the various components making up the missile are described and explained in detail. A section on the homing stage thrust requirements has also been included to show promise of further reducing the initial weight of the missile. Howwver, to maintain as nearly equal bases of comparison with model UMA-1 missile, as possible, identical homing stage thrust requirements are used. Graphs and formulas are presented in the Appendix ~~this report~~ with brief comments on their derivation.

SUMMARY

The missile presented in the report is a three stage missile utilizing a solid propellant stage for launching, a ram-jet stage to accelerate it to a high altitude and velocity, and a rocket stage for controlled flight outside the atmosphere to the target. Initial weight and overall length of the ram-jet missile are considerably lower than that of the liquid rocket missile for identical interception conditions, and are due primarily to the extremely high specific impulse of the ram-jet. The possibility of further reducing the initial weight of the ram-jet missile with consequent reduction of its physical dimensions appears quite favorable. It should be stated, however, that the performance results of the ram-jet are contingent upon the practical solution of combustion problems and upon the design of a practical, completely variable diffuser and tail nozzle.

Discussion

The preliminary analysis and design of the missile was begun with an attempt to determine quantitatively the following conditions in order to fix its structural requirements: (1) maximum axial and radial accelerations; (2) maximum internal and external temperatures; (3) maximum internal and external aerodynamic forces; and (4) stability requirements. Brief calculations of these conditions showed that axial acceleration and internal and external temperature, would be the most important design considerations. Internal pressures were calculated to be approximately 300 psia maximum throughout the flight path. (See Figure 1, Reference UMM-14).

Third Stage Design

Since, from IMC-46, third stage equipment was known, the problem remaining was to position each piece of equipment to hold a fixed c.g. position and to perform its job properly. Thrust requirements were based upon the method used in IMC-46 (see page 9). From these, fuel requirements giving tank and motor sizes were found, and the bulk of the third stage missile dimensions were known, permitting the ram jet stage to be designed.

Second Stage Design -- Ram Jet

The diffuser selected was of the Oswatitsch type, utilizing the third stage missile for the central body. An isentropic spike type of nose was selected for the central body to provide optimum pressure recovery in the subsonic portion of the diffuser. Dimensions of the diffuser were compromised to give optimum pressure recovery throughout with a variable diffuser does the normal shock occur necessarily?

a wide range of Mach numbers, and were obtained from USCAL report R-4-7. Stability of the normal shock is obtained by means of bleed holes located in the forward part of the diffuser hood, according to experiments by Evvard of NACA. Proper design of the bleed holes will minimize/the loss in mass flow through them, while still insuring stability, and almost isentropic pressure recovery. ontraction ratios for Mach numbers from 1.5 to 6.0 were provided for as well as possible in the design by a maximum of fore and aft movement of the central body.

Fuel is injected into the subsonic air stream forward of the flame holders by means of a tubular ring fitted with injection nozzles. The fuel is preheated in tanks surrounding the subsonic portion of the diffuser, and is pumped to the ring from a circumferential tank holding 1200 lbs of fuel, through four high volume, medium pressure fuel pumps. Ignition of the fuel is continuous and provided by a hot wire system latticed across the flame holders at the forward part of the combustion chamber.

Unique?? has this proven successful? - good idea anyway!

The combustion chamber is designed to withstand combustion temperatures up to 3500° F, and incorporates a heavy skin plus a ceramic coating. Heavier guage skin is also provided for other parts of the missile subjected to high temperatures.

Tail nozzle design was complicated by the requirement that for maximum thrust coefficients for Mach numbers of flight less than approximately 3, the tail nozzle should be a straight tube. Above flight Mach numbers of 3, contraction ratios of combustion chamber area to throat area should be greater than 1. These requirements could not be designed

might explain a little

MIGHT be a little hard for someone else to swallow into the nozzle section, and a compromise was made. Control systems for the fore and aft movement of the after body and forward body are provided by means of hydraulically actuated pistons, actuated by an interconnected hydraulic pressure source. Hydraulic pressure is maintained by high pressure hydraulic pumps. Regulation of the positions of the inner bodies is accomplished automatically by a valve operating between subsonic diffuser pressure and atmospheric pressure.

Jet deflectors, similarly controlled as the inner bodies, are provided at the aft end of the ram jet, to control its direction in the atmosphere. ARE THERE WINGS?

First Stage Design - Boost

Solid boost attachment consists of four solid propellant rockets, each capable of delivering approximately 75,000 lbs of thrust for a period of four seconds. The boost rockets were located on the outside of the ram jet shell for three reasons: (1) reduction of overall length for ease of handling; (2) operation of the ram jet during boost; and (3) action as part of the required fin surface. Reason (2) is probably most important, since at the low Mach numbers of boost, any retarded operation of the ram jet could conceivably result in an unsuccessful flight due to the high drag. Provision for jettisoning the boost rockets is made by means of explosive bolts attached to each boost unit. Fins were added to the boost units to complete the required total fin area.
what happens to the fin area when boost rockets are jettisoned??

Weights

Weights of each component and the shell structure were calculated according to standard aircraft practice and are listed in the appendix.

PERFORMANCE ANALYSIS

The final requisite for the performance analysis of the ram jet missile, is, of course, to determine the thrust and drag coefficients as functions of Mach number and altitude. From Report UMM-7, thrust coefficient curves were calculated (see Fig. 4). The diffuser efficiencies used are considerably lower at low supersonic Mach numbers than can actually be realized, tending to keep performance results conservative. It must be stated at this point, however, that the thrust coefficient curves ~~based upon UMM-7~~ require a straight tube of combustion chamber diameter at Mach numbers less than 3, and further, require that the diffuser entrance area equals the combustion chamber area at Mach numbers greater than 3. Although these requirements are not entirely met by the ram jet missile designed as shown, it is felt that by the time an actual flying missile is built, sufficient progress will have been made in design to insure the thrust coefficients used, and that the performance given is not at all optimistic.

Drag coefficients for the first two stages were obtained from Continental Aviation and Engineering Report 340, which used a missile of similar design (see Fig. 4). Coefficients plotted are based upon combustion chamber area. Drag coefficients for the third stage are taken from NACA report ACR-L5H08 and are based upon a missile of L/D = 5.5.

With thrust and drag coefficients plotted, it was tentatively decided to launch to M = 2.0 with a total burning time for Stages I and II of 23 seconds. A few climbs to required altitude were run to check the performance. The altitude at the end of boost was calculated from the relationship (see page 14 for derivation):

$$\frac{(cM)^2}{2gS} = \frac{I\dot{w} - D_{av} - w_o + \frac{\dot{w}S}{cM}}{w_o - \frac{\dot{w}S}{cM}}$$

c - velocity of sound - fps

S - vertical distance - ft.

I - specific impulse - sec.

$\dot{w}$ - Rate of burning - lbs/sec

w_o - initial weight - lbs.

D_{av} - average drag during launching

M -

suggest separate page & complete notation + symbols

Launching is vertical, and the missile is swung through an angle of 26° from the vertical at the start of ram jet operation. ~~how?~~

Several calculations to the required final altitude were then made, using a step by step process for the second and third stages (see Appendix, page). When the drag became less than 100 pounds, the following equations were used (UMR-8, Appendix A, page 6):

$$V_{v(n)} = V_{v(n-1)} + \frac{Tg}{\dot{w}} \cos \quad \log \frac{w_o}{w_o - \dot{w}t_{(n)}} - g\, t_{(n)}$$

$$V_{H(n)} = V_{H(n-1)} + \frac{T_g}{\dot{w}} \sin \log \frac{w_o}{w_o - \dot{w}t_{(n)}}$$

$$S_{V(n)} = S_{V(n-1)} + V_{V(n-1)} t_{(n)} + \frac{T_g \cos}{\dot{w}^2} \left[wt_{(n)} - (w_o - \dot{w}t_{(n)}) \log \frac{w_o}{w_o - \dot{w}t_{(n)}} \right] - \tfrac{1}{2} g\, t_{(n)}^2$$

$$S_{H(n)} = S_{H(n-1)} + V_{H(n-1)} t_{(n)} + \frac{T_g \sin}{\dot{w}^2} \left[wt_{(n)} - (w_o - \dot{w}t_{(n)}) \log \frac{w_o}{w - \dot{w}t_{(n)}} \right]$$

V_V - vertical velocity - fps.

V_H - horizontal velocity - fps.

S_V - vertical distance - ft.

S_H - horizontal distance - ft.

T - thrust - lbs.

g - 32.2 ft/sec^2

$\dot{w}$ - rate of burning - lbs/sec

n - stage number

$t_{(n)}$ - time of burning during nth stage - sec.

w_o - weight at beginning of nth stage

θ - flight path angle from vertical = 26°

Rates of burning of the third stage control and stabilization motors were calculated by the method given in IMC-46 pp. 1 - 4.

	$\dot{w}$ 170-180 sec	$\dot{w}$ 23-170 sec
Lateral thrust	21.9 lbs/sec	4.10 lbs/sec
Pitch, yaw & roll	3.13	3.13
Turbine	1.07	1.07
	26.10	8.30

Total fuel = 26.1 x 10 + 8.3 x 147 = 1485 lbs.
(third stage)

Check for Horizontal Distance Correction (23-170 sec):

Required distance = 58,300 ft.

$$S_1 = \frac{T_g}{\dot{w}^2}\left[\dot{w}t_{(n)} - (w_o - \dot{w}t_{(n)}) \ln \frac{w_o}{w_o - \dot{w}t_{(n)}}\right]$$

$$= \frac{4.10}{21.9} \times \frac{2800}{(8.3)^2} g\left[8.3 \times 147 - (3492 - 1224) \ln \frac{3492}{3492 - 1224}\right]$$

$$S_1 = 246\left[1224 - (2268) \ln \frac{3492}{2268}\right] = 60,000 \text{ ft.}$$

Since S_1 is greater than 58,300 ft. these burning rates sufficed.

The motors used were: (1), Lateral thrust motors; ~~(8)~~, at 700 lbs. thrust each; Arranged in cruciform, two fired in each quadrant, consuming fuel at the rate of 4.10 lbs/second; and (2) Pitch, yaw, and Roll stabilization motors, ~~two (2)~~ motors, universally swiveling, providing 400# thrust each. Performance calculations have included the 800 lbs. thrust from these motors during the entire third stage. what 300#s? Explain!

Ram jet burning time was varied to find that minimum burning time necessary for the missile to reach the required altitude, distance, and time specified for the UMA-1 missile.

Next, a complete analyses were made upon missiles designed for 10 g, 25 g, and 40 g to determine a weight versus acceleration curve for comparable performance (see Fig. 2). The minimum point showed which acceleration gave the missile of the least initial weight, or, in other words, the most efficient missile.

Figures 6 through 8 show plots of Mach number after launching and boost fuel weight versus initial weight of the missile. Parameters of rates of burning are shown. Three design missiles are used, 10 g, 25 g, and 40 g. Equations used are:

$$M = \frac{g}{c\dot{w}}\left[Iw - D_{av} - \left(w_o - \frac{\dot{w}t}{2}\right) \ln \frac{w_o}{w_o - \dot{w}t} \right]$$

(Symbols used are the same as on page 7 , except that t = time of boost - sec.)

$$wt_{10} = \frac{w_o - 7586}{1.25} \qquad \text{(10 g missile)}$$

$$wt_{25} = \frac{w_o - 7886}{1.25} \qquad \text{(25 g missile)}$$

$$wt_{40} = \frac{w_o - 8186}{1.25} \qquad \text{(40 g missile)}$$

(See Appendix, pp. 15 and 16 for derivations)

Plots of maximum acceleration and launching Mach number versus launching time are included (See Figs. _ through _) for 10 g, 25 g, and 40 g missiles. Maximum accelerations of the flight path are attained during launching. Equations used were:

$$G_{max} = \frac{I\dot{w} - D_{av} - (w_o - \frac{\dot{w}t}{2})}{w_o - \dot{w}t} + \frac{1}{2} \ln \frac{w_o}{w_o - \frac{\dot{w}t}{2}}$$

and $$t_{10} = \frac{\dot{w}t}{\dot{w}} = \frac{w_o - 7586}{1.25\,\dot{w}} \qquad \text{(10 g missile)}$$

$$t_{25} = \frac{w_o - 7886}{1.25\,\dot{w}} \qquad \text{(25 g missile)}$$

$$t_{40} = \frac{w_o - 8186}{1.25\,\dot{w}} \qquad \text{(40 g missile)}$$

(See Appendix, p. 16 for derivation).

CONCLUSIONS AND RECOMMENDATIONS

1. The initial gross weight of the ram jet missile = 14460 lbs. as compared to an initial gross weight of 17600 lbs. for model UMA-1.

2. Lowest initial gross weight is obtained by designing the missile for 25-30 g acceleration.

3. It appears that no great benefit will be derived in reanalyzing a ram jet missile until more test data to substantiate or disprove present assumptions are available.

4. It is apparent that the ram jet indicates a tremendous future as a means of propulsion, primarily due to its extremely high specific impulse. Secondary factors such a ease of servicing, manufacture, and storage stability are also not to be overlooked in a comparison with a liquid rocket missile.

REFERENCES

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3. University of Michigan UMM-14. "Aerodynamic Heating of a Supersonic Missile During Accelerated Flight" D. W. Lueck. (October 1947).

4. University of Michigan UMM-7. " A Simplified Method of Calculating Ram Jet Performance Applicable to High Mach Numbers." J. R, Gannett. (July 1947).

5. Continental Aviation and Engineering Corporation report no. 340. "Estimation of Drag Coefficients of Mach-3 Missile." J.F.Roth. (September 1947).

6. NACA-ACR-L5H08. "Supersonic Tunnel Tests of Projectiles in Germany and Italy." Antonio Ferri. (October 1945).

7. JPL CIT, Memo no. 4-17. "The Applicability of Solid Propellant to High Performance Rocket Vehicles." Summerfield, et al. (October 1947).

8. University of Southern California R-4-7. "Performance and Geometry of High Thrust and Long Range Ram Jets." (February 1948).

APPENDIX

Derivation of Equations

A. Vertical Distance Climbed During Boost.

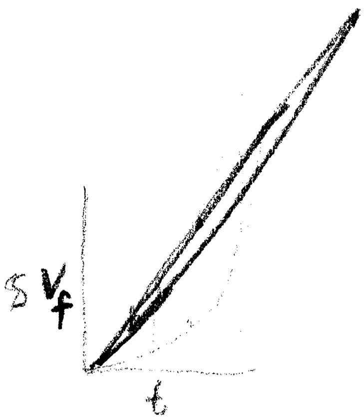
$$S = S_0 + v_0 t + \frac{1}{2} a t^2 \qquad S_0 = v_0 = 0$$

$$S = \frac{1}{2} a t^2 \qquad a = a_{average}$$

$$a_{av} = \frac{F_{net\ av}}{Mass_{av}} = \frac{\text{Gross Thrust} - Drag_{av} - Weight_{av}}{(Weight_{av})(\frac{1}{g})}$$

$$= \frac{\left[I\dot{W} - D_{av} - (W_o - \frac{\dot{W}t}{2})\right] g}{(W_o - \frac{\dot{W}t}{2})}$$

$$W_{av} = W_o - \frac{\dot{W}t}{2} = W_0 - \frac{\dot{W}S}{2\frac{v_f}{2}} = W_o - \frac{\dot{W}S}{v_f}$$



Since $t = \frac{S}{\frac{v_f}{2}}$ which, plotted, is nearly linear, this relation may be used.

Also, $a_{av} = \frac{v_f - v_o}{t} = \frac{v_f}{t} = \frac{v_f}{\frac{2S}{v_f}} = \frac{v_f^2}{2S}$

$$\frac{v_f^2}{2S} = \frac{(I\dot{W} - D_{av} - W_o + \frac{\dot{W}S}{v_f})\, g}{W_o - \frac{\dot{W}S}{v_f}}$$

or, $$\frac{(cM)^2}{2gS} = \frac{I\dot{W} - D_{av} - W_o + \frac{\dot{W}S}{cM}}{W_o - \frac{\dot{W}S}{cM}}$$

B. Launching Mach Number vs. Initial Weight.

Assumptions: Vertical Flight Path

$W = W_0 - \dot{w} t$

Net Thrust = Gross Thrust - Drag - Weight

$$\frac{W}{g}\frac{dv}{dt} = I\frac{dw}{dt} - C_D \frac{1}{2}\rho A V^2 - W$$

$$\frac{dv}{dt} = \frac{Ig}{W}\frac{dw}{dt} - C_D\frac{1}{2}\rho g A V^2 \frac{1}{W} - g$$

$$dv = Ig\frac{dw}{W} - \text{Drag}\, g \frac{dt}{W} - g\,dt$$

$$\int_{v_0}^{v_f} dv = \int_{W_0}^{W} Ig\frac{dw}{W} - D_{AV}\, g\int_{t_0}^{t}\frac{dt}{W} - \int_{t_0}^{t} g\,dt$$

$$\int \frac{dt}{W_0 - \dot{w}t} = \qquad \int\frac{dt}{a+bt} = \frac{1}{b}\log(a+bt)$$

clarify → $v_0 = t_0 = 0$ since $W = W_0 - \dot{w}t$

$- Ig \log\left(\frac{W}{W_0}\right)$

$$v_f = Ig\log\left(\frac{W_0}{W}\right) - D_{AV}\, g\left[-\frac{1}{\dot{w}}\log W\right]_{t_0}^{t} - gt \qquad t_0 = 0$$

$$v_f = Ig\log\frac{W_0}{W}\left[1 - \frac{D_{av}}{I\dot{w}}\right] - gt$$

Since $$t = \frac{v_f}{a_{av}} = \frac{v_f}{\left(\frac{I\dot{w} - D_{av} - W_{av}}{W_{av}}\right)g}$$

then $$v_f = Ig\log\frac{W_0}{W}\left[\frac{I\dot{w} - D_{av}}{I\dot{w}}\right] - \frac{v_f W_{av}}{I\dot{w} - D_{av} - W_{av}}$$

$$v_f \left(1 + \frac{W_{av}}{I\dot{w} - D_{av} - W_{av}} \right) = Ig \log \frac{W_0}{W} \left[\frac{I\dot{w} - D_{av}}{I\dot{w}} \right]$$

$$v_f = \frac{g}{\dot{w}} \left[I\dot{w} - D_{av} - W_{av} \right] \log \frac{W_0}{W}$$

$$v_f = \frac{g}{\dot{w}} \left[I\dot{w} - D_{av} - \left(w_0 - \frac{\dot{w}t}{2} \right) \right] \log \frac{W_0}{w_0 - \dot{w}t}$$

C. Maximum Acceleration vs. Initial Weight

DEFINE

$$\frac{d v_f}{dt} = G = \frac{I\dot{w} - D_{AV} - \left(W_0 - \frac{\dot{w}t}{2} \right)}{W_0 - \dot{w}t} + \frac{1}{2} \log \frac{W_0}{W_0 - \frac{\dot{w}t}{2}}$$

D. Ratio of Initial Launching Weight to Final Launching ?

Weight. EXPLAIN !!

From Alleghany Ballistics Laboratories: FOR what Rocket B.J. !!

$$\frac{\text{Propellant Weight}}{\text{Gross Weight}} = 0.80$$

or $$\frac{\text{Propellant Weight}}{\text{Shell Weight}} = 4.0$$

let W_o = Initial Weight

$$W_o = W_{\text{stage III}} + W_{\text{stage II}} + W_{\text{boost fins}} + W_{\text{boost fuel}} + W_{\text{boost shell}}$$

For a 10 g missile:

$$W_o = 3492 + 3484 + 610 + W_f\left(1 + \frac{W_s}{W_f}\right)$$

$$= 7586 + 1.25\,W_f = 7586 + 1.25(W_o - W)$$

$$W = \frac{0.25\,W_o + 7586}{1.25}$$

$$\frac{W_o}{W} = \frac{1.25\,W_o}{0.25\,W_o + 7586}$$

Adding 200 lbs. for each additional 10 g:

↖ THIS IS OBVIOUSLY THE THING TO DO I GUESS!!

$$\frac{W_o}{W} = \frac{1.25\,W_o}{0.25\,W_o + 7886} \qquad \text{EXPLAIN} \quad (25\text{ g})$$

$$\frac{W_o}{W} = \frac{1.25\,W_o}{0.25\,W_o + 8186} \qquad (40\text{ g})$$

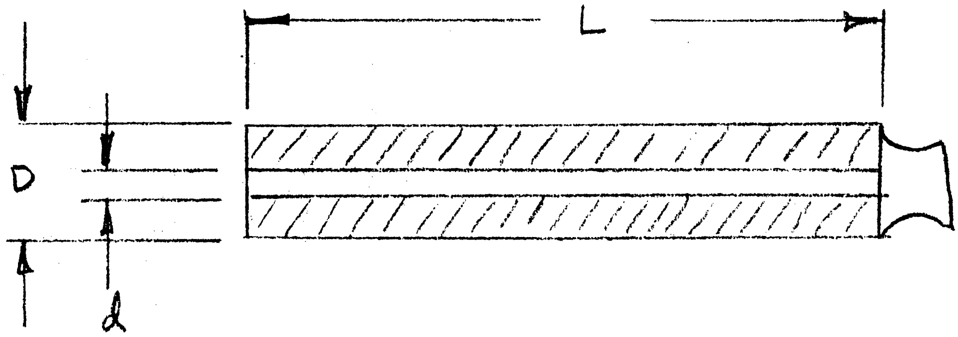
Boost Size Calculation - Separate Boost Units

For M = 2.1, graph of 25 g missile gives:

$\dot{W}$ = 1600 lbs. per sec.

W_o = 14460 lbs.

$W_{boost\ fuel}$ = 6080 lbs.



$d = \frac{1}{3} D$

$$V = \frac{\pi}{4} (D^2 - d^2)l = \frac{\pi}{4} (D^2 - d^2) 10D = 6.98\ D^3$$

From JPLCIT - M- 4-17.q

$w_{boost\ fuel}$ = 84.6 lbs. per cu. ft.

Total volume boost fuel = $\frac{6080}{84.6}$ =124,000 in^3

Volume per cylinder = 31,000 cu. in.

$D^3 = \frac{31000}{6.98}$ = 4450 cu. in.

D = 16.4 in.

L = 164.0 in.

STAGE 1 ITEM	WEIGHT	ARM	MOMENT
STRUCTURE	(610)		
Fins	420		
Shell Structure	160		
Fairing	30		
POWER PLANT GROUP	Variable- Depends upon Mach Number & Desired Burning Rate		
NOTE: Add 200 lbs. for each 10-G Acceleration above 10-G			

STAGE 2 ITEM	WEIGHT	ARM	MOMENT
WEIGHT EMPTY	(2284)		
BODY GROUP	(1783)		
Shell	1483		
Stage III Supports & Act.	225		
Stage I Supporting Struct.	75		
POWER PLANT GROUP	(295)		
Injector Ring	10		
Pumps (4)	40		
Fuel Tanks	25		
Flame Holder	15		
Afterbody	75		
Afterbody Supports & Act.	90		
Propellant System	40		
CONTROL SYSTEM	(206)		
Deflectors & Supports	150		
Control Actuators	56		
USEFUL LOAD			
Propellant	1200		
Stage 3	3492		
STAGE II TOTAL	3484		
TOTAL STAGE III & II	6976		

STAGE 3 ITEM	WEIGHT	ARM	MOMENT
WEIGHT EMPTY	(16[illegible])		
BODY GROUP	(529)		
Nose Section	18		
Engine-Tail Group	511		
POWER PLANT GROUP	(374)		
Lateral Disp. Motors (8)	200		
Stabilization Motors & Inst.	30		
Turbine Pumps	59		
Motor Propellant System			
Turbine Propellant System			
Oxidizer Tank & Installation	30		
Fuel Tank & Installation	18		
N_2 - Fuel Tank System & Inst.	16		
N_2 - Oxidizer Tank " "	21		
FIXED EQUIPMENT GROUP	(571)		
Generator	33		
Power Pack	115		
Homing Radar Equip't	250		
Command Equipment	78		
Control System	85		
Stage Disconnect Switch	10		
STAGE SUPPORT	30		
USEFUL LOAD	(1798)		
Warhead	300		
Propellant	(1485)		
Oxidizer	1147		
Fuel	338		
Nitrogen	(13)		
Oxidizer	8		
Fuel	5		
TOTAL	3492		

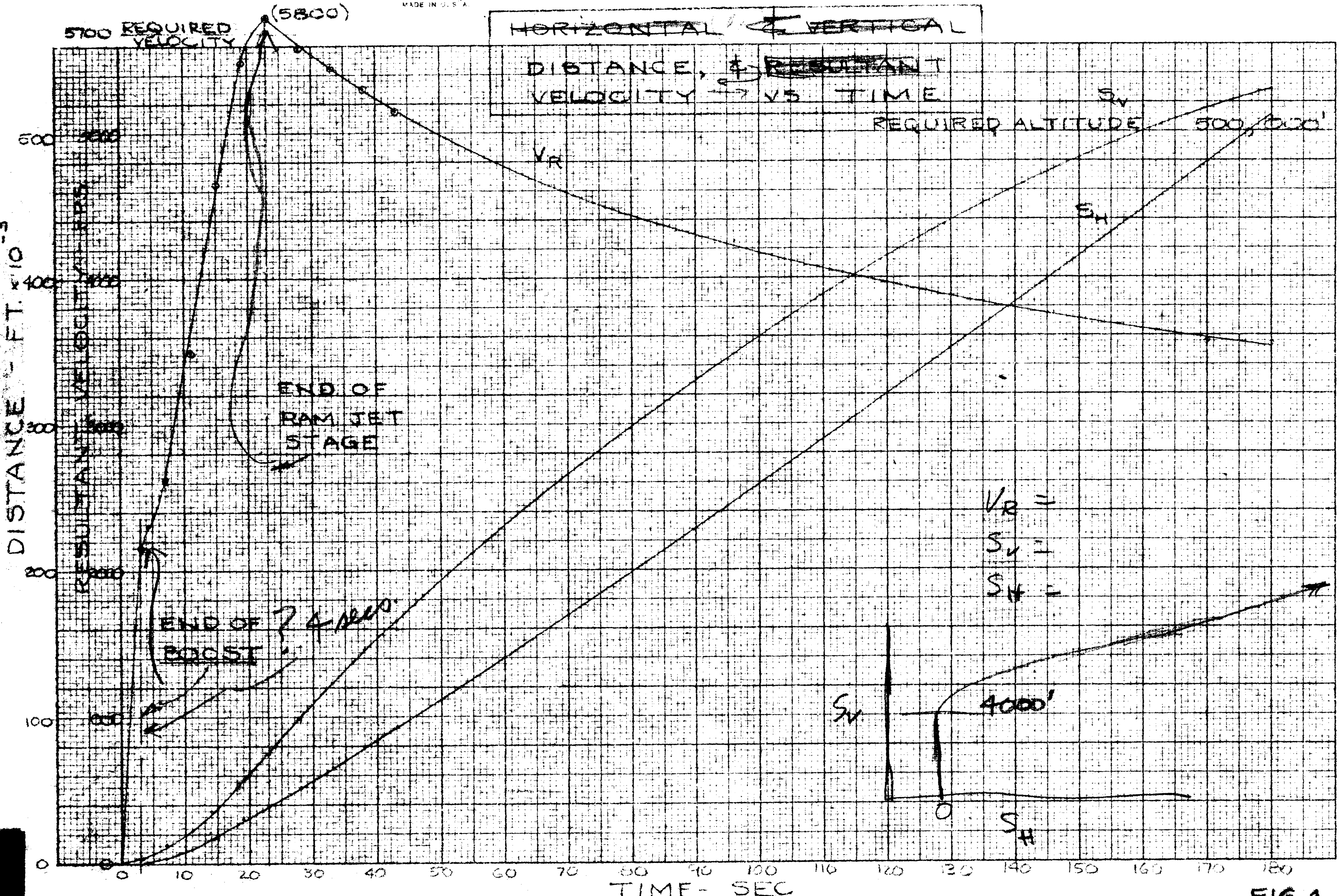
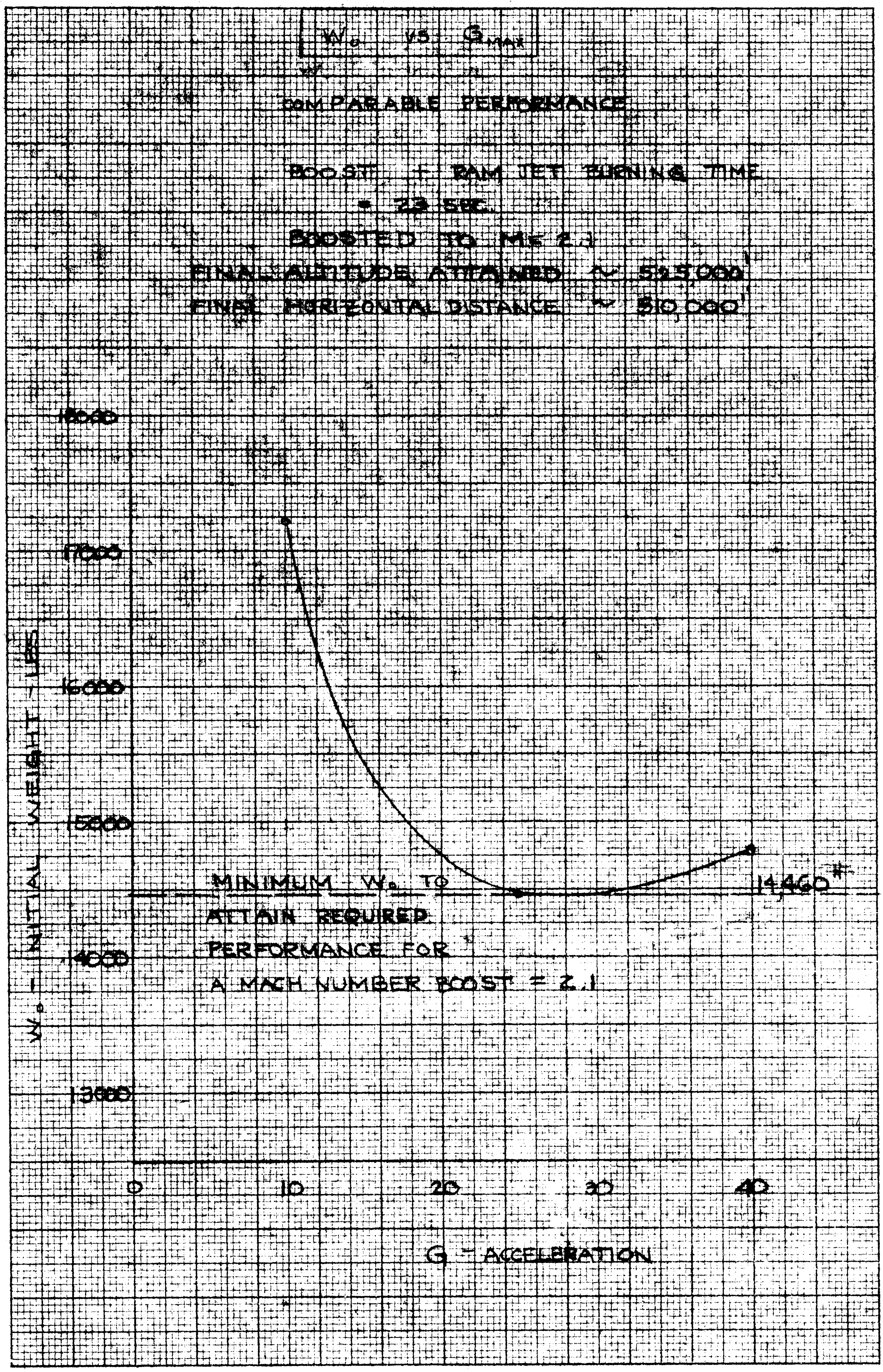

HORIZONTAL & VERTICAL
DISTANCE, & RESULTANT
VELOCITY VS TIME
REQUIRED ALTITUDE 500,000'
5700 REQUIRED VELOCITY
(5800)
V_R
S_V
S_H
END OF RAM JET STAGE
END OF BOOST
? 4 secs
V_R =
S_V =
S_H =
4000'
0
DISTANCE - FT. × 10⁻³
RESULTANT VELOCITY - FPS
TIME - SEC
0
100
200
300
400
500
1000
2000
3000
4000
5000
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150
160
170
180

FIG. 1



W₀ VS G MAX
COMPARABLE PERFORMANCE
BOOST + RAM JET BURNING TIME = 23 SEC.
BOOSTED TO M = 2.1
FINAL ALTITUDE ATTAINED ~ 525,000'
FINAL HORIZONTAL DISTANCE ~ 510,000'
18000
17000
16000
15000
14000
13000
W₀ - INITIAL WEIGHT - LBS
MINIMUM W₀ TO ATTAIN REQUIRED PERFORMANCE FOR A MACH NUMBER BOOST = 2.1
14,460#
0
10
20
30
40
G - ACCELERATION

FIG. 2

APPROXIMATE
SKIN TEMP. VS TIME & ALTITUDE

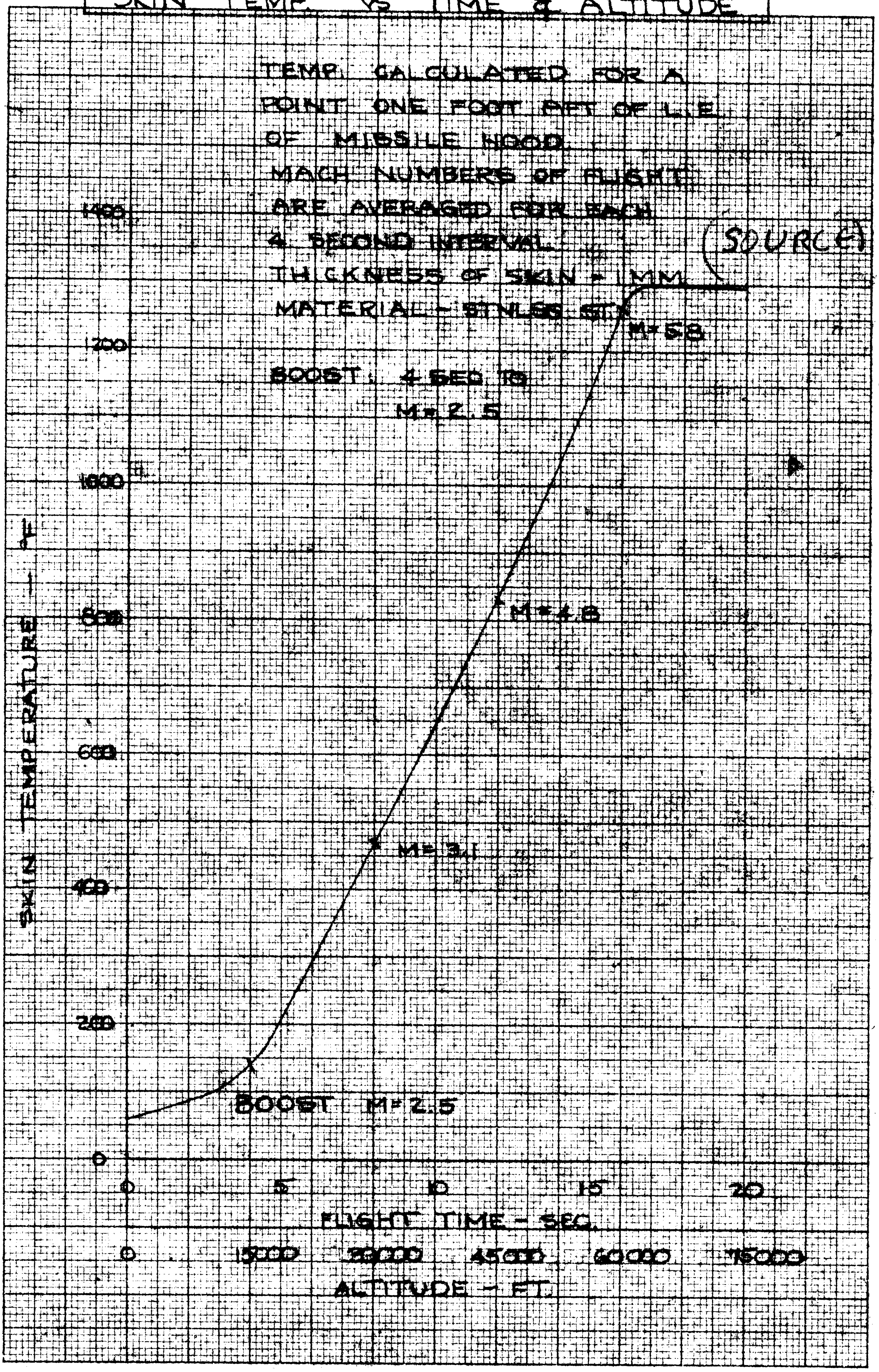


FIG. 3

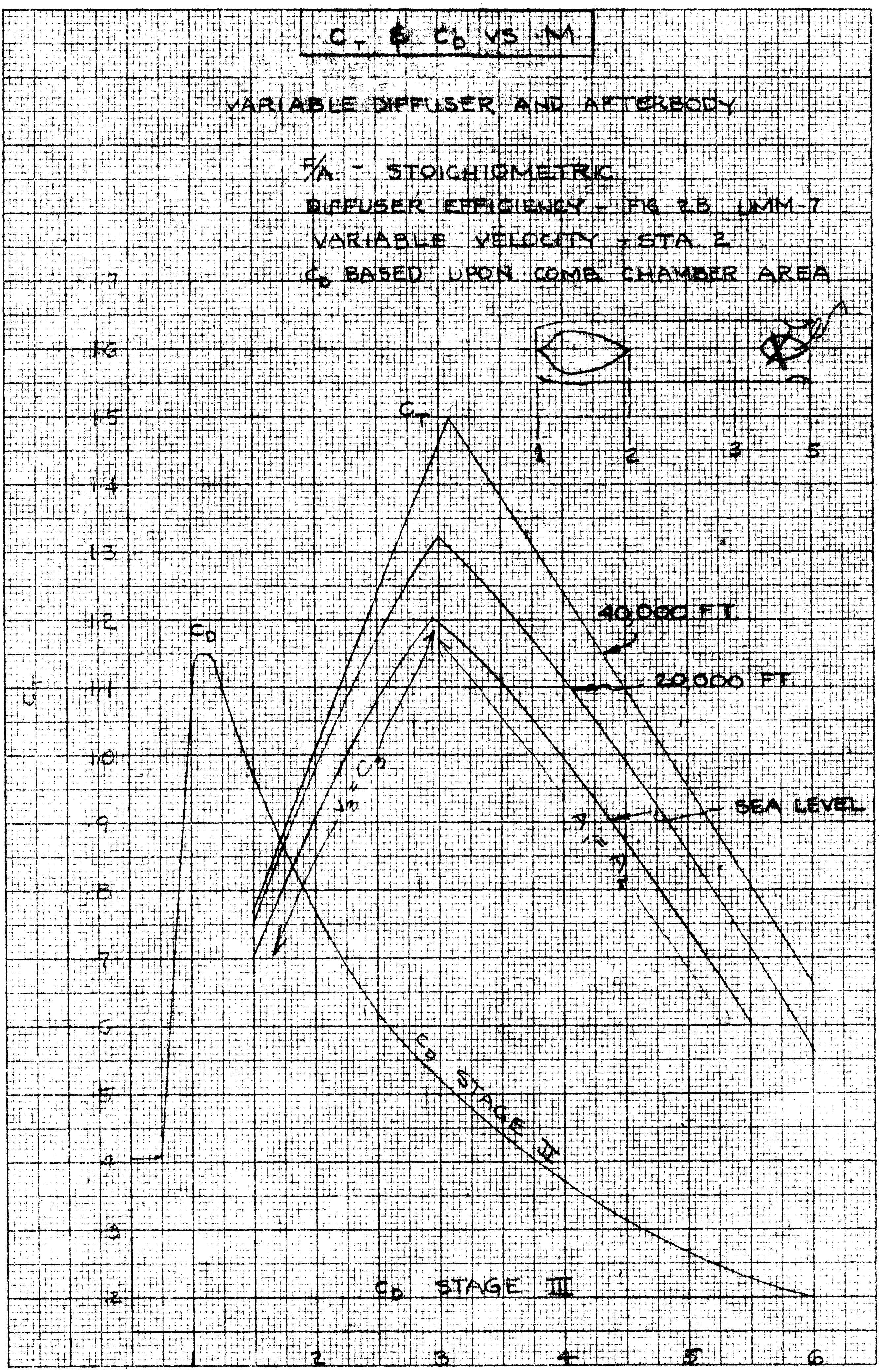

C_T & C_D VS M
VARIABLE DIFFUSER AND AFTERBODY
F/A - STOICHIOMETRIC
DIFFUSER EFFICIENCY - FIG 2B UMM-7
VARIABLE VELOCITY - STA 2
C_D BASED UPON COMB CHAMBER AREA
C_T
C_D
40,000 FT
20,000 FT
SEA LEVEL
C_D STAGE II
C_D STAGE III
FLIGHT MACH NUMBER
1
2
3
5

FIG. 4

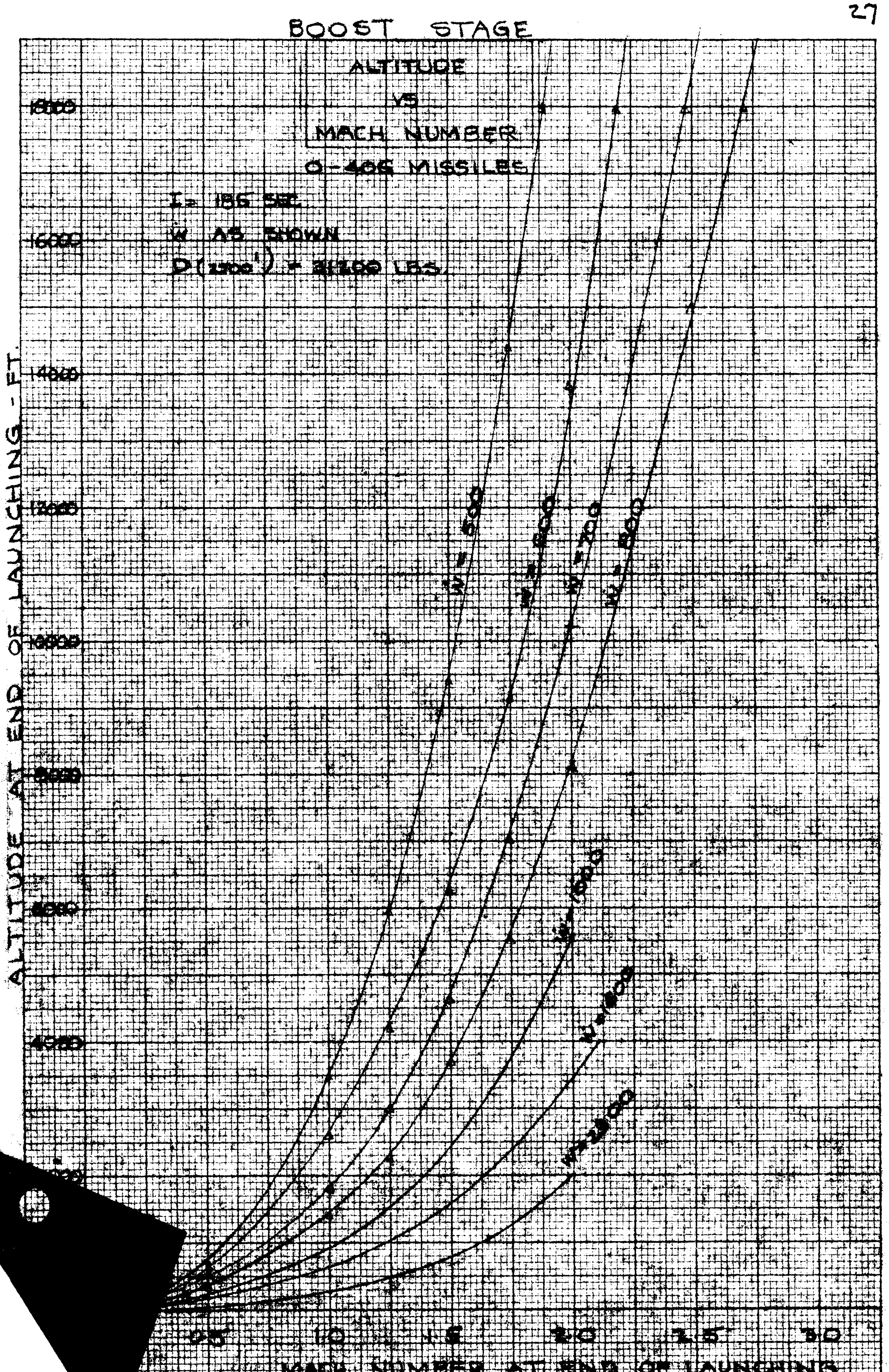

BOOST STAGE
ALTITUDE
VS
MACH NUMBER
O-406 MISSILES
I = 185 SEC
W AS SHOWN
D (2700') = 21200 LBS.
ALTITUDE AT END OF LAUNCHING - FT.
MACH NUMBER AT END OF LAUNCHING
W = 500
W = 600
W = 700
W = 800
W = 1000
W = 1500
W = 2500
1800
1600
1400
1200
1000
800
600
400
0.5
1.0
1.5
2.0
2.5
3.0

FIG. 5

BOOST STAGE

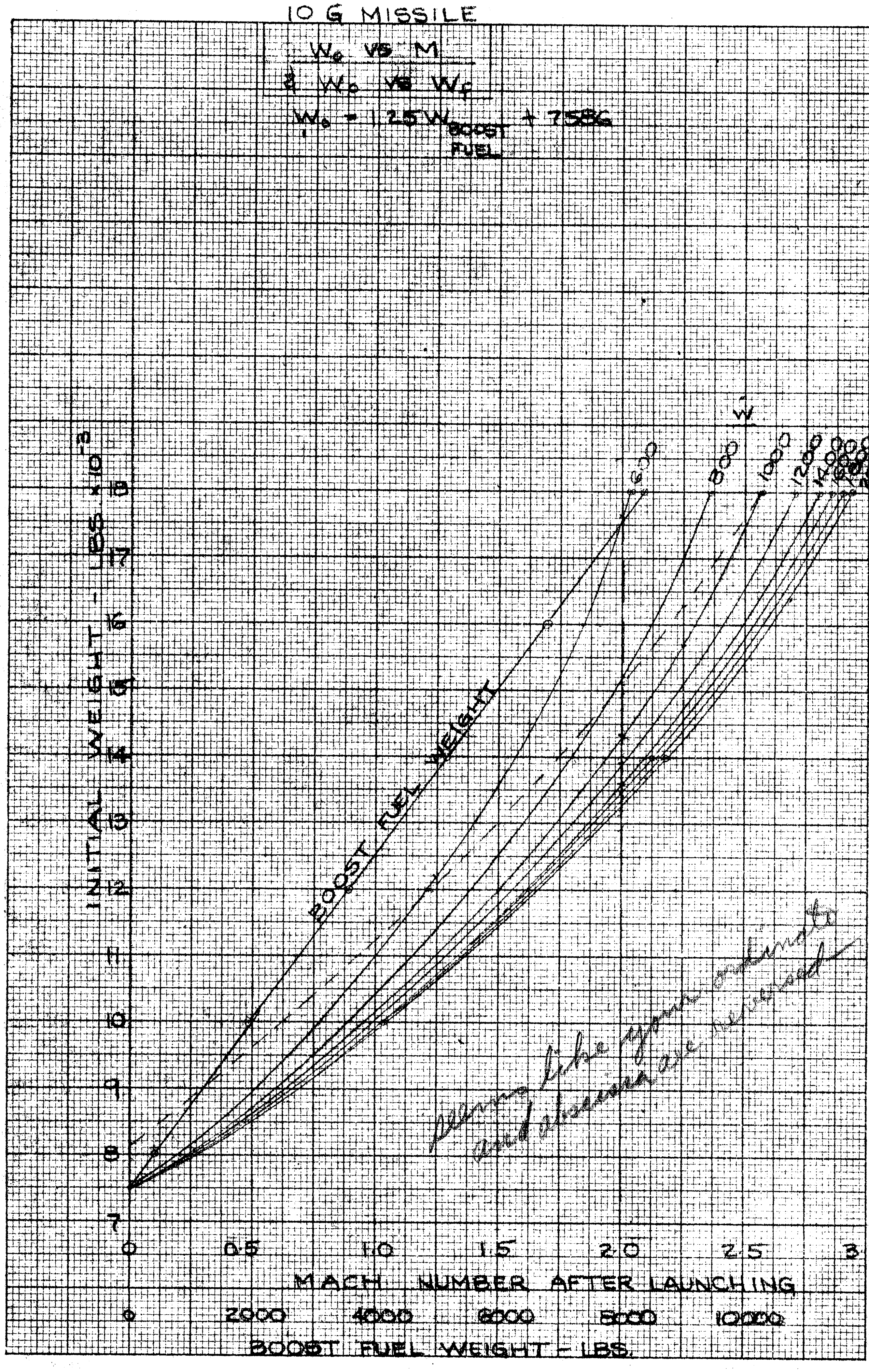


FIG. 6

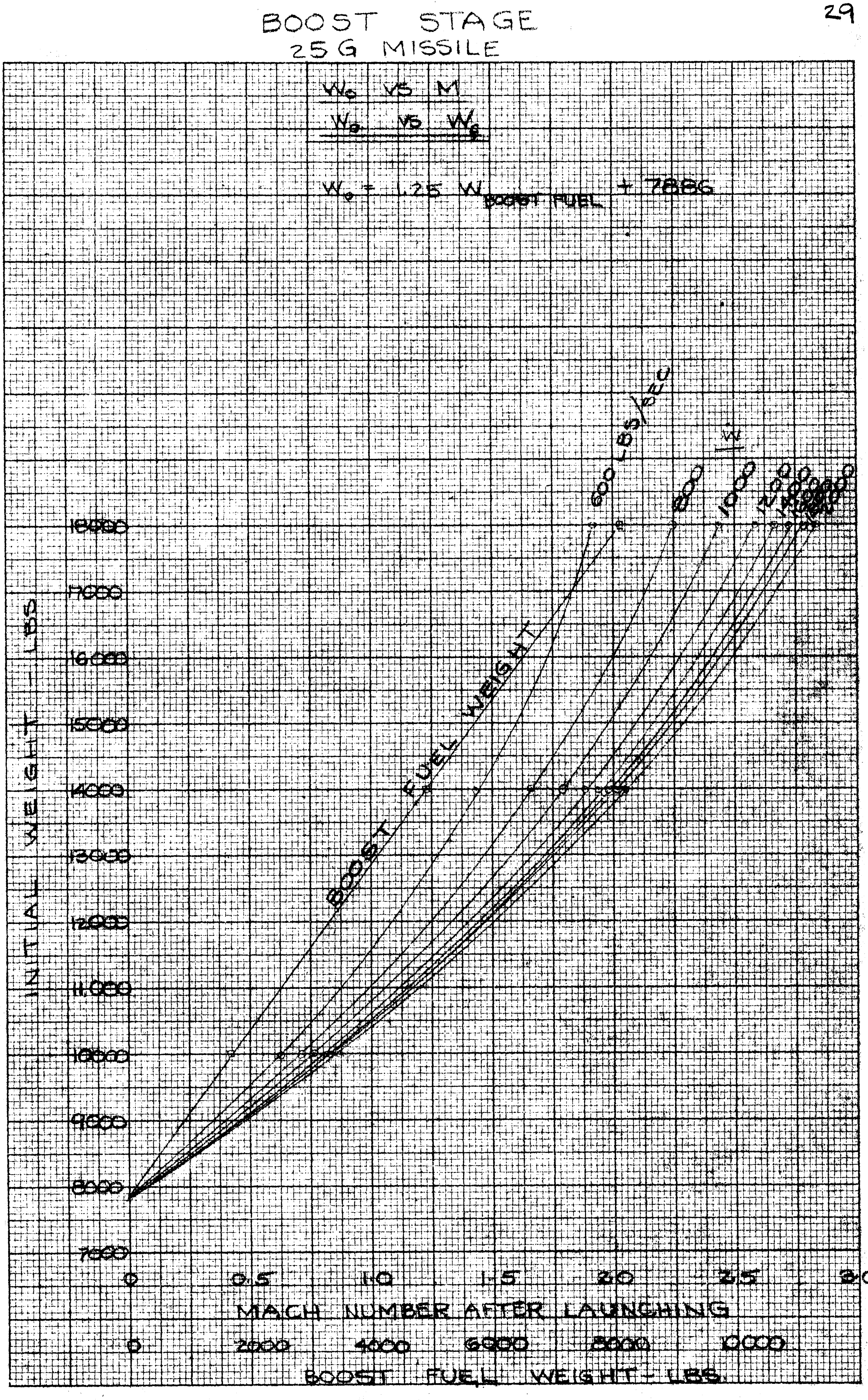

BOOST STAGE
25G MISSILE
W_o VS M
W_o VS W_f
$W_o = 1.25\ W_{BOOST\ FUEL} + 7886$
600 LBS/SEC
800
1000
1200
$\dot{W}$
BOOST FUEL WEIGHT
INITIAL WEIGHT - LBS
18000
17000
16000
15000
14000
13000
12000
11000
10000
9000
8000
7000
0
0.5
1.0
1.5
2.0
2.5
3.0
MACH NUMBER AFTER LAUNCHING
0
2000
4000
6000
8000
10000
BOOST FUEL WEIGHT - LBS

FIG. 7

BOOST STAGE
40 G MISSILE

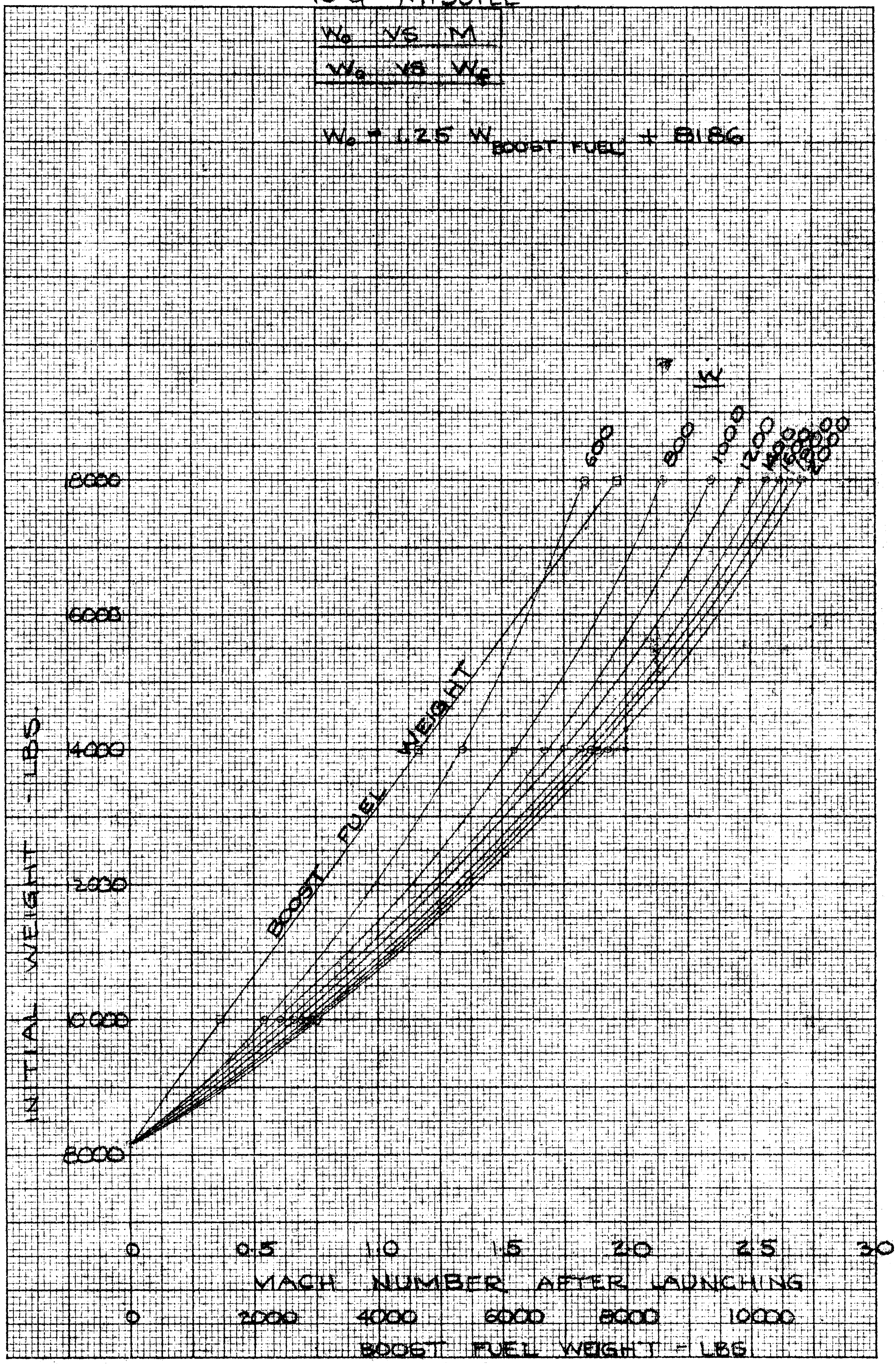
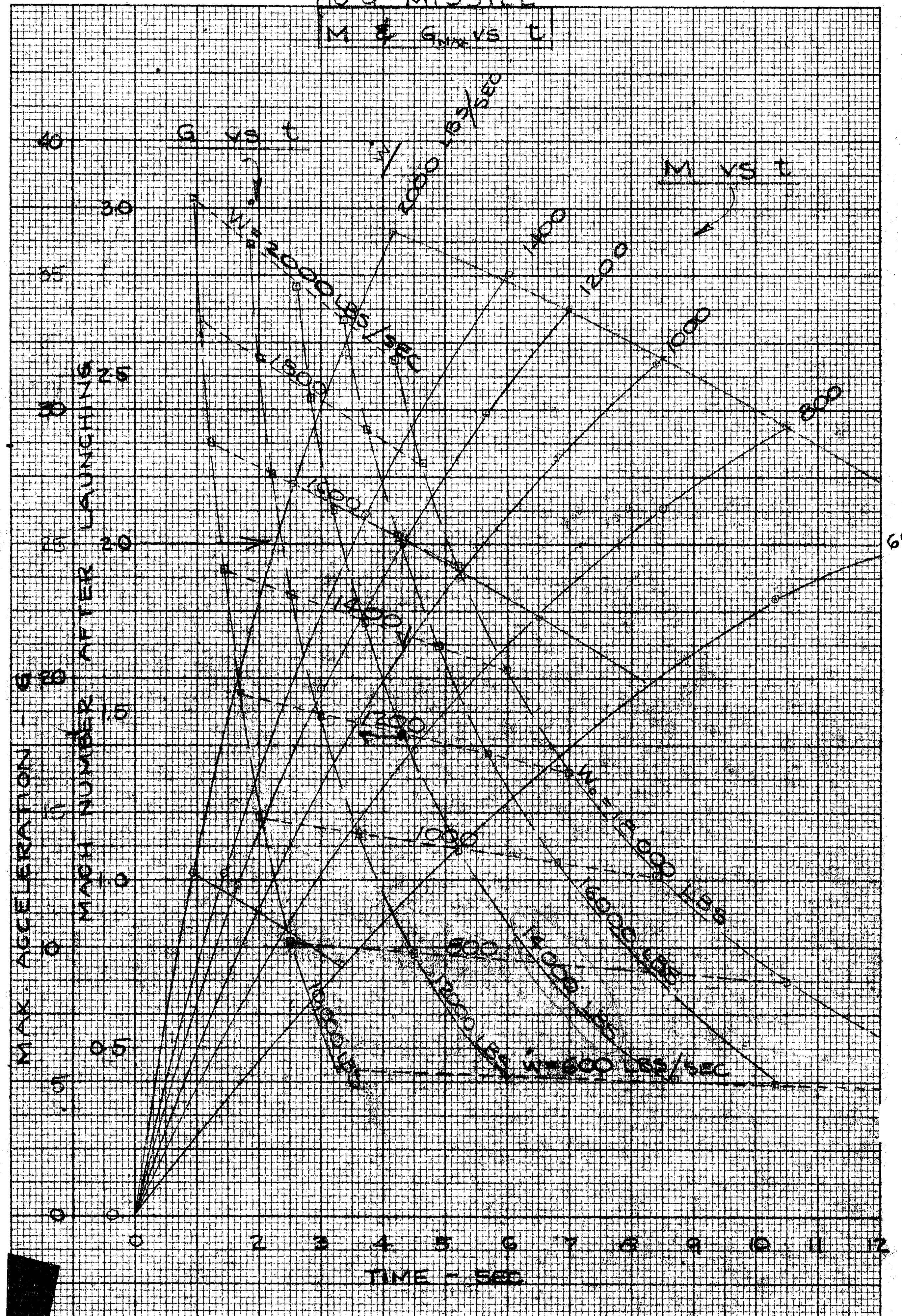


FIG. 8

BOOST STAGE
(10 G) MISSILE



EXPLAIN USE OF ↑

FIG. 9

BOOST STAGE
25 G MISSILE

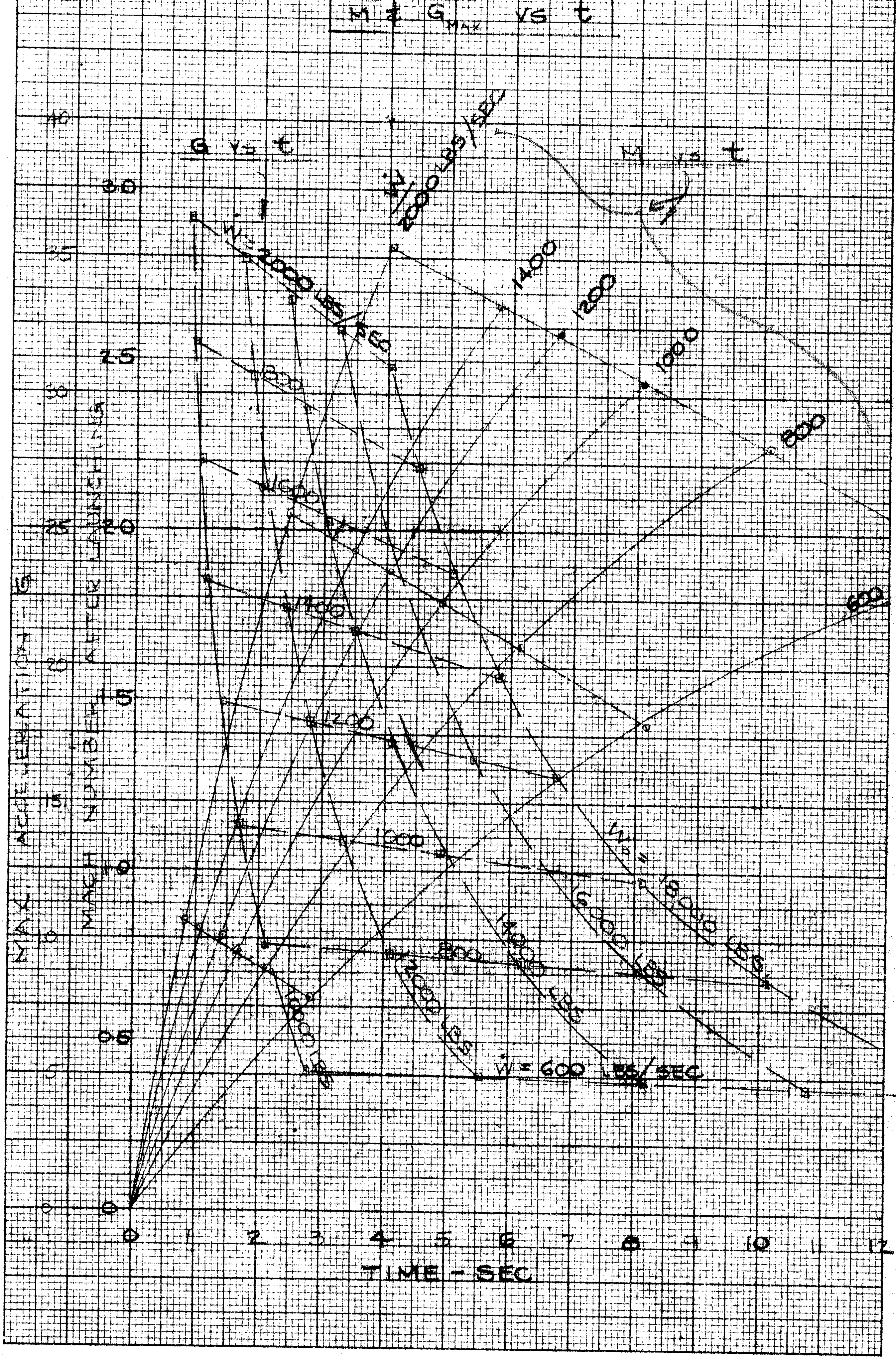


FIG. 10

BOOST STAGE

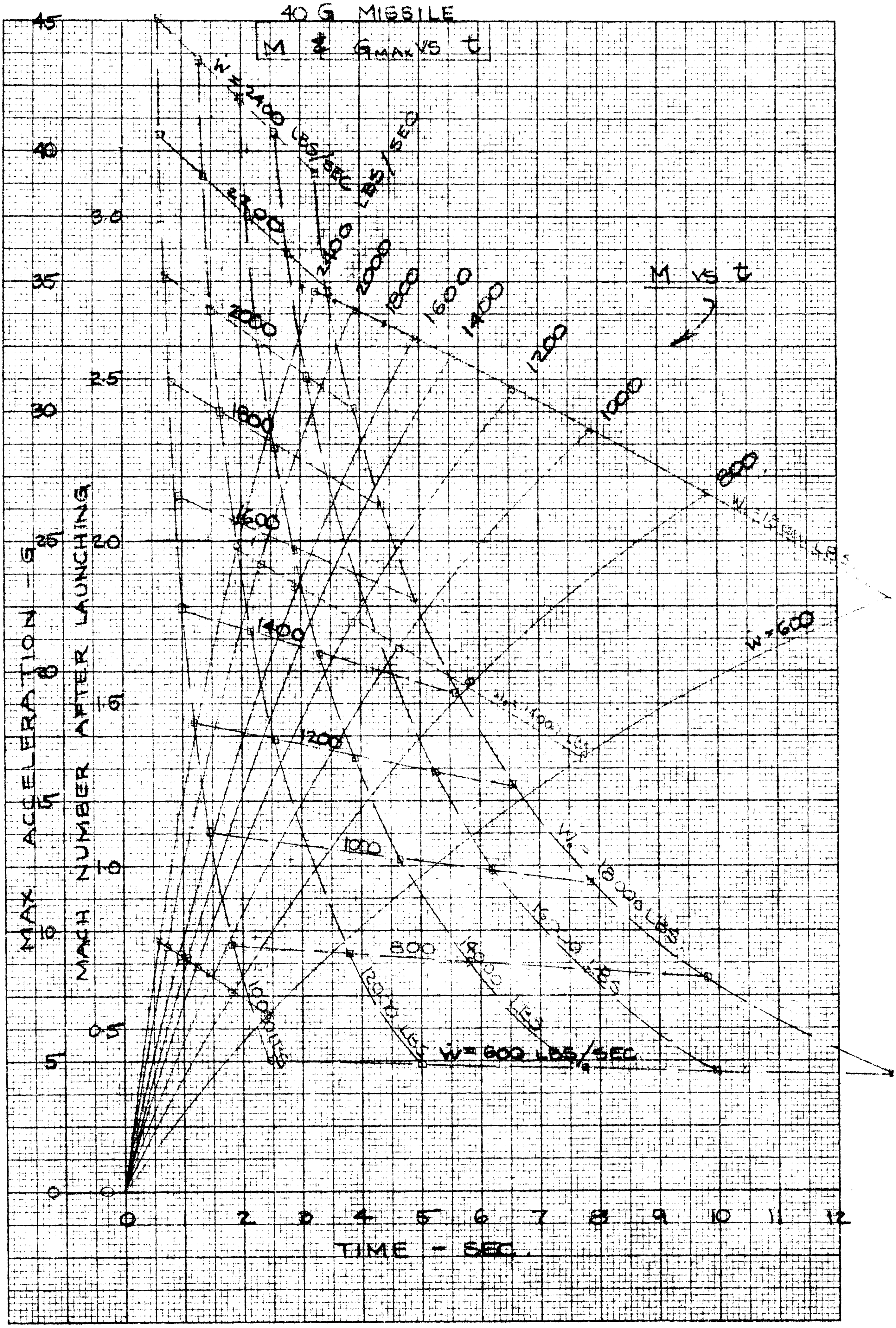


FIG. 11