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NAVAIR ltr dtd 26 Jun 1972; NAVAIR ltr dtd 26 Jun 1972

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NAVY DEPARTMENT

BUREAU OF AERONAUTICS

WASHINGTON, D. C.

DESIGN RESEARCH DIVISION

REPORT NO. 1030

AN ESTIMATION OF THE OPERATIONAL LIMITS OF PILOTLESS AIRCRAFT USING VARIOUS JET ENGINES

NOVEMBER 1947

FOREWORD

Due to an error in Design Research Report No. 1030, dated June 1947, it has been found necessary to replace completely the report by the following report.

ATI-9996

NAVAER

AN ESTIMATION OF THE OPERATIONAL LIMITS OF PILOTLESS AIRCRAFT USING VARIOUS JET ENGINES

DR REPORT NO. 1030

BuAer November 1947 Navy Dept.

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INTRODUCTION

This report was prepared in response to a request of the Power Plant Division of BuAer for a comparative study of five power plant types; turbo-jet, turbo-jet with afterburning, ram-jet, pulse-jet, and rocket, for use in pilotless aircraft. A less extensive study of this nature was prepared by Diels[1] of Marquardt Aircraft Company. In Diels' report no consideration was given to the aircraft design and one might conclude from the graphs that it is possible to construct a vehicle which could fly any prescribed distance at a given velocity and altitude. This may or may not be possible. Surely, there are upper bounds of attainment. Several other reports have been written upon the comparison of the types of engines for aircraft performance. However, most of them have not mentioned upper bounds, only relative bounds, some of which may be beyond the attainable limits. Driggs'[3] report on comparison of three types of engines for piloted aircraft is a notable exception. In his report he gives upper bounds of attainment for two types of aircraft.

Another thing has been common to all previous reports. The comparisons have been based upon a single engine within each type operating throughout the entire range. It is clear that the results based upon this hypothesis could not give the optimum regions. Practically, this might be sound from a development standpoint after it has been clearly demonstrated that the individual engines are the best for most tactical problems.

This report attempts to estimate upper bounds of attainment based upon certain optimistic design assumptions. In the analysis both the airplane and its engine have been tailored to fit the particular range and speed required. The aircraft and the power plant are assumed continuous functions of the velocity, range, and altitude. A change in any one of the latter variables produces a change in the vehicle and engine.

SUMMARY

This report deals with the comparison of the five types of engines as applied to the special problem of propelling an air vehicle in level flight at a <u>constant speed</u>. The five engines considered are the turbo-jet, turbo-jet with afterburning, ram-jet, pulse-jet, and liquid rocket. The study covers

the intervals of Mach number .5 to 3, range from 0 to 4,000 nautical miles, and altitude from 0 to 70,000 feet.

It is assumed that the vehicles are brought to the altitude and speed by an external booster rocket; the airframe is constructed so as to give its maximum lift/drag ratio at the velocity and altitude of flight; and there are an infinite number of engine units so that the power can be continually reduced as the weight is reduced by cutting out engines instead of throttling them. It is further assumed as indicated in the introduction that the engines are designed specifically for each speed, altitude, and range so as to give optimum obtainable performance.

From these hypotheses the percentage of the initial gross weight required for the engine, the airframe structure, the fuel, and the tanks are each determined for a series of Mach numbers, altitudes, and range. The sum of these four percentages are plotted as a function of the range in Figs. 1-35. Obviously, a mission cannot be accomplished in the prescribed manner when the sum of these percentages exceeds 100. In any case the percentage of payload is equal to 100 per cent minus this sum. (Payload consists of all other items in the aircraft other than tanks, fuel, engine, engine accessories, and structure). The aforementioned working graphs were employed to obtain upper bounds of attainment for the aircraft with various percentages of payload for each of the engines. The bounds are presented in Figs. 36 to 50. These charts were utilized to determine the best engine for a given region as shown in Figs. 51 to 60, where best engine means the engine which admits the least initial gross weight of the aircraft to accomplish the mission. From the graphs it is seen that the turbo-jet predominates the subsonic region with the ram-jet and pulse-jet close competitors in the supersonic region.

The effect of fuel/air ratio is very important in the performance of the airflow engines. Since the lowest fuel/air ratio for continuous combustion in the ram-jet is questionable, the computations were made for two fuel/air ratios, .01 and .03. The results show that the lower of these two fuel/air ratios gives the best performance.

The weights and the specific fuel consumptions of the turbo-jets and the turbo-jets with afterburning employed here are based upon studies made in the

Design Reeearch Division. The results will be published later in DR Report No. 1032[4]. The weighte for ram-jet are eesentially those given in Diels' report. (Spot checks indicate these values are quite optimistic). The weights of the pulse motor are comparable to those for the ram-jet, and the specific fuel consumption ie the optimum under the aesumptions outlined in this report. Rooket data were obtained from the specification for the RMI A6000C4 engine[5].

CONCLUSIONS

The results given here are goals and are truly beyond exieting development. Even with intensive development it ie not very probable that these upper bounds will be attained in the near future. Here it was assumed that all component parts have their peak performanoe throughout the operating time. This cannot be accomplished in practice. The aesumption of infinite number of engine units oan at best be approximated. The assumptions on engine weights and the low fuel ratios have a pronounced effect on the range; in case either cannot be aocomplished, the picture would be changed. Nevertheless, the working curves, Figs. 1-35, give a clear picture of the relative merits of each engine even if the upper bounds of range are several hundred milee too high.

Since the weights of the engines do have euch an important effect upon the maximum ranges, a more thorough study ie being made of the weighte of ram-jets. The valuse ueed here for the ram-jet are of the same order of magnitude as thoee in Marquardt's report; and there are indications that they are optimietic at high Mach numbers and may be in error by a factor as great as five or ten. However, the percentage of engine weight ie so small that a greater error would reduce the maximum range by only a small amount. (This amount is a function of the percentage payload, but 400 nautical milee is a rough figure). On the other hand, the weighte of the turbo-jet at low Mach numbers might be elightly greater than those required for an expendable engine.

It muet not be concluded that the rocket engine ie of little value for pilotless aircraft propulsion. The conditions of level flight at constant speed and rocket propulsion are not harmonious conditions. The other engines would be equally bad if the problem had epecified a wide range of operating speeds and altitudes, for then the ram pressure recovery would not be so efficient. Moreover, if the problem included the climb and acceleration portione of the flight path, the rocket would compare much more favorably. Furthermore, if the flight path would be included as a parameter in determining the optimum engine then the rocket may be the best for a greater number of situations.

It must be stressed further that while the rocket is known to operate under all of the conditions assumed, the other engines have not been tested at all of these conditions. If the ram-jet engine is to be employed, it will require a booster for initial launching. Hence, research on rocket motors should be continued.

RECOMMENDATIONS

Since the pulse-jet is definitely superior to the ram-jet in a given region and is a close competitor in the other regions, it is reoommended that more stress be placed upon the research of pulse motors.

It is further recommended that this study be continued to include the weight of the booster rockets and thereby evaluate the overall gross weight necessary for the vehicles to deliver a given payload. Then compare this with the gross weight of a rocket propelled vehicle which delivers the same payload to the same final point but travels along a course more suitable for the rocket performance.

SYMBOLS

a_o	velocity of sound at sea level
AR_e	equivalent aspect ratio
b	wing span - ft.
c	specific fuel consumption - lbs./hr./thrust horsepower
C	specific fuel consumption - lbs./hr./lb. of thrust
C_D	drag coefficient
C_T	thrust coefficient for rocket motor
c_p'	an average specific heat for gas at constant pressure
c_v'	an average specific heat for gas at constant volume
D	total drag of airplane - lbs.
e	airplane efficiency
f	equivalent parasite area - $(ft.)^2$

SYMBOLS (Cont'd)

g	acceleration due to gravity - ft/sec.^{2}
h_i	specific enthalpy of gas (i = 1, 2, 3, 4) - BTU/lb.
H	heating value of fuel - BTU/lb.
J	mechanical equivalent of heat
L	total lift of airplane - lbs.
M	free stream Mach number
M_o	free stream Mach number
N_c	number stages of compressor in the turbo-jet engine
N_T	number stages of turbine in the turbo-jet engine
P_o	ambient pressure - lbs./(in)^2
P_i	total pressures (i = 1, 2, 3, 4) - lbs./(in)^2
P_c	combustion chamber pressure of rocket motor - lbs./(in)^2
P_r	relative pressure (base at 400°R)
R	range - miles
R_1	gas constant (pre-combustion)
R_2	gas constant (post-combustion)
s	specific entropy - (BTU/lb./°R)
S	wing area - (ft.)^2
T	thrust - lbs.
t_i	total temperature (i = 1, 2, 3, 4) - °Rankine
u_i	specific internal energy of gas (i = 1, 2) - BTU/lb.
V	velocity - mph
v	velocity - ft./sec.
v_j	jet velocity - ft./sec.
W	gross weight - lbs.
W_o	initial gross weight - lbs.
W_1	final gross weight - lbs.
W_a	weight of air for air flow engines - lbs/sec.
W_E	weight of engine - lbs.
W_F	total weight of fuel - lbs.
W_f	fuel consumption - lbs./sec.
W_S	weight of structure - lbs.

SYMBOLS (Cont'd)

W_T weight of fuel tanks - lbs.

σ ratio of density of air at altitude to the density at sea level

δ ratio of pressure of air at altitude to the pressure at sea level

θ ratio of temperature of air at altitude to the temperature at sea level

γ ratio of specific heat of gas at constant pressure to specific heat at constant volume

η_v valve intake efficiency

η_c combustion efficiency

η_D diffuser efficiency

F/A fuel/air ratio

γ' ratio of average specific heat of gases at constant pressure to average specific heat of gases at constant volume

ANALYSIS

A. Aerodynamics

The generalized drag curve (6) as obtained from the approximation of an airplane polar by a parabola is

$$(1) \qquad D = .00256\, f\sigma V^2 + \frac{124.8\left(\frac{W}{be}\right)^2}{\sigma V^2}$$

where V is statute miles per hour, f is the equivalent parasite area, W is the aircraft weight, b is the wing span, and e is the airplane efficiency. This relation may be utilized to obtain the ratio of the thrust to the gross weight required for flight. The substitution $\sigma = \frac{\delta}{\theta}$, $f = C_D S$, $(be)^2 = S\, AR_e$ and $V = a_o \sqrt{\theta}\, M$ (a_o the velocity of sound at sea level in miles per hour) and the division of both sides of the equation by W yields

$$(2) \qquad \frac{T}{W} = \frac{1{,}487\, C_D M^2 \delta}{\frac{W}{S}} + \frac{.000215\, \frac{W}{S}}{AR_e M^2 \delta} \text{ which holds for subsonic flow}$$

The corresponding formula for supersonic flow is

(2') $$\frac{T}{W} = \frac{1,487\ C_D M^2 \delta}{\frac{W}{S}} + \frac{.0001685\ W/S}{M^2 \delta} \sqrt{M^2 - 1}$$

This study requires the determination of the maximum range attainable. For this work it is suitable to utilize Breguet's formula

(3) $$R_{(\text{statute miles})} = 863.5 \frac{1}{c} \frac{L}{D} \log_{10} \frac{W_o}{W_1}$$

where c is the specific fuel consumption in lbs. per hour per thrust horsepower, W_o is initial gross weight, and W_1 is the final gross weight or $W_1 = W_o - W_f$, and L/D is an average value determined by the mean value of W/S. When written in terms of the variables employed in this report it becomes

(4) $$R_{(\text{nautical miles})} = -1523 \frac{L}{D} \frac{\sqrt{\theta} M}{C} \log \left(1 - \frac{W_F}{W_o}\right)$$

where R is now in nautical miles and C is the lbs. of fuel per hour per lb. of thrust, and $\frac{W_F}{W_o}$ is the ratio of the fuel to initial gross weight. Under the hypothesis of this study; namely, level flight at a constant velocity, the specific fuel consumptions of the engines are determined. Hence, from equation (4), the maximum range will be obtained when L/D is a maximum. Or, in other words, T/W must have a minimum value. Equation (2) shows that T/W (for constant speed and altitude) is a function of the three variables, C_D, W/S and AR_e. These three variables are not independent. Nevertheless, within the range of practical construction limits there exists at least one set of these values which gives T/W its minimum value. The exact determination of the values cannot be obtained precisely without making detail designs. Here only estimates of these quantities were made. After careful observation of test results and theoretical calculations, the drag curve as a function of Mach number given in Fig. 61 was assumed to be the best attainable. In order to obtain this drag coefficient the angle of sweep of the wing, the wing section and the planform of the missile were all varied with the Mach number. In other words, the curve does not represent the coefficient of one configuration, but an envelope of coefficients for a series of configurations. Moreover, in order to eliminate the dependence of C_D on scale effects, it was assumed that the missile is a flying wing. Next, the effective aspect ratio of all the missiles was chosen to be either 4 or 8; 8 at subsonic Mach numbers and 4 for

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all other conditions. It is obvious from equation (2) that other things being equal, the greater the aspect ratio, the smaller the value of T/W. Unfortunately, other things do not remain equal. Not only does the aspect ratio have an effect upon C_D but also upon the structural weight. As a compromise, the above values were chosen.

It was assumed that average values of W/S lie in the interval from 10 to 200. After the values of C_D and AR_e were determined, then W/S was taken to be the value in the above interval which makes T/W a minimum. The values in the interior of the interval are determined from the relations

(5) $W/S = 2630\, M^2 \delta \sqrt{C_D\, AR_e} \quad (M < 1)$ (5') $W/S = \dfrac{2975\, M^2 \delta \sqrt{C_D}}{(M^2 - 1)^{.25}} \quad (M > 1)$

which ~~were~~ obtained by equating to zero the derivative of T/W with respect to W/S. (This is somewhat in error since it was assumed that neither C_D nor AR are functions of W/S). When the value of W/S was greater than 200 or less than 10, the end values 200 and 10 respectively were taken.

After the minimum value of T/W and the specific fuel consumption of the engines have been determined, the percentage of fuel weight of the initial gross weight required for the various ranges is computed from equation (4). The weight of the fuel tanks is assumed to be 12 per cent of the fuel weight.

The percentage of the total weight required for structure was assumed to depend only upon the wing loading. According to a statistical and analytical study of wing weights by Kelley[13] the percentage of structural weight is inversely proportional to the wing loading to the .21 power. Since this relation seemed to check very closely with other structural studies,[15] it was adopted here, and the constant of proportionality was determined so that $\frac{W_S}{W_o} = .25$ when $W/S = 100$. Or stated in algebraic form it was assumed that

$$\frac{W_S}{W_o} = .657 \left(\frac{W_o}{S}\right)^{-.21}$$

B. Engines

I. General considerations.

This section outlines the considerations which are common to a majority of the engines.

1. The size of the engine is determined by the thrust required. The ratio of the weight of the engine to the thrust delivered $\frac{W_E}{T}$ is determined for the specific engines. The product of this quantity and the thrust over initial gross weight required gives the fractional portion of the initial weight required for the engine

$$\frac{W_E}{T} \cdot \frac{T}{W_o} = \frac{W_E}{W_o} .$$

2. It is assumed that the minimum specific fuel consumption is maintained throughout the flight. Since in the aerodynamic consideration it was assumed that the minimum value of T/W is maintained throughout, the thrust must continually reduce as the fuel is consumed. This requires throttling or cutting off some engines which cannot be accomplished without increasing the specific fuel consumption because the engine operation is considered only at the optimum point. So to satisfy this hypothesis, it is assumed that there is a continuum of engines all operating at the ideal point and that the power is reduced continuously by cutting off engines. The weight of the engines is carried throughout the flight.

3. The assumptions made regarding the ram pressure recovery were the same for all airflow engines. At subsonic speeds the total pressure recovery was assumed to be 98% of the ram. At supersonic speeds the total pressure recovery was based upon the assumption of an Oswatitsch diffuser, which produces two oblique shocks followed by one normal shock. The resultant total pressure head given under this assumption in reference (16) was reduced by 2% to allow for the loss in the subsonic part of the diffuser.

4. It is assumed that the jet velocity of all engines is .98 per cent of the theoretical attainable when

the expansion is to atmospheric pressure. In order to achieve this it is necessary to have convergent-divergent nozzles for most regions of operation. In cases where there is an increase in projected area resulting from nozzle divergence, it is assumed that the additional drag is included in the drag coefficient. This has an equalizing effect on the drag for the configurations for the various engine types.

II. Specific Engine Hypothesis

TURBO-JET

The turbo-jet engines referred to in this report are theoretically possible engine designs (only axial flow compressor considered) as determined by DR Report No. 1032[4]. The object of this latter study was to find the optimum combination of compression stages and top temperatures for turbo-jets operating at various altitudes, speeds, and ranges; optimum being defined as that which will result in a minimum total fuel plus engine weight required to meet a prescribed set of operating conditions, altitude, speed, and range (or time).

The uniqueness of this latter report lies in the fact that time (or range) is one of the design parameters. It is evident that for short ranges the engine weight is the prime factor to be taken into consideration since total fuel consumption is comparatively small; while for long ranges the converse is true. Since the number of compression stages (and hence the compression ratio) is one of the important parameters of specific fuel consumption and engine weight, performance and weight calculations were made for engines containing 0-20 compression stages. These calculations were made for velocities of $\frac{V}{\sqrt{\theta}}$ = 150 to 750 miles per hour. The same procedure was employed here to extend the results to a Mach number 3 such that they are applicable to any present or near future problems to be fulfilled by turbo-jet powered vehicles.

By adding the engine weight of a particular design to the weight of fuel required for this engine to operate for a specified time under specified conditions, and comparing this result with other designs operating for a like time and under similar conditions, it was possible to determine which engine met the "optimum" requirement. By varying flight times as well as operating conditions, several series of points were established and plotted. From these graphs the engines for this study were selected. The fuel consumption is computed according to the methods of the referred report. The number of compression stages required decreased with Mach number until it reduced to zero or, in other words, until the engine became a ram-jet. This always occurred before $M = 3.0$ was reached. The engine is called a turbo-jet so long as it has at least one compressor and turbine stage.

The basic assumptions in calculating the performance characteristics of these engines are as follows:

(a) Compression ratio is equal to 1.2 per compressor stage.

(b) Compression ratio across the turbine is .665 per turbine stage.

(c) The small stage efficiency is .90 for both the compressor and the turbine.

An additional assumption employed herein is that of metallurgical limitations. The limitations are the following: those engines operating for a period of sixty minutes or less were limited to a top temperature of 2000°R; while those operating for periods of longer than sixty minutes could not exceed a top temperature of 1600°R.

The fuel/air ratios for these engines varied within the approximate range of .005 to .045, dependent upon the enthalpy rise in the combustion chamber required to meet the conditions set by the assumed operating temperatures. It was assumed the weights could be determined by an equation of the following form:

$$W_E = 3.54\, N_C \left(W_A \frac{\sqrt{\theta}}{\delta}\right)^{.64} + \left(W_A \frac{\sqrt{\theta}}{\delta}\right)\left(3.71\, N_T + 6.09\right) + 64.1 \left(W_A \frac{\sqrt{\theta}}{\delta}\right)^{.224}$$

where N_C and N_T represent the number of compreesor stages and number of turbine stages, W_E the weight of the engine, and W_A the airflow through the engine in pounds per second. The constants used were determined empirically and are based upon engines designed to operate for many hours at low speeds. Hencc, the weights may be heavier than are required for expendablc engines which operate for a short period of time at subsonic speeds.

TURBO-JET WITH AFTERBURNING

The deeign data of the turbo-jets with afterburners used herein are also taken from DR Report No. 1032. The method used for determining these engine designs is the same as for the turbo-jet.

Basis for performence and weight calculations are essentially the eame as for the turbo-jet plus e few additions. They are:

(a) The ratio of the temperature increment produced by the afterburner is to the combustion chamber temperature as 900 is to 2600, and the total entropy loss in the afterburner is assumed to be .0028 BTU/lb./°R.

(b) The fuel/air ratios for these engines were higher than those of the turbo-jet because of the additional fuel required in the afterburner and varied from approximately .0075 to .065. NOTE: The afterburner ie assumed to operate during the entire flight.

(c) The added weight of the afterburner ie equal to three times the weight of the engine airflow per second.

RAM-JET

The efficiencies of the engine components other than thoee for the diffueer, which have been previously coneidered, will now be discussed.

Combustion Chamber The combustion efficiency is assumed to be 100%; that is, the BTU's/lb fuel imparted to the air stream is equal to the heating value of the fuel (18,700 BTU/lb. fuel). However, the pressure losses due to the combustion processes; i.e., frictional and momentum losses, were estimated from reference (40).

No assumption has been made regarding combustion chamber inlet speed except that the latter and the prescribed fuel/air ratios do not result in choking within the combustion chamber. Thus, one of the design parameters would be the fulfillment of the above condition.

Nozzle It is to be noted that a convergent-divergent nozzle is essential in the region of operation for the ram-jet ($M = 2.0 - 3.0$) since an underexpanded nozzle will result in a lose of thrust ~ 15% which is many times the loss in thrust produced by the increased drag created by a convergent-divergent nozzle.

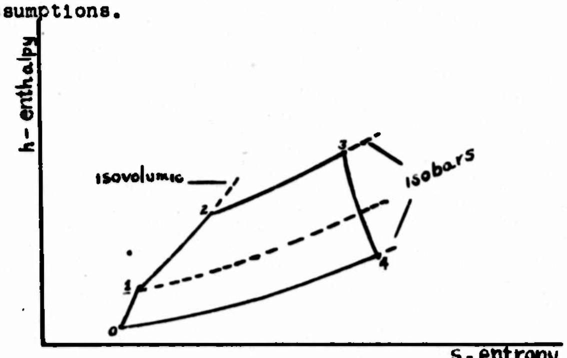
The estimated weights per lb. of thrust of the ram-jet engines considered were taken from reference (1). However, a spot check of the weights indicated that the estimated weights per lb. of thrust are too large by a factor of 2 in the subsonic region and too small by 3-10 times in the supersonic region. The latter figure of "10" is applicable to the weights per lb. of thrust equal to ~ .005. The resultant error of the weights used may be offset here by the assumed value of 12% for the tankage factor (weight tank = 12%) of weight fuel. Even if the last point is neglected the error in estimated weight results in an error of ~ 200 - 400 miles for the obtainable range.

It was found that the weight of the ram-jet engine is negligible in comparison with the fuel weight required even for short operating times. Therefore, the optimum engine is that engine which has the lowest specific fuel consumption. From the analysis made the thrust specific fuel consumption as a function of fuel/air ratio was found to be a continuously increasing function between the F/A values .01 - .04 at all operating conditions. As the basis required to determine the lower limit of the fuel/air ratio which can sustain combustion does not exist and the lowest fuel/air ratio used gives the lowest specific fuel consumption, which in turn results in the optimum engine, two ratios were assumed for this study; namely, .01 and .03. The latter is an approximate existing lower limit, and the former is an anticipated value. The justification for considering this as an anticipated value is the experimental work of Pabst[20], which indicated normal combustion at .01 with combustion chamber inlet speeds up to 450 ft/sec.

With the assumptions outlined in the above paragraphs, the values of specific fuel consumption as a function of Mach number for different altitudes were found and are given in Figs. 63 and 64.

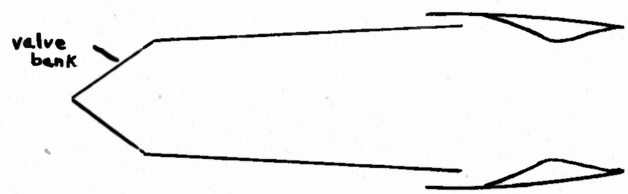
PULSE-JET

The pulse-jet engine was treated here on the basis of a thermodynamic cycle simulating pulse-jet operation (see Fig. 1); that is, constant volume, constant pressure combustion. To date, two types of theoretical treatments exist: (1) theories tailored to satisfy the data on the German V-1; (2) sophisticated treatments not readily applicable to a study of this type. The calculated performance is only as valid as the assumptions made. The method used for the calculations will be considered in detail after a discussion of the assumptions.



Cycle for Pulse Jet

The pulse-jet as considered here consists of a pressure recovery device, valve bank, and duct. In the superson'; region the engine i shrouded completely and, therefore, operates in a subsonic medium whose pressure is ram pressure. The duct geometry must conform to assumptions made herein; however, the general shape is a conical valve bank followed by a continuously diverging tube. To avoid velocities greater than sonic at the nozzle exit at high subsonic flight speeds and still obtain complete expansion, an auxiliary nozzle is required (see accompanying sketch).



Schematic Diagram of Partially
Shrouded Pulse Jet.

Assumptions

1. The spring constants of the valves can be modified for each altitude and speed so that no leakage through the valves occurs.

2. The ratio of peak combustion chamber pressure to free stream pressure is given in Table 1 where in the supersonic case P_1 is the apparent pressure and is, therefore, the ram pressure.

Table 1. Pressure Ratios for Pulse Jet

Mach No.	P_2/P_o	P_2/P_1
.50	5	
.85	5	
1.0		3.8
1.5		3.5
2.0		3.0
3.0		2.5

The reduction of P_2/P_1 at supersonic speeds is to avoid excessive pressures and temperatures in the combustion chamber.

It is felt that the possibility of obtaining the ratios given in Table 1 will be a reality when decreased burning time is accomplished.

3. The efficiency of intake of the air passing through the valves; i.e., resultant total pressure is assumed to be as follows:

Table 2. Pressure Efficiency of Valve Intake.

Mach No.	η_v
.50	.80
.85	.75
1.0	.72
1.5	.70
2.0	.68
3.0	.65

These efficiencies separate from the diffuser efficiencies, η_D , which are common to all the engines, as given in Section B.1.

4. The fundamental assumption made here is with regard to the combustion efficiency, η_c ; that is, the ratio of BTU's obtained per lb. of fuel to the heating value of the fuel. Experimental data are conspicuous by their absence; therefore, estimates were made that are seemingly in the right direction.

As defined here the combustion efficiency determines the operating fuel/air ratio for the engine. The excess of fuel above the theoretical fuel/air ratio is consumed in the engine by either reheating the combustion products or remaining unburned. On the basis outlined here, fixed p_2 and t_2, increasing t_3 along constant pressure line, indicates increased burning time and, therefore, a decrease in combustion efficiency. Some representative values of combustion efficiency used here are given in Table 3.

The available experimental performance data for several types of pulse-jet engines were compared with the theoretically computed performance based on combustion efficiencies comparable to those listed above and good agreement was obtained (1% deviation).

It is further assumed that combustion at all flight conditions can be sustained at fuel/air ratios from .03 to .05. The latter is found to be possible experimentally. The former fuel/air ratio occurs in this study only at supersonic flight speeds with the air stream

Table 3. Sampls Values of Combustion Efficiencies for Pulse Jet.

Mach No.	Peak Temperature °R	Combustion Efficiency Sea Level	25,000	35,000	50,000	70,000
.5	3000	55	50	45	38	30
	3300	48	42	36	28	20
.85	3000	65	60	58	50	45
	3300	60	53	47	40	35
1.0	3000	75	72	70	65	63
	3300	65	60	55	48	43
1.5	3000	80	77	75	70	65
	3300	75	70	66	60	55
2.0	3500	95	92	90	88	85
	4000	90	87	85	80	75
3.0	4000	95	95	90	88	88

properties being $t_1 \sim 800 - 1400°R$ and speed $v_1 \sim 300$ ft/sec. Thus, it is safe to assume that combustion under these conditione can be sustained at a fuel/air ratio = .03.

At subsonic speeds the weight of engine per lb. of thrust was obtained by an extrapolation of SR500A[38] data, using the above efficiencies. A further extrapolation (in proportion to ram-jet estimates) was made for supersonic speeds. The proportionality factor wae the ratio of combustion chamber pressuree of the respective engines.

As with the ram-jet the weight of the pulse-jet engine ie negligible in comparison with fuel weight required even for short operating times. Therefore, the optimum engine is that engine which hae the lowest epecific fuel consumption.

Since these curves differ from the usual onee presented, wherein the specific fuel consumption for various speeds and altitudes remains constant, a short explanation is given. First, precise data of s.f.c. variation with altitude are non-existent and, therefore, any theoretical calculations should consider the pulse-jet as an engine which operates best (if at all) in high density air. Secondly, the variation of specific fuel coneumption with speed would show a decrease in C if the design point of the valves is changed continuously. As a further aid to the reader, a brief summary of the performance method in step-by-step form is now given (consult Fig. 1 for cycle representation).

Step 1: Find the conditions at point 1.

$$\frac{t_1}{t_o} = \left(1 + \frac{\gamma - 1}{2} M_o^2\right) \tag{1}$$

where $\gamma = 1.4$ and t_o is free stream static temperature.

$$\frac{p_1}{p_o} = \left(1 + \frac{\gamma - 1}{2} M_o^2\right)^{\frac{\gamma}{\gamma - 1}} \tag{2}$$

where $\gamma = 1.4$ and p_o is free stream static pressure.

$$(p_1)_{act.} = (p_1)_{isentropic}\ \eta_V \cdot \eta_D \tag{3}$$

The values of h_1 and u_1 are obtained from Ref. (18).

Step 2: From point 1 to point 2 a constant volume combustion process is assumed; therefore, the changes in the internal energy should be considered. The expression for this change is:

$$\Delta u = u_2 - u_1 = c_v' (t_2 - t_1) = c_v' t_1\left(\frac{t_2}{t_1} - 1\right) \tag{4}$$

$$= \frac{c_p'}{\gamma'} t_1\left(\frac{t_2}{t_1} - 1\right)$$

or

$$\Delta u = u_2 - u_1 = \frac{c_p'}{\gamma'} t_1\left(\frac{R_1}{R_2}\ \frac{p_2}{p_1} - 1\right) \tag{5}$$

(See Ref. (17) for values of R_1 and R_2).

From the assumed values of P_2/P_1 given in Table 1, Δu can be computed. The value of h_2 and t_2 can be found (see Ref. (18).

Step 3: From point 2 to point 3 a constant pressure combustion process is assumed. Further, by fixing the peak temperatures, t_3, h_3 can be found from Ref. (18) and corrected to account for combustion products.

Step 4: From point 3 to point 4 complete expansion is assumed; i.e., $p_4 = p_o$.

h_4 can be found directly from the air charts in the following manner.

From t_3, a value of Pr_3 is found where Pr_3 is the pressure ratio

$$Pr_3 = \left(\frac{T_3}{400\,^oR}\right)^{\frac{\gamma}{\gamma-1}}$$

which includes the variation of γ with temperature.

Therefore,

$$\frac{P_4}{P_3} = \frac{Pr_4}{Pr_3}$$

or,

$$Pr_4 = \frac{P_4}{P_3} Pr_3 = \frac{P_o}{P_3} Pr_3 \qquad (6)$$

With the value of Pr_4 given, h_4 is found from the air charts.

The average jet velocity is then given by the expression:

$$(\bar{V}_J)_{theo.} = \sqrt{2gJ(h_4 - h_3)} \qquad (7)$$

and since this process is 98% efficient

$$(\bar{V}_J)_{actual} = .98\sqrt{2gJ(h_4 - h_3)}. \qquad (8)$$

Step 5:

The fuel/air ratio is found from the expression:

$$\left(1 + \frac{W_a}{W_f}\right)\left((h_3 - h_2) + (u_2 - u_1)\right) = \eta_C H \qquad (9a)$$

or,

$$\frac{W_a}{W_f} = \frac{\eta_C H}{(h_3 - h_2) + (u_2 - u_1)} - 1 \qquad (9b)$$

From whence one obtains

$$\frac{W_f}{W_a} = \frac{(h_3 - h_2) + (u_2 - u_1)}{\eta_C H - [(h_3 - h_2)+(u_2 - u_1)]} \cong \frac{(h_3 - h_2) + (u_2 - u_1)}{\eta_C H} \qquad (10)$$

when $\eta_C H >> (h_3 - h_2) + (u_2 - u_1)$

Step 6:

The specific fuel consumption can now be derived.

$$C = 3600 \frac{W_f}{T} = 3600 \frac{W_f}{W_a} \Big/ \frac{T}{W_a} \qquad (11)$$

where T/W_a is the resultant of steps $1 \rightarrow 4$ and

$$\frac{T}{W_a} = g \left[\left(1 + \frac{W_a}{W_f}\right) v_j - v_o \right] \qquad (12)$$

where W_a = lb./sec.

To find the absolute values of thrust, fuel flow, and airflow, steps 7 and 8 are given.

Step 7:

The total fuel consumed/cycle is

$$\frac{\text{lb.fuel}}{\text{cycle}} = W_A \cdot \frac{W_f}{W_a} \quad \text{where } W_A = \text{lb./cycle} \qquad (13)$$

where W_f/W_a is defined by equation (10).

Then,

$$\frac{\text{lb.fuel}}{\text{sec.}} = \frac{\text{cycles}}{\text{sec.}} \cdot \frac{\text{lb.fuel}}{\text{cycle}} \qquad (14)$$

The airflow (W_a) must be estimated separately for each engine. A good first approximation can be made as follows:

(a) find total volume of engine

(b) take 1/7 of this volume and find the weight of the air that occupies this volume at a pressure equal to ram pressure. This is indicated by the work of Schmidt as reported in Ref. (38) as the quantity of air participating in the first charge.

(c) The airflow (lb./cycle) is then .4 of the value of the air weight given in (b). The quantity (.4) is approximately the ratio of the area of one air intake portion of grill to combustion chamber cross sectional area and is found to be on the average (.4) for all existent engines.

Step 8:

From steps 1 - 4 one obtains

$$\text{Thrust} = \frac{W_a}{g}\left[\left(1 + \frac{W_f}{W_a}\right) v_j - v_o\right] \tag{15}$$

where the quantities v_o is given, v_j is computed and $W_f/W_a = .05$. The latter estimation results in a small error for the values of thrust since W_f/W_a varies from .03 to .05 in this study. The computed value of W_a is substituted into equation (11) and the thrust is obtained.

If experimental data are available; i.e., thrust and fuel flow, combustion efficiencies can be found immediately with the use of the estimated airflow and the experimental value of s.f.c. should equal the theoretical one found by Step 6.

ROCKET

Since low velocities are unsatisfactory conditions for rocket propulsion, the first estimates shown in Figs. 1-50 for Mach No. ≤ 1 are not based upon an optimized engine but upon an existing liquid rocket RMI A6000 C[4] (alcohol and oxygen fuel system) where the increase in thrust due to altitude was given due consideration. At velocities greater than Mach No. 1, in the construction of Figs. 51-60, which give the best engine for various speeds and ranges, optimized rocket engines employing liquid hydrogen and liquid oxygen fuel were used.

Illustrative Examples of Use of Charts

Given Conditions

Altitude - 25,000 ft.

Speed - Constant M = 1.3

Range 1700 nautical miles

Problem 1: Find Optimum engine

Solution: (a) From graph No. 57 it is seen that the coordinates of the range and Mach No. for the given conditions intersect in the region of the ram-jet whose F/A = .01. (b) If, however, the fuel/air ratio .01 cannot be attained, then one must enter graph 52. Here the pulse-jet is the optimum engine for the given conditions.

Problem 2: Find Minimum Gross Weight (after reaching the prescribed flight conditions) of vehicle to carry a payload = 2400 lbs.

Solution: (a) If solution 1a is used; i.e., ram-jet engine, then interpolate between graphs 39 and 40 and find that the maximum payload percentage is 12.5. Therefore, the minimum gross weight (as defined in problem) is $\frac{2400}{.125}$ = 19,200 lbs.

It is to be noted that in this series of graphs those which represent payload percentage other than 30% were put on transparencies for ease in interpolating or extrapolating. (b) If solution 1b is preferred; i.e., pulse-jet engine, use of the same graphs (Nos. 39 and 40) indicate a maximum payload percentage of 6.1 and, therefore, a minimum gross weight of $\frac{2400}{.061}$ = 39,350 lbs.

Problem 3: Find maximum payload if gross weight of vehicle is limited to 19,200.

Solution: (a) Use of graphs 39 and 40 again show the same allowable payload percentages as before; 12.5% if the ram-jet whose F/A = 0.01 is used, and 6.1% if the pulse-jet is used. From these figures the maximum payload is found to be (.125) (19,200) = 2,400 lbs. for the ram-jet and (.061)(19,200) = 1,170 lbs. for the pulse-jet.

Since factors other than minimum weight must be taken into consideration, (launching problems, availability of materials and tools for production, economy, etc.) care should be taken before a final engine choice is made. It is, therefore, advisable to evaluate the relative merits of different engine types and compare these with those of the engine chosen on the basis of minimum weight or maximum payload estimation alone. Then, if it is found that several different engine types will give similar performance, the final choice will depend upon factors other than those on which this study is based.

For instance, the flight requirements of the problems may also be met by the turbo-jet engine. Interpolation between graphs 39 and 40, once again, indicates that the turbo-jet can meet the requirements of range, speed, and altitude with a maximum payload percentage of 1.7. From this the maximum payload, if gross weight is limited to 19,200 lbs., is found to be (.017) (19,200) = 327 lbs. for the turbo-jet vehicle as compared with the 2,400 lbs. payload of the ram-jet vehicle. If the payload desired is 2,400 lbs., then the gross weight of the turbo-jet missile will be $\frac{2400}{.017}$ = 141,000 lbs. as compared with 19,200 lbs. for the ram-jet missile. With these or similar figures in mind, it is possible to approach the problem with a knowledge of the effect upon the vehicle performance if an engine other than that dictated by gross weight evaluation is to be considered.

Graphs 36-50 are merely cross plots of graphs 1-35; the former are a representation of upper bounds of attainment for aircraft with various engines and various payload percentages, while the latter is a series representing the percentage of the gross weight of the missile required exclusive of the payload, plotted versus range, at a specified altitude and Mach number. In this latter group 100% minus the necessary percentage of gross weight as determined by range, velocity, and altitude, will be the maximum allowable payload percentage. Therefore, when operating conditions to be met are identical with those of charts 1-35, it is advisable to refer to these latter charts for performance evaluation.

It is to be noted in this series of graphs that whenever a particular design type has not been able to exceed the 100 mile range under the given conditions, it has been omitted; for example, the rocket is not shown in chart No. 4 and the turbo-jet is not shown in chart No. 32.

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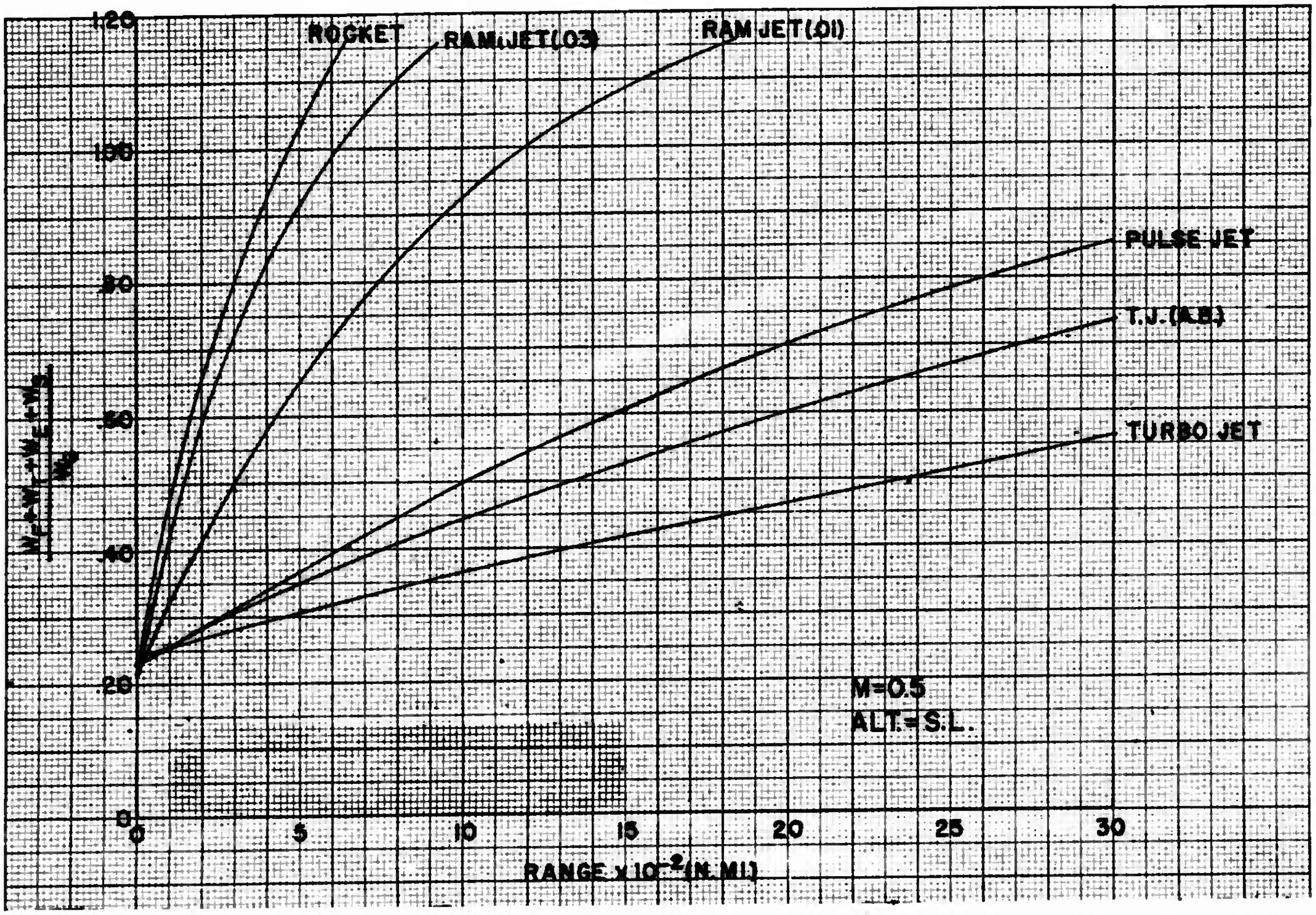

FIGURE 1 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

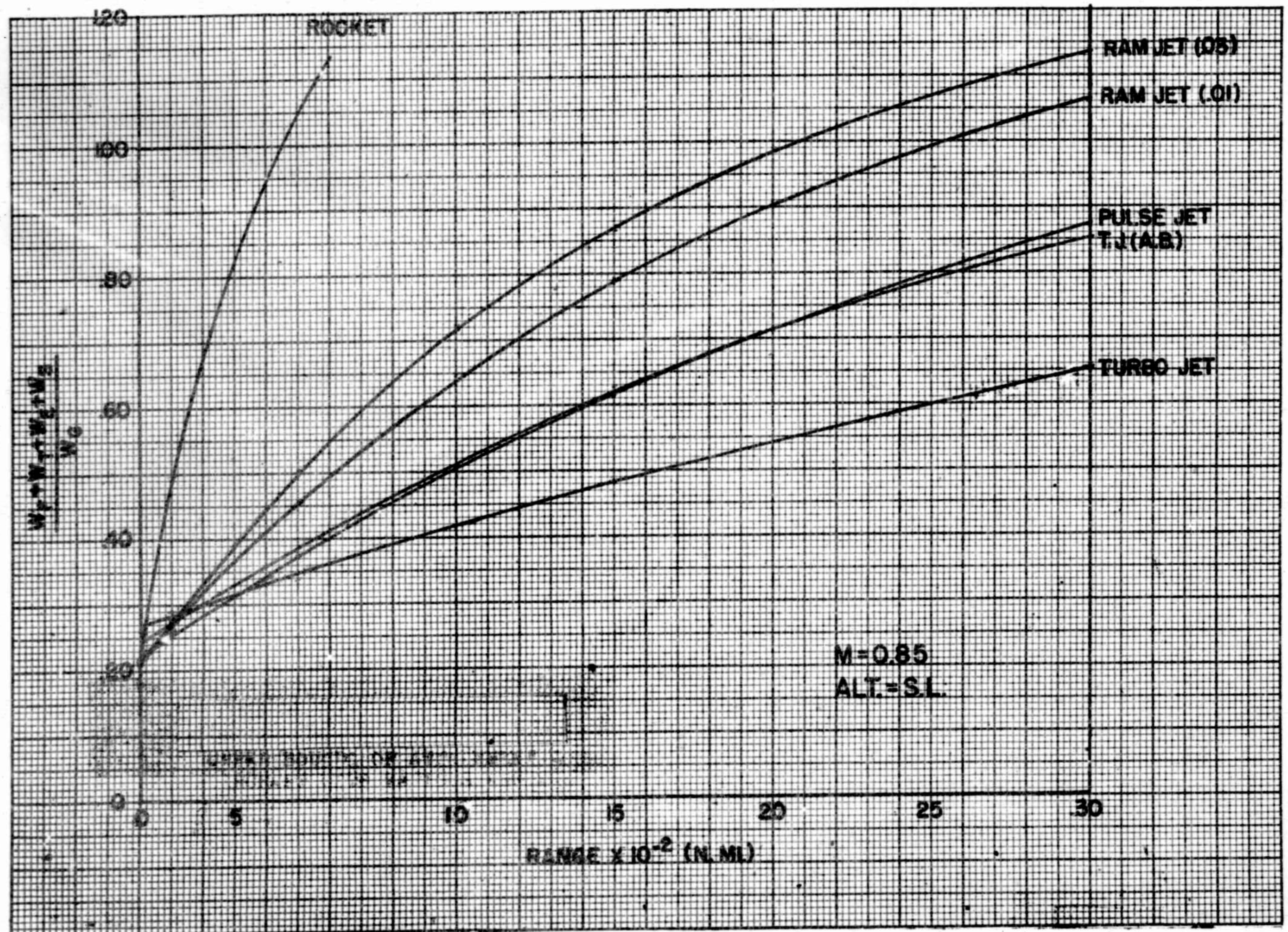

FIGURE 2 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

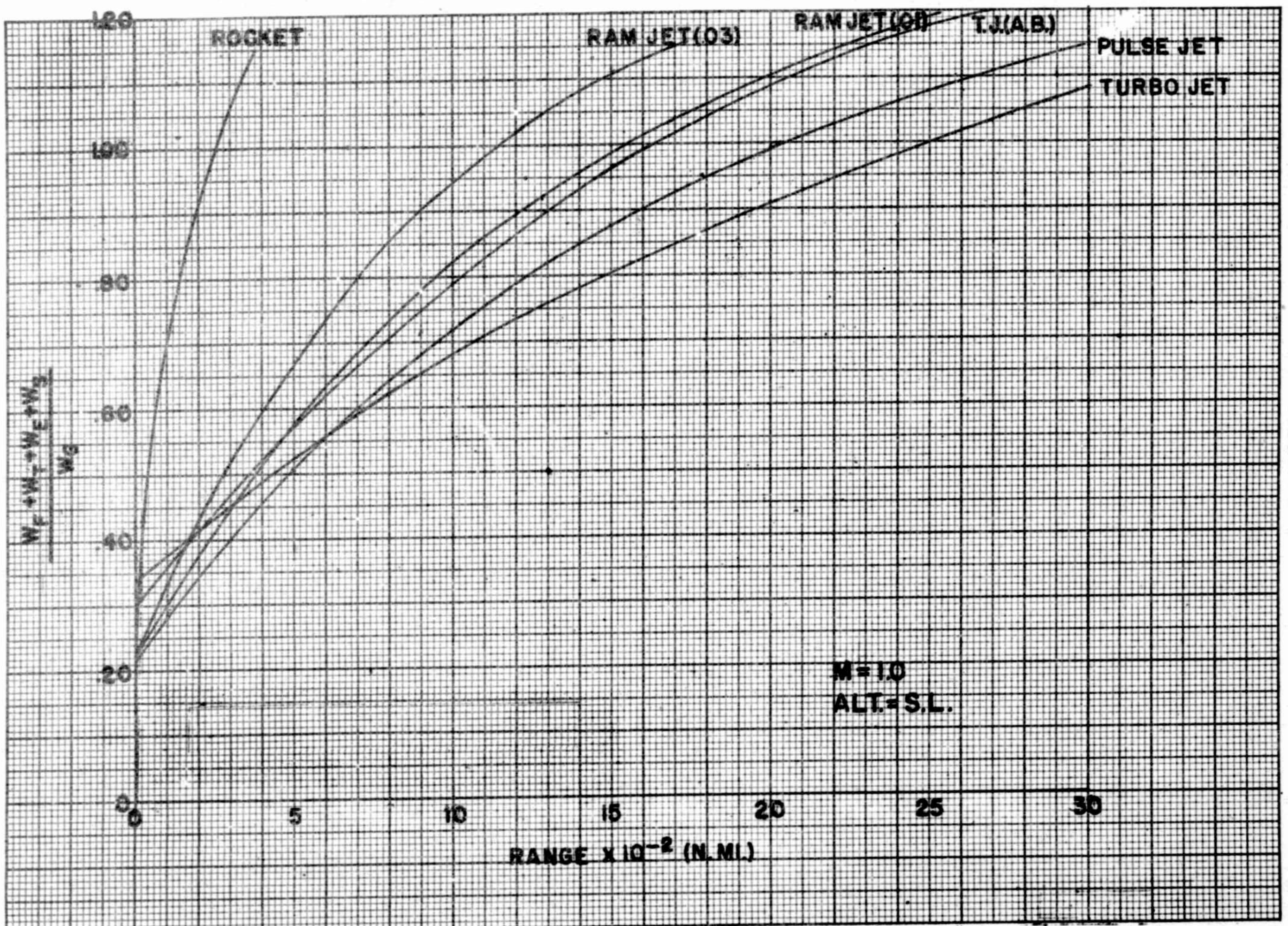

FIGURE 3 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

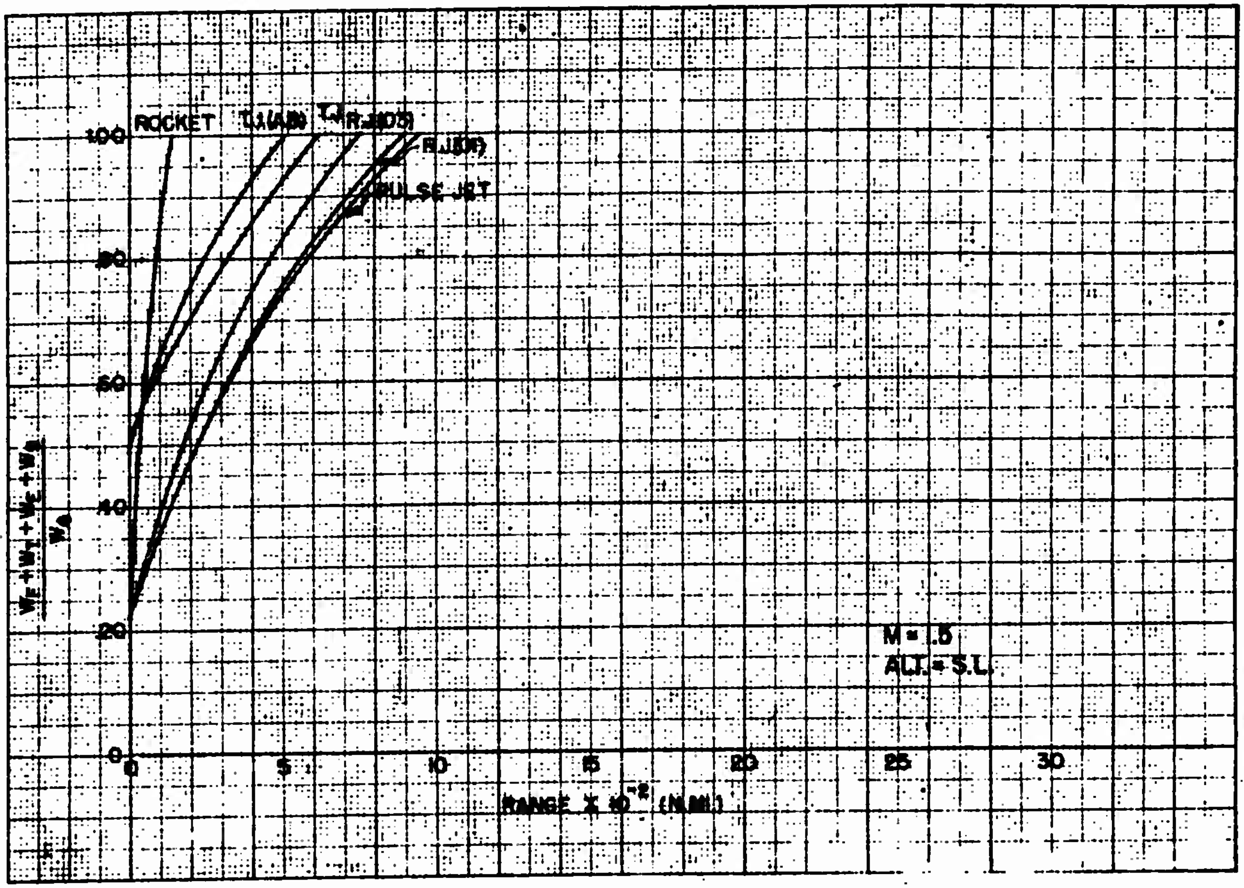

FIGURE 4 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

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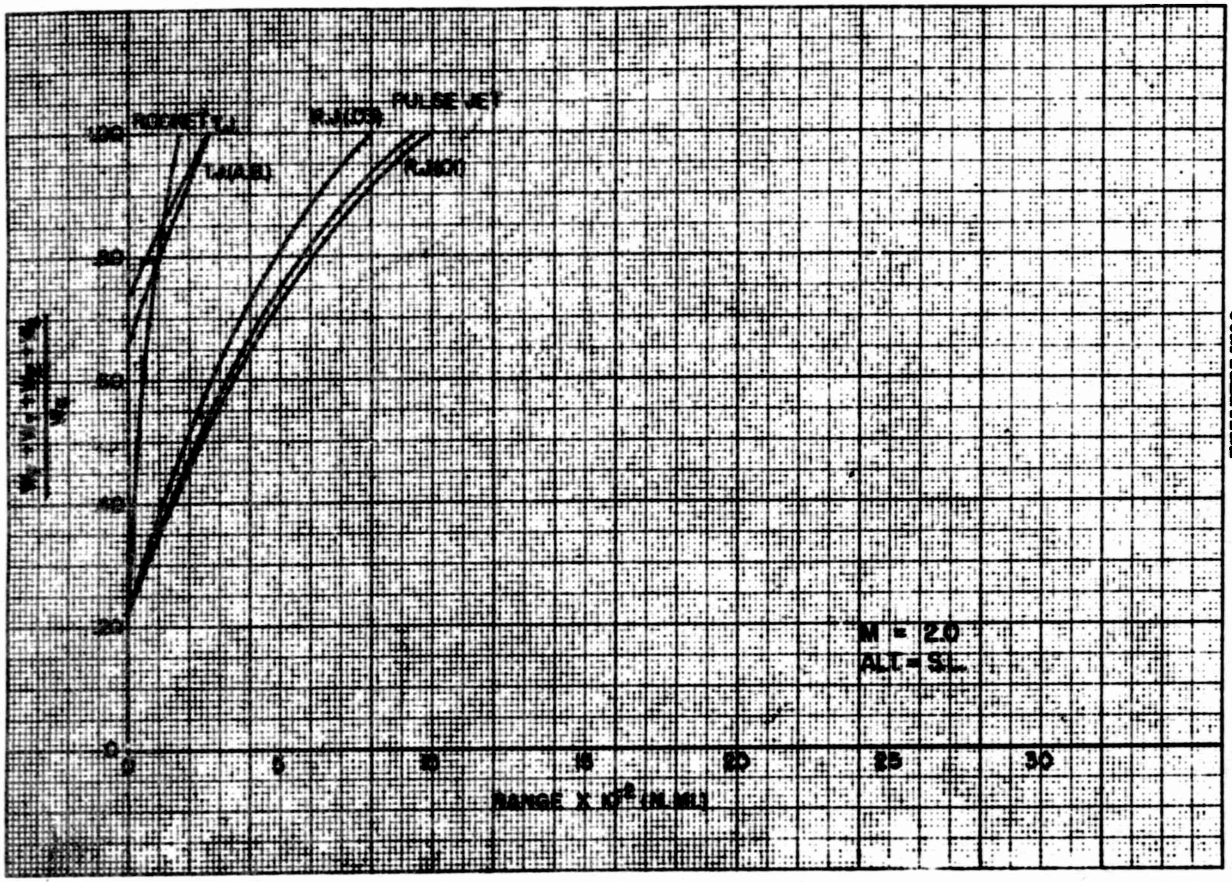

FIGURE 5 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

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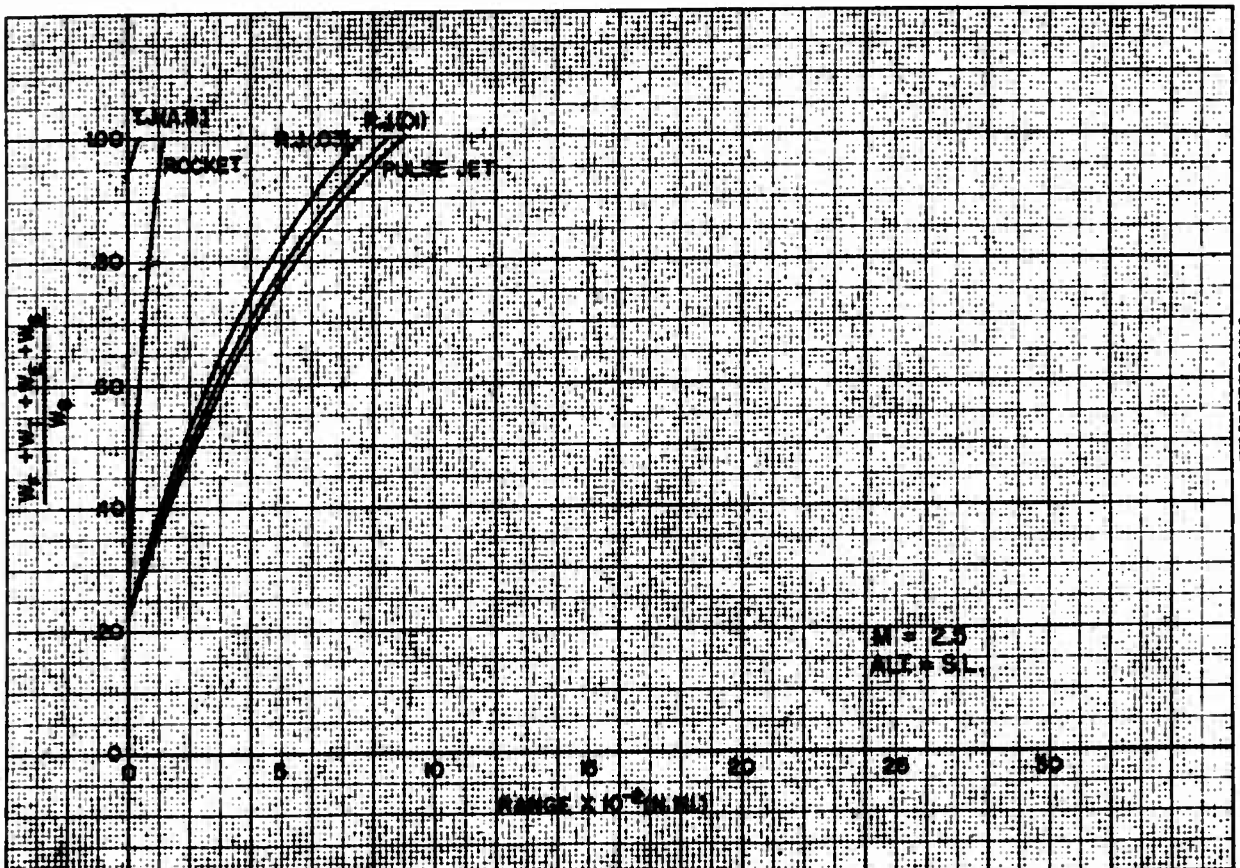

FIGURE 6 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

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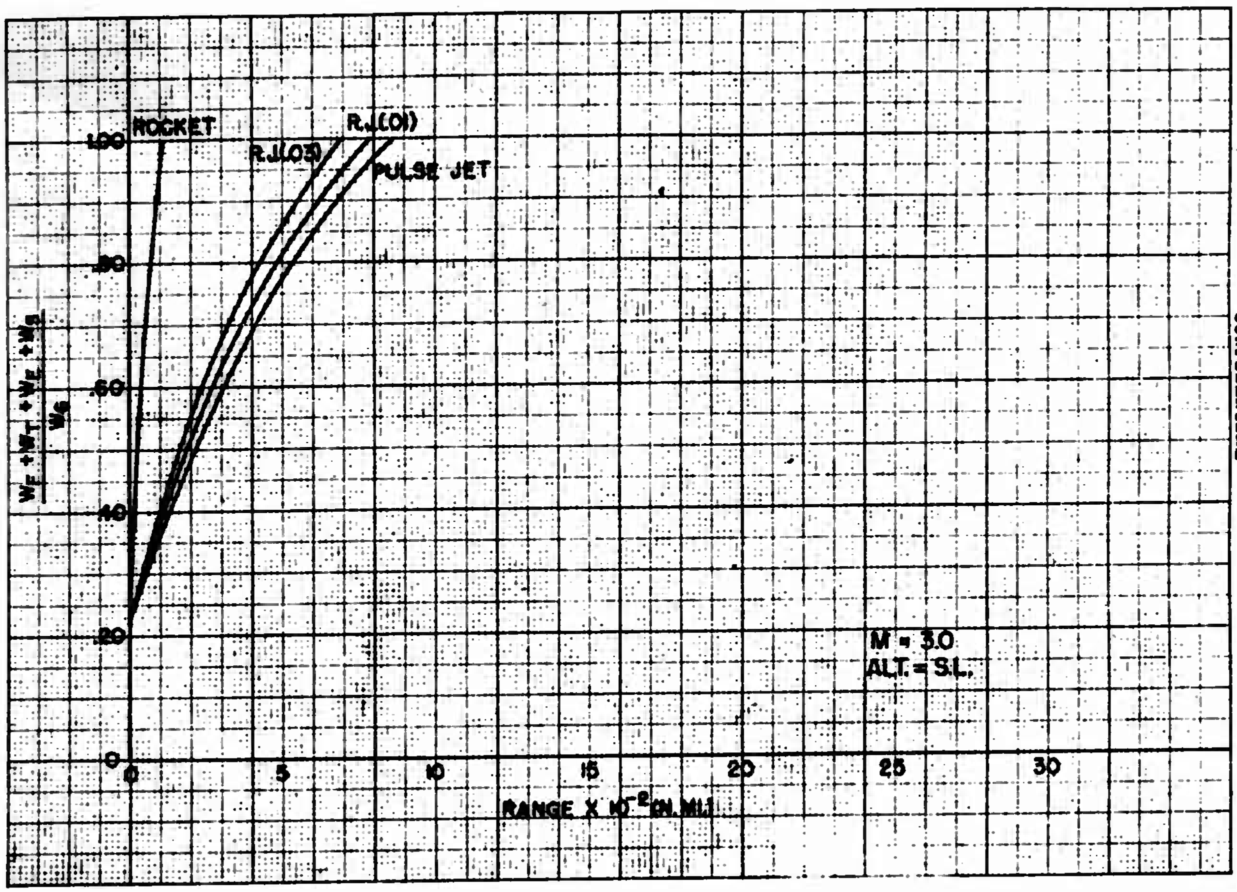

FIGURE 7 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

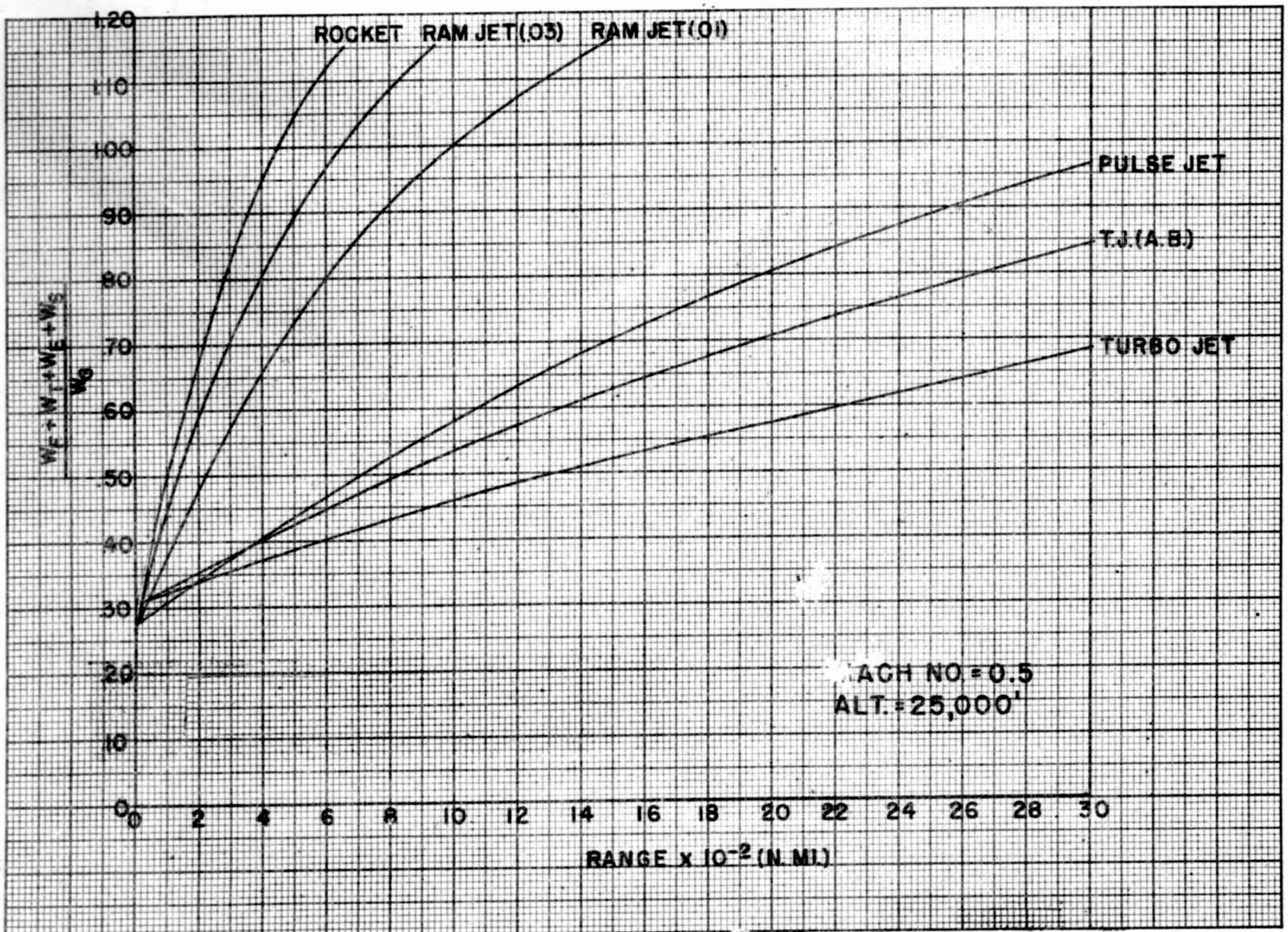

FIGURE 8 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

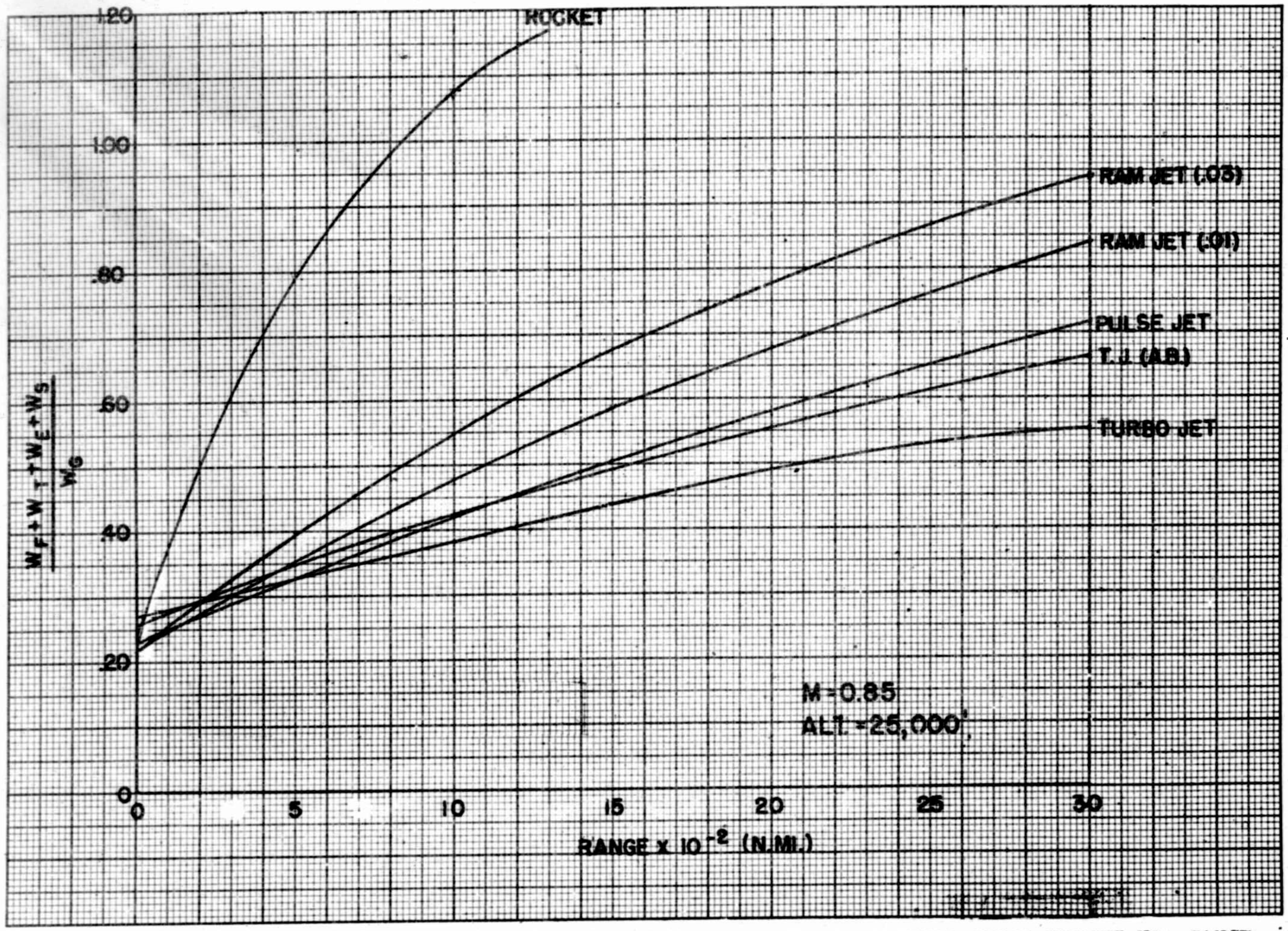

FIGURE 9 - RATIO OF WEIGHT OF ENGINE, FUEL. TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

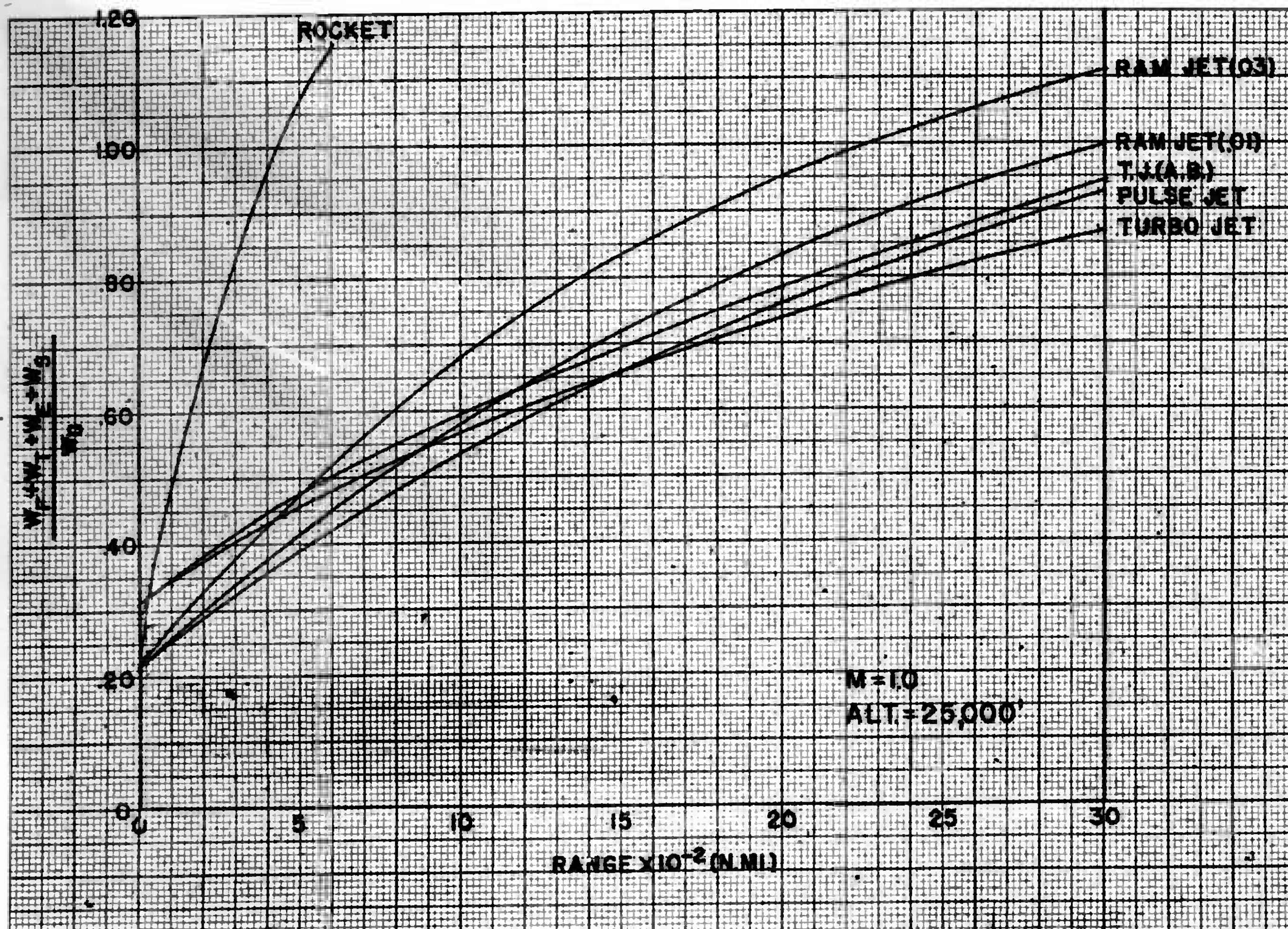

FIGURE 10 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

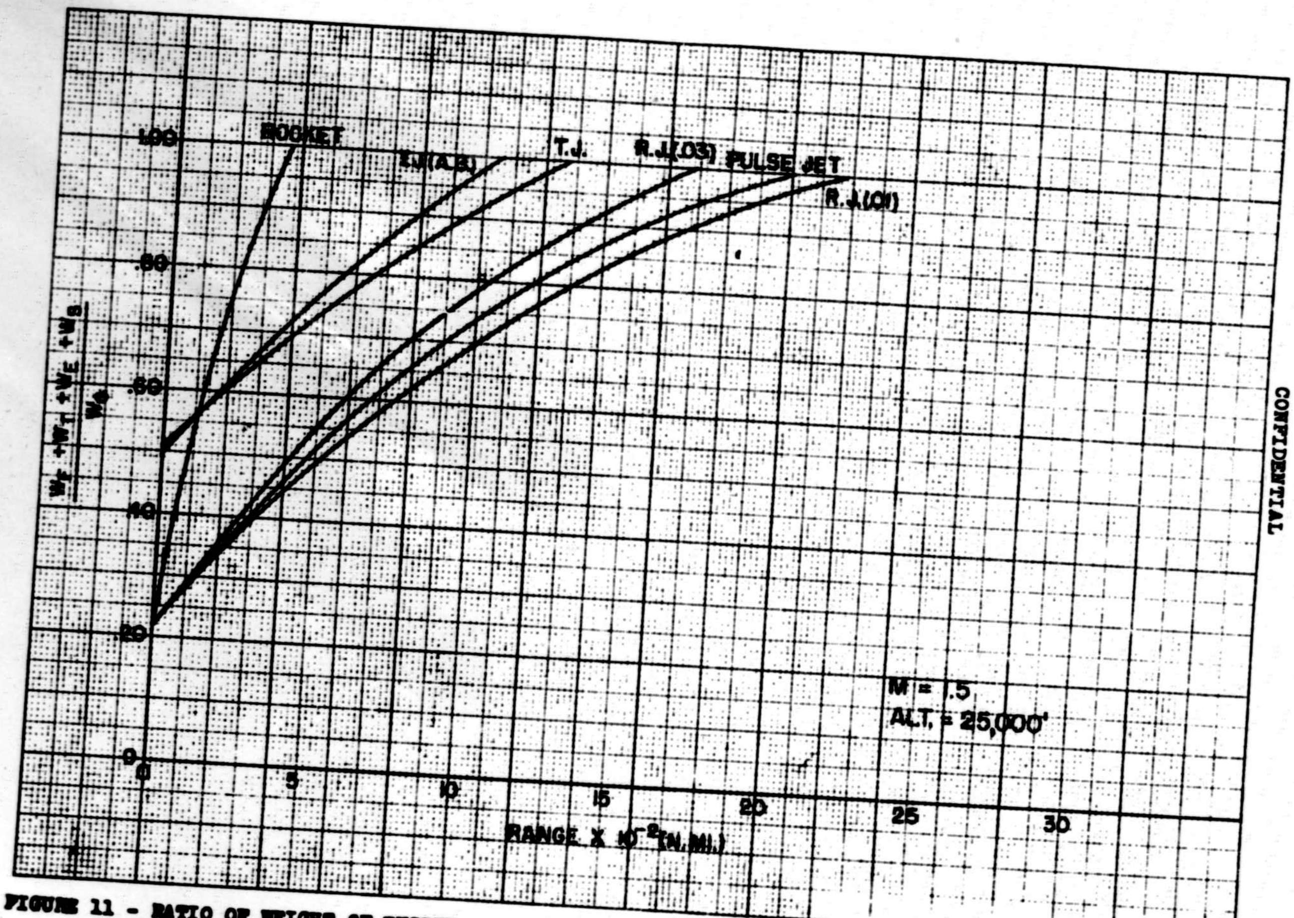

FIGURE 11 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

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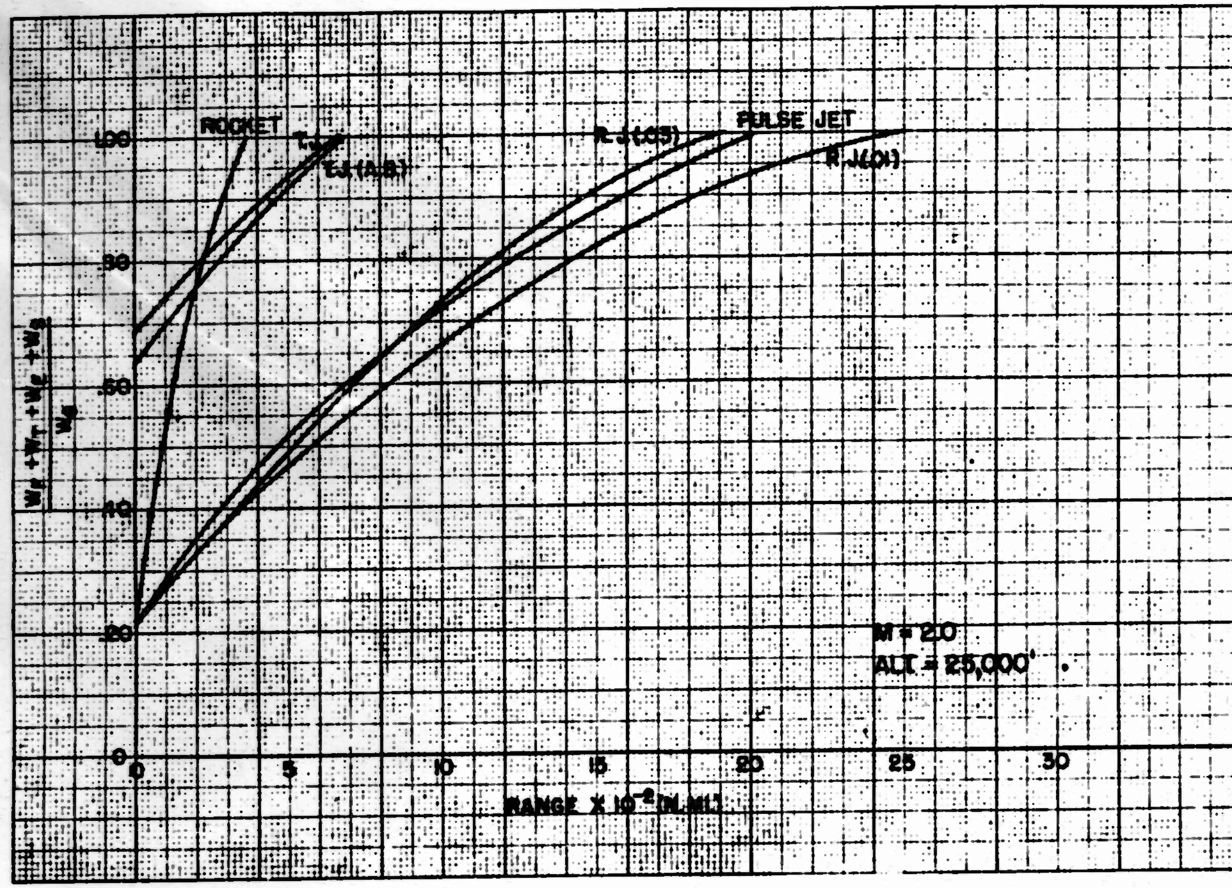

FIGURE 12 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

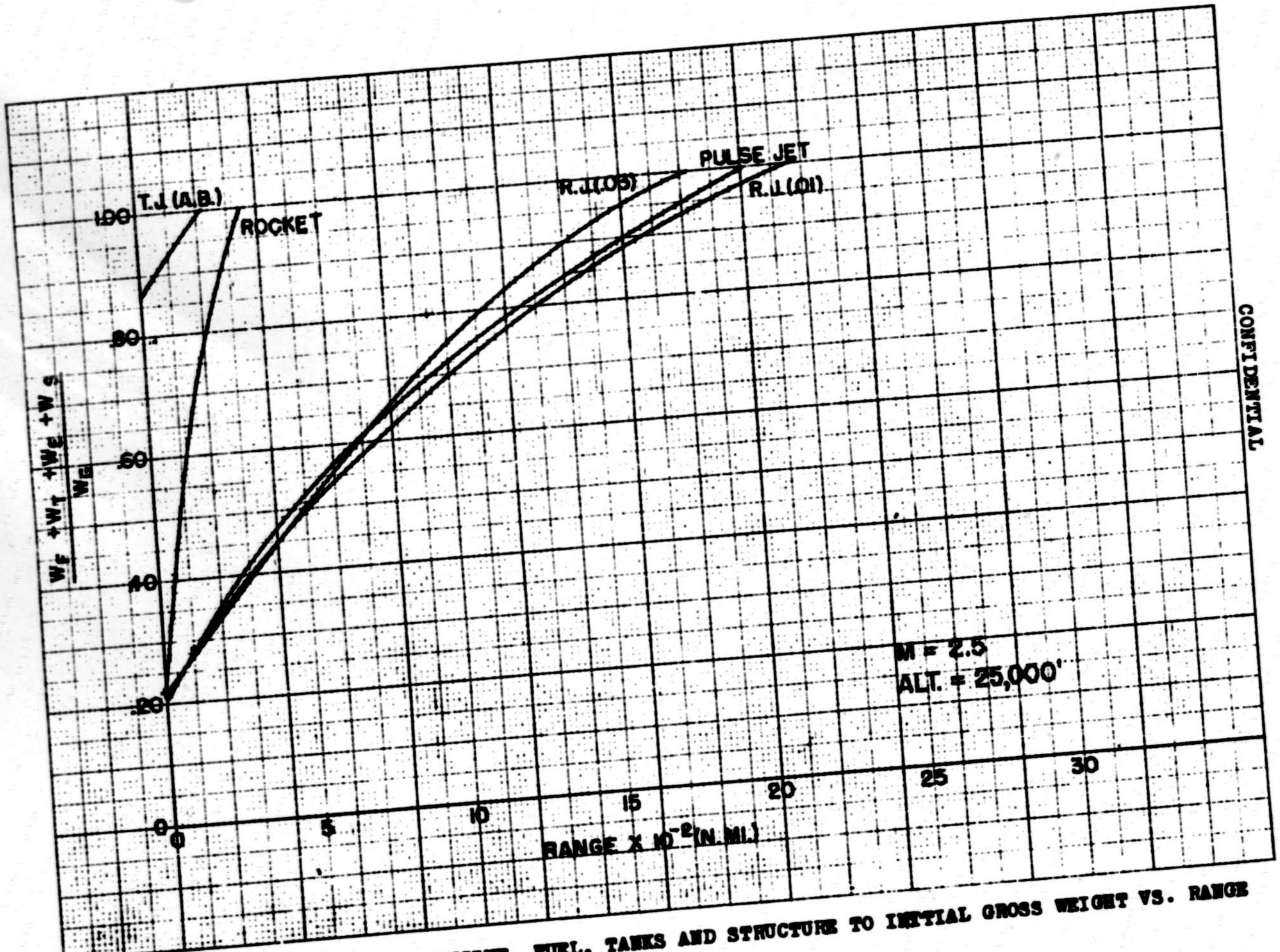

FIGURE 13 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

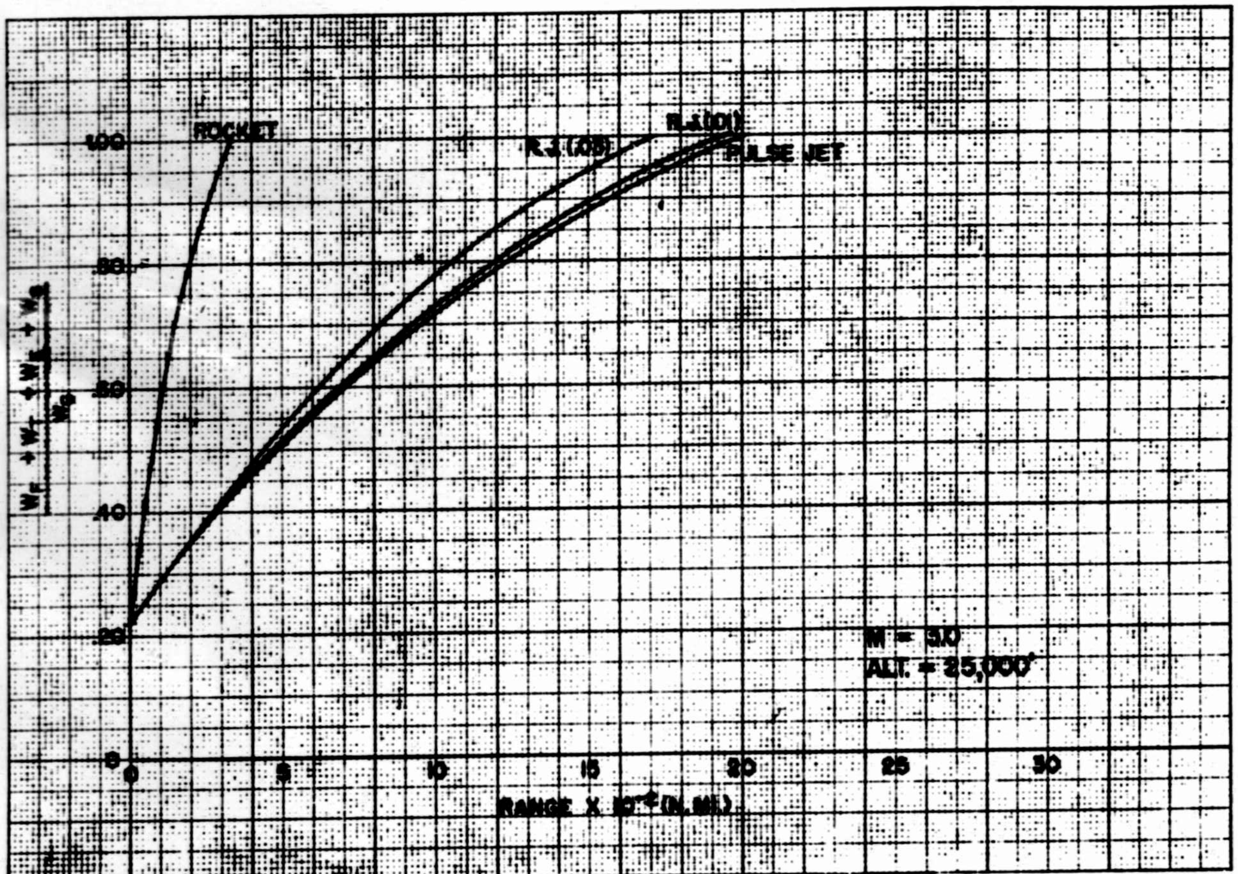

FIGURE 14 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

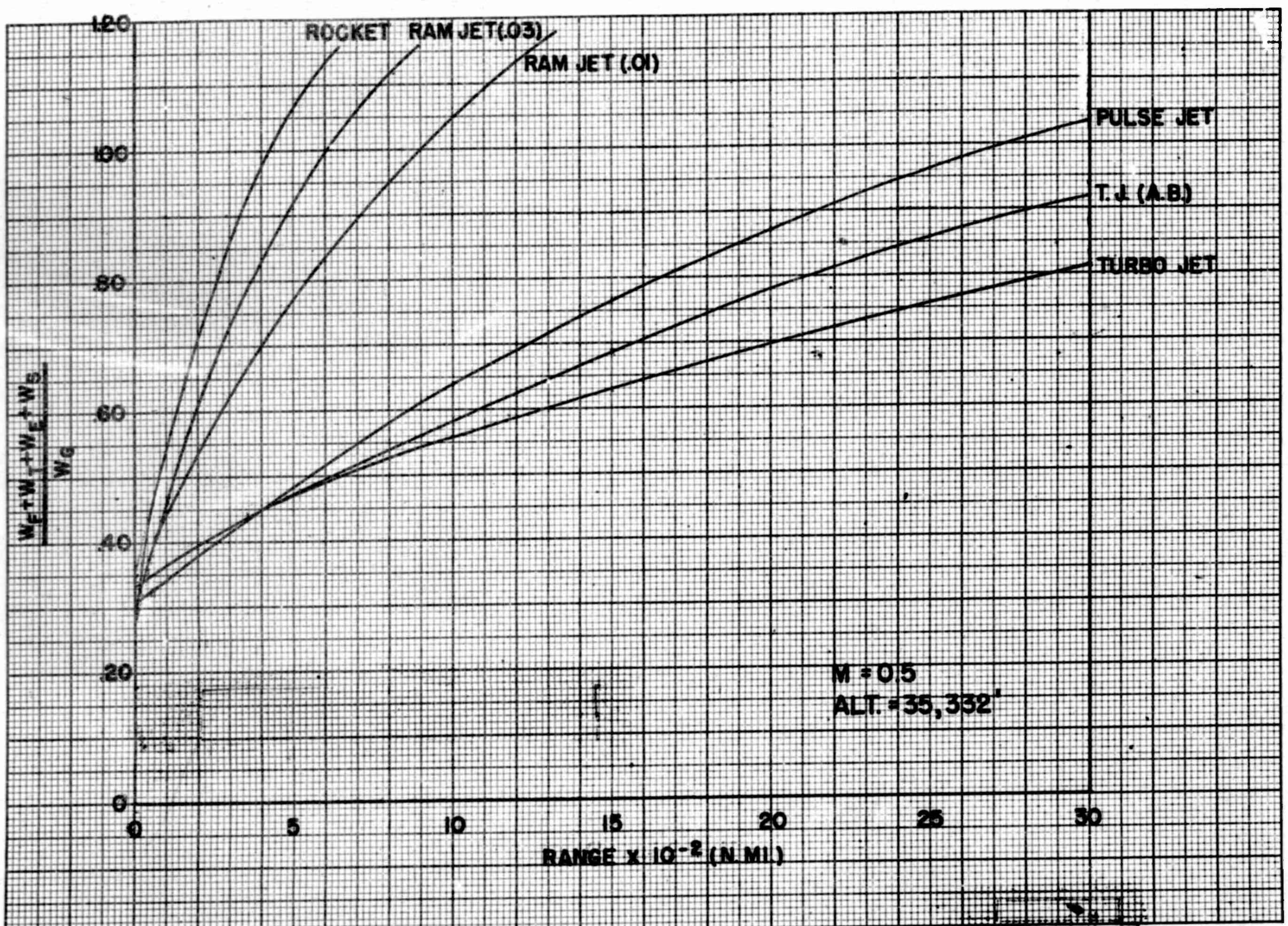

FIGURE 15 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

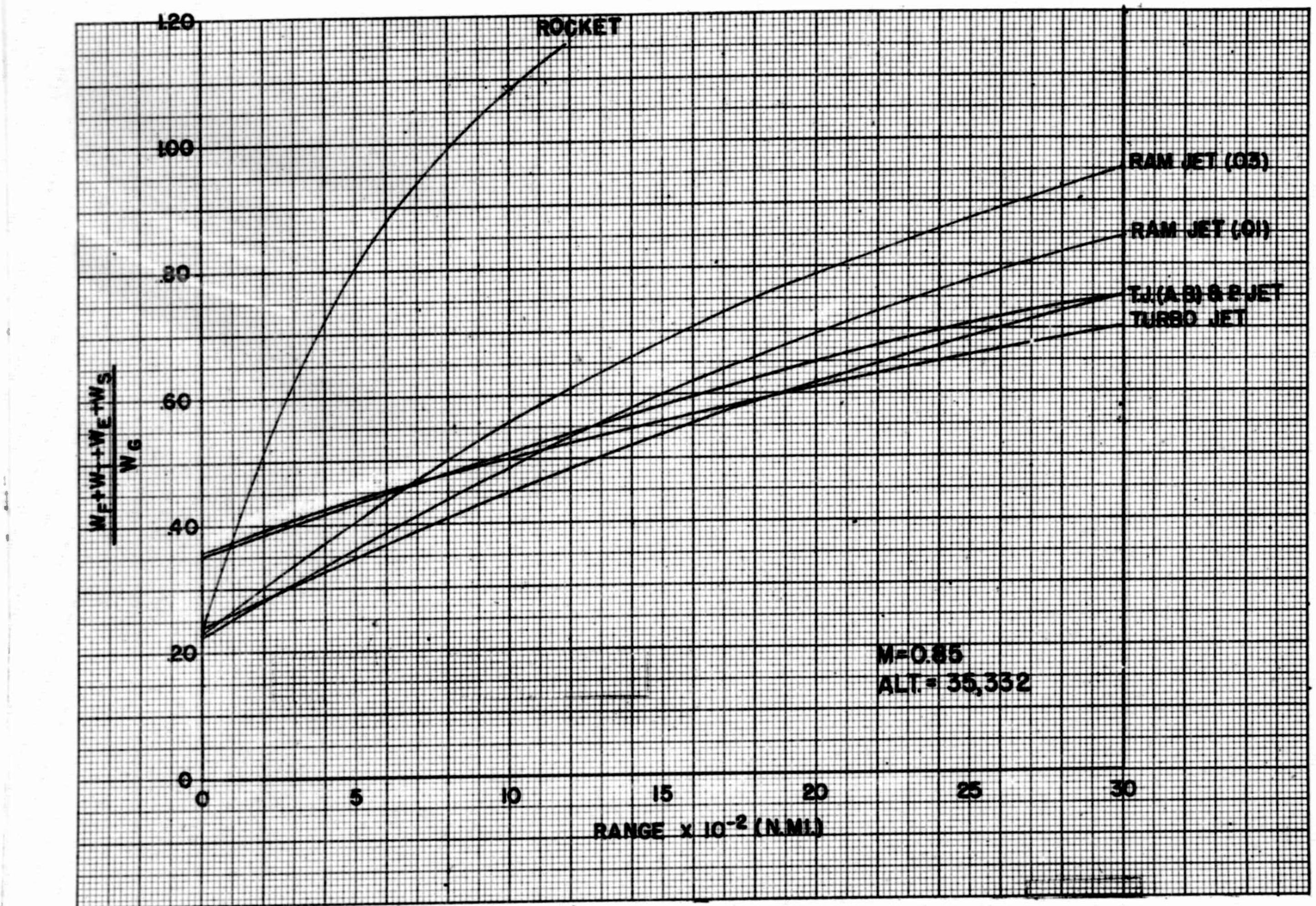

FIGURE 16 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

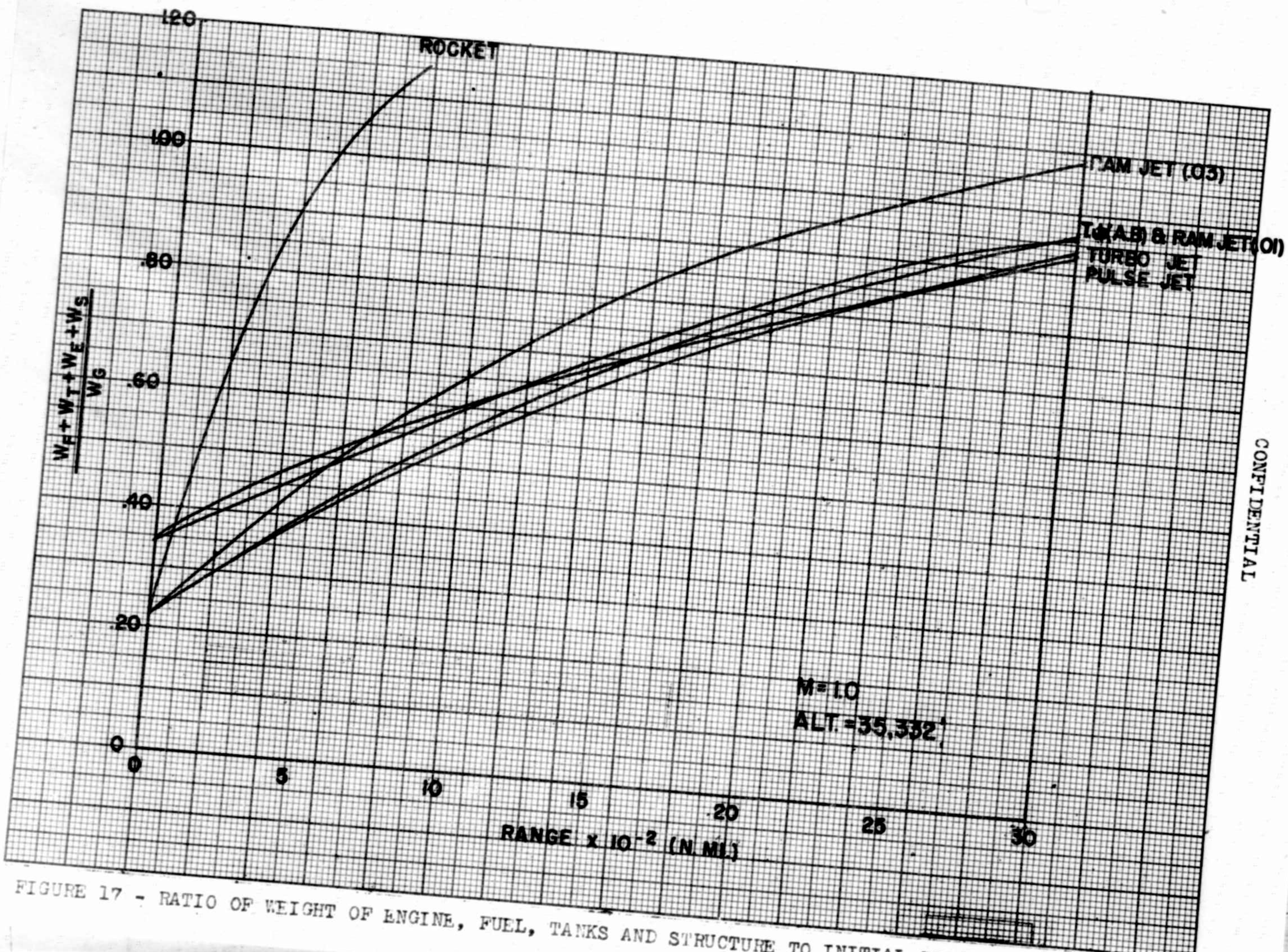

FIGURE 17 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

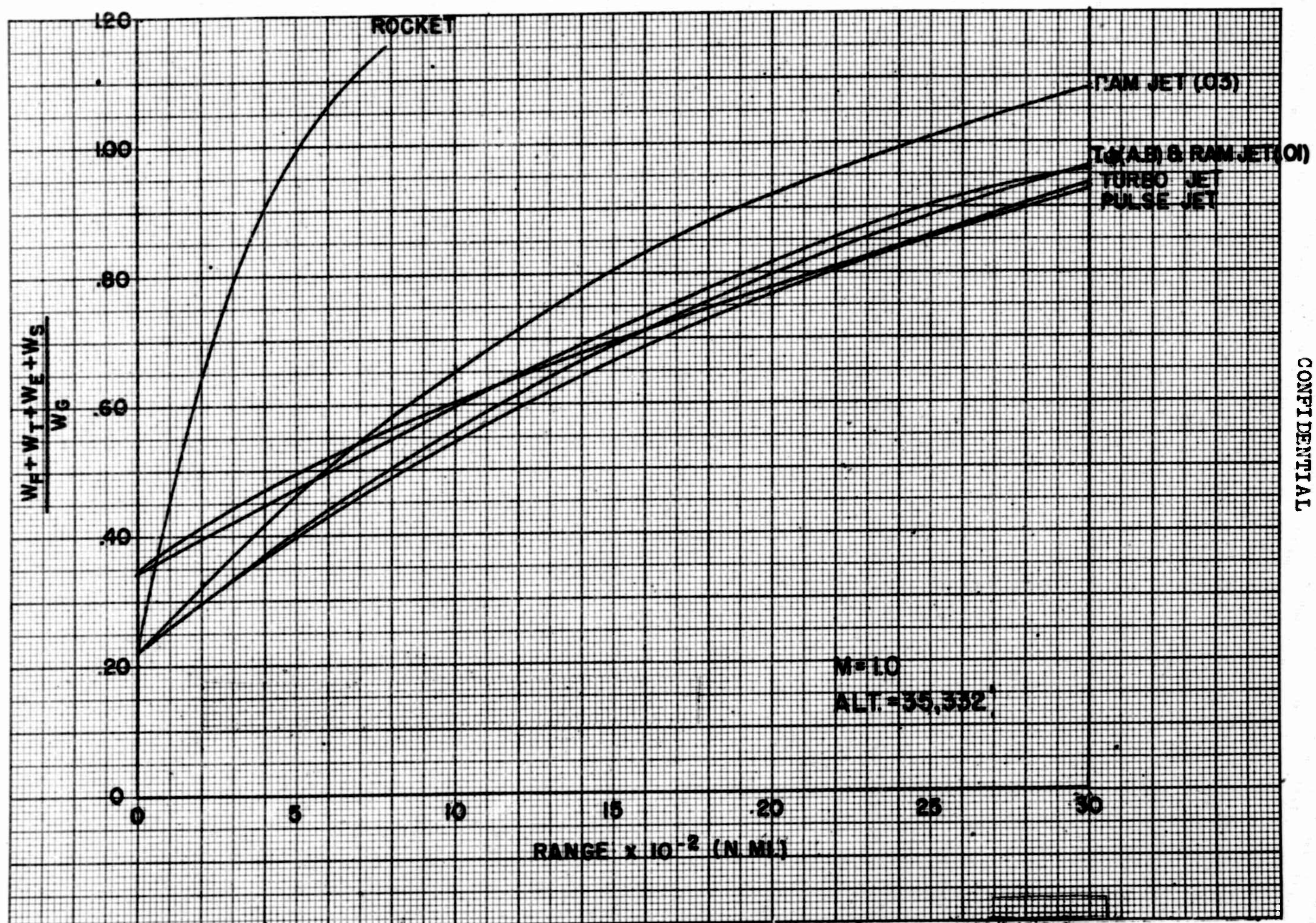

FIGURE 17 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

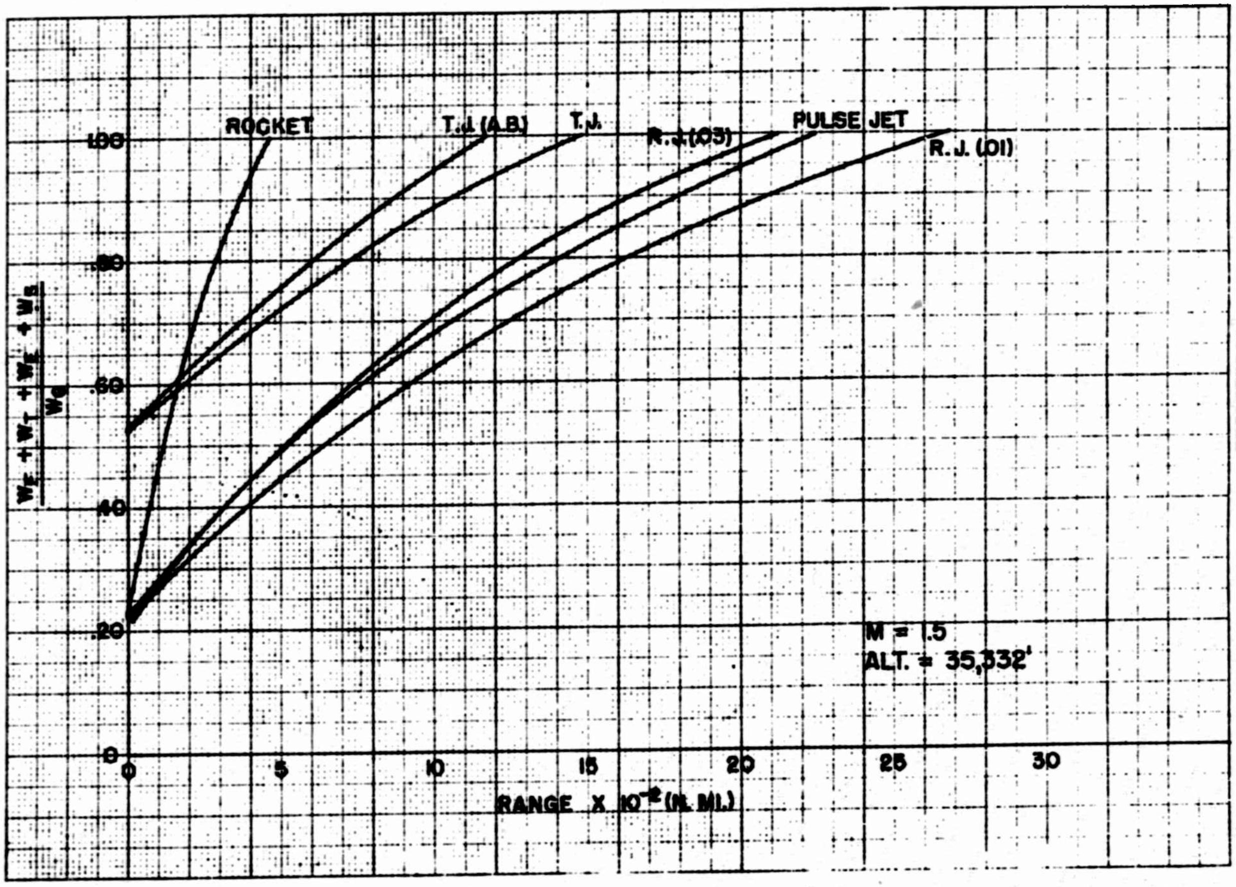

FIGURE 18 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE.

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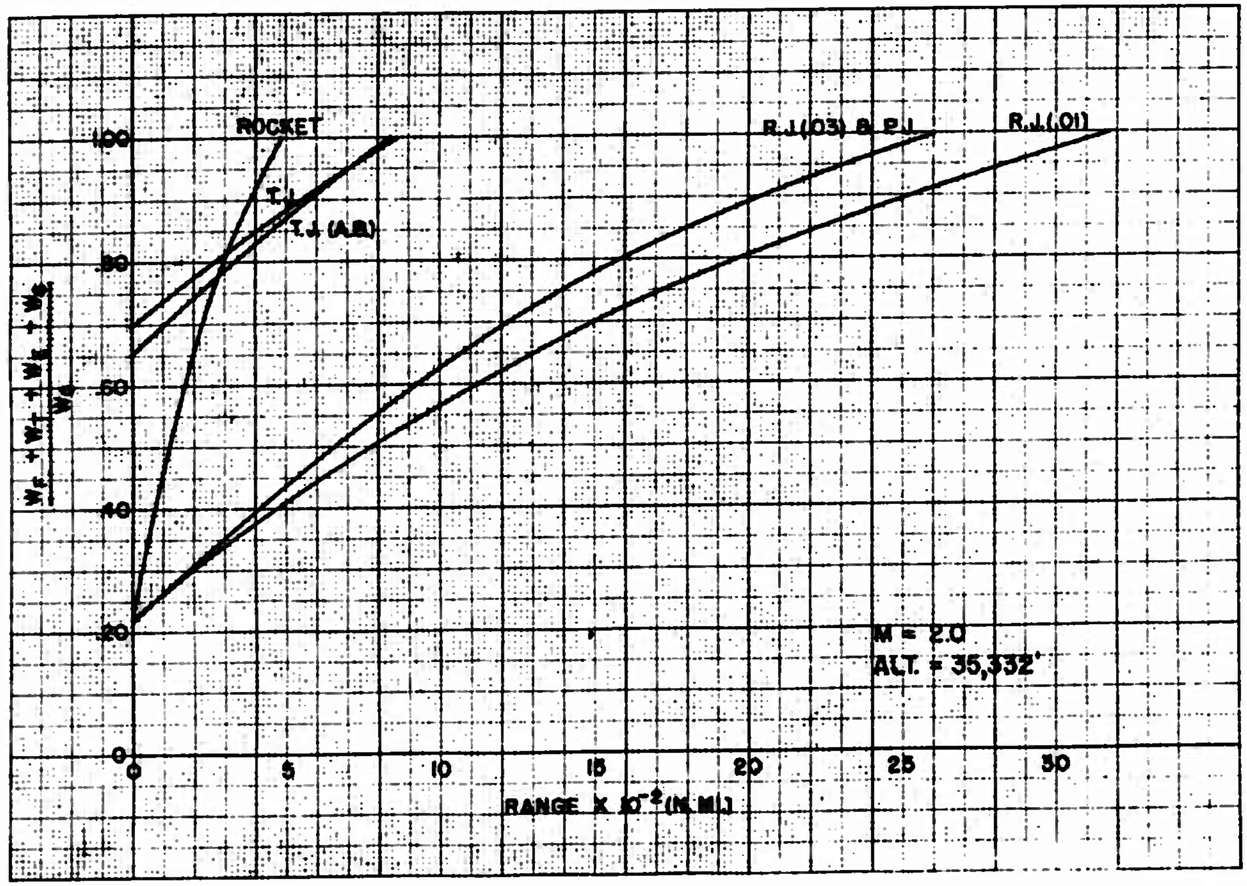

FIGURE 19 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

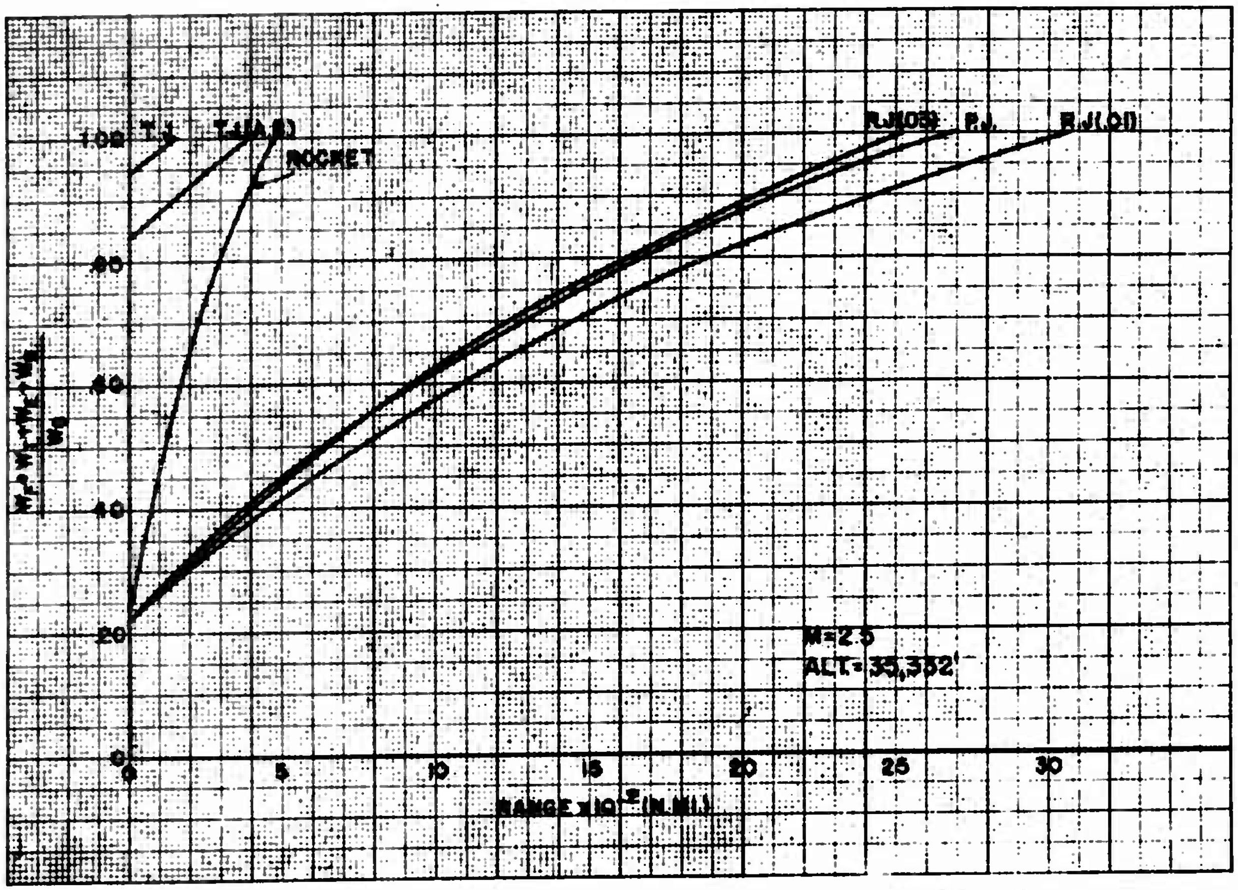

FIGURE 30 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

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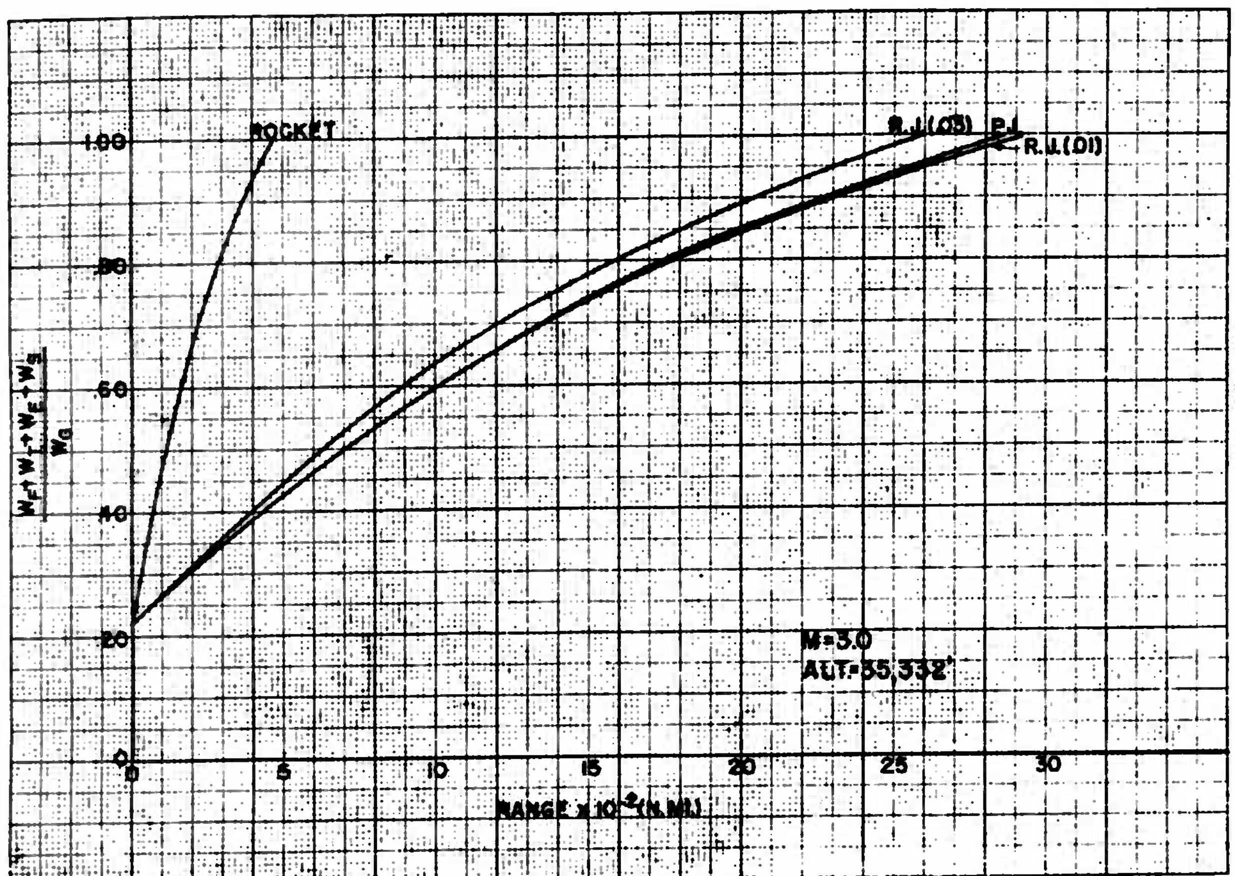

FIGURE 21 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

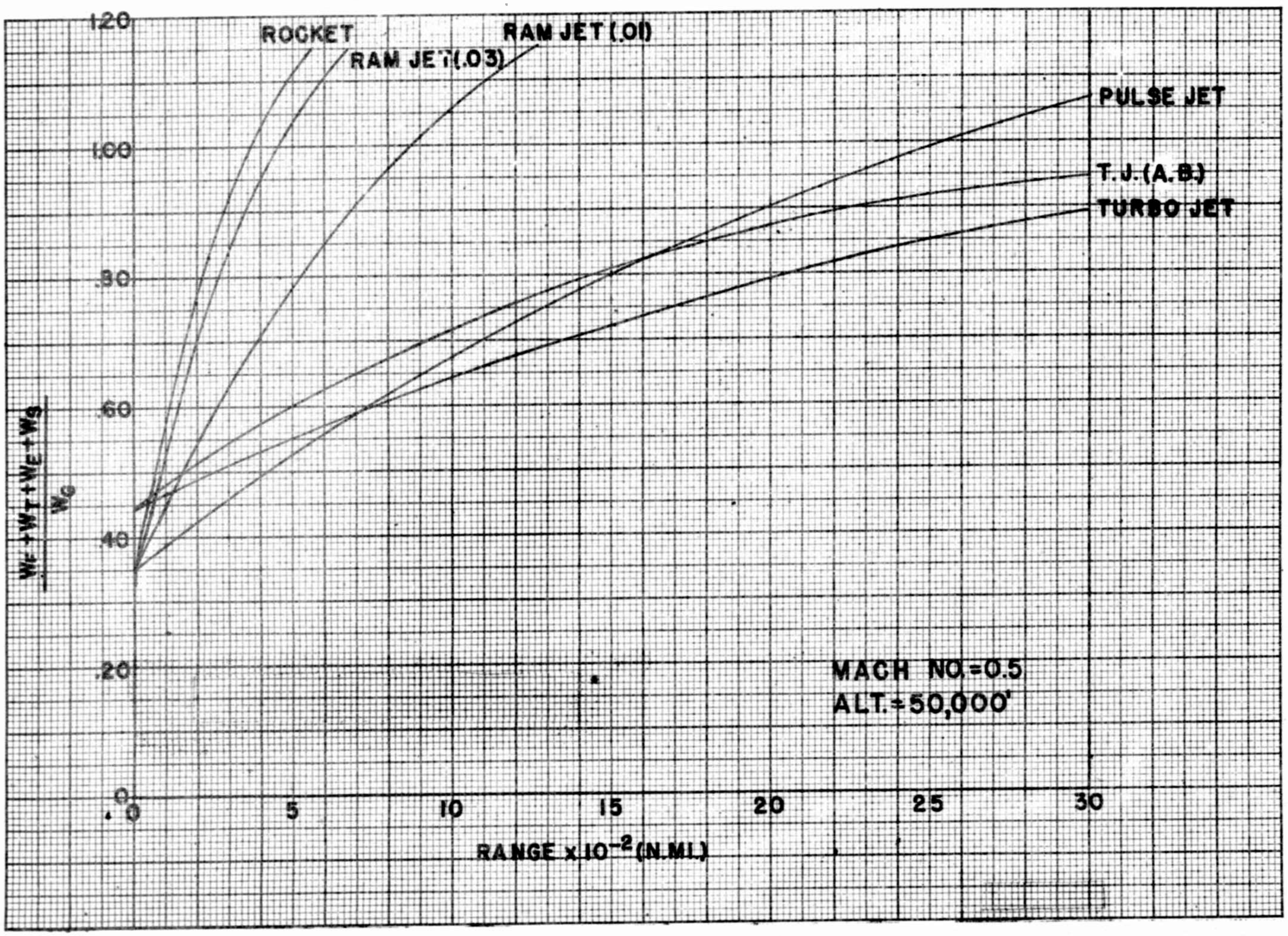

FIGURE 22 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

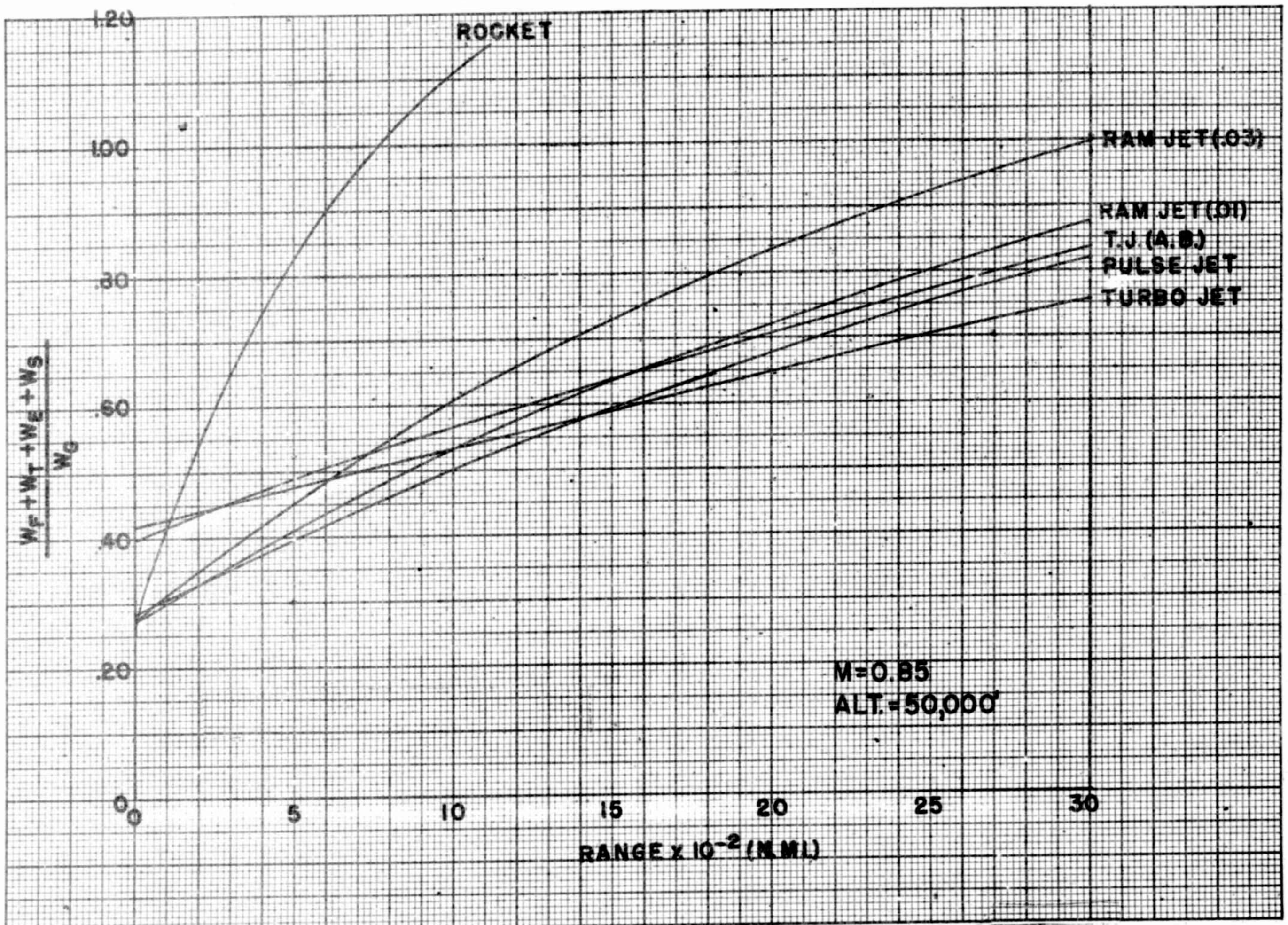

FIGURE 23 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

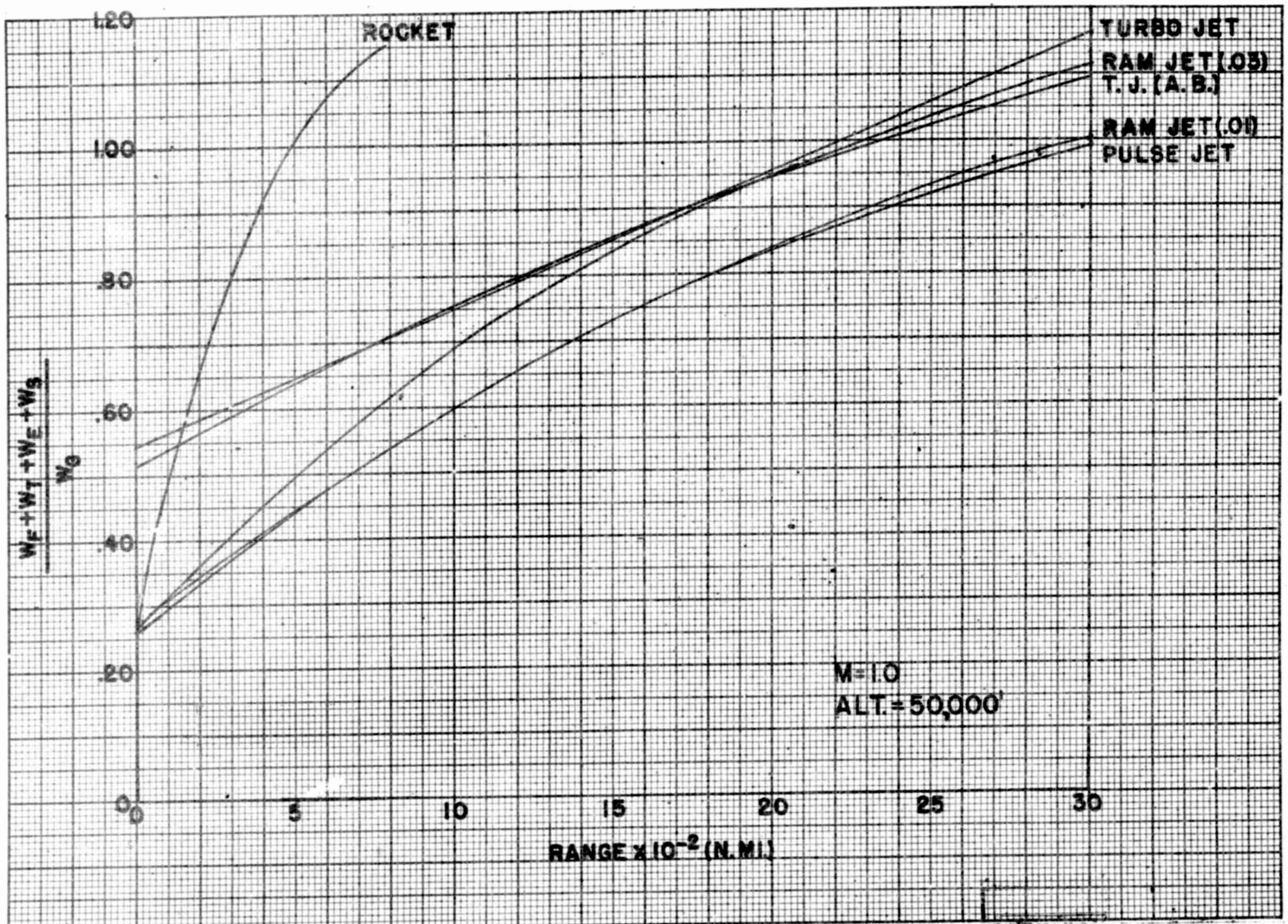

FIGURE 24 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

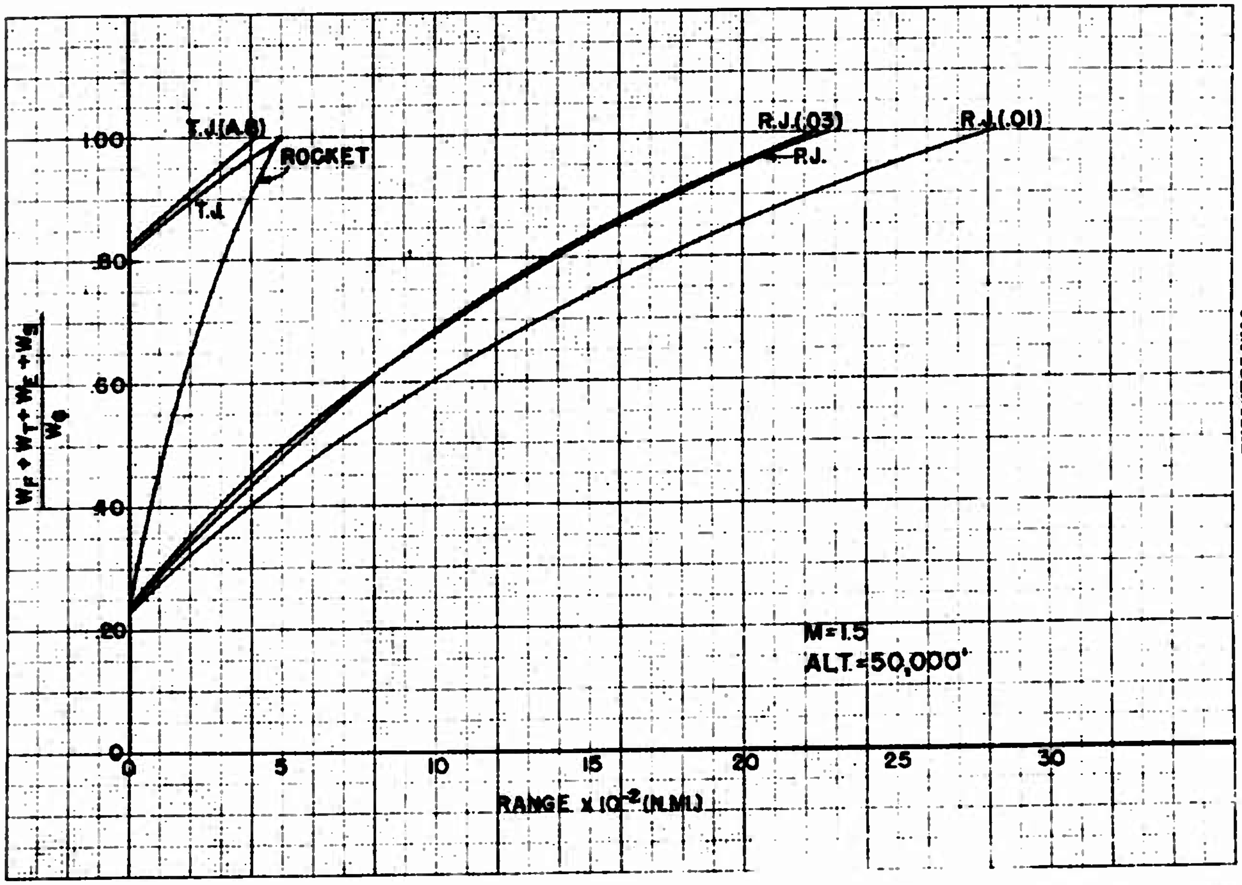

FIGURE 25 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

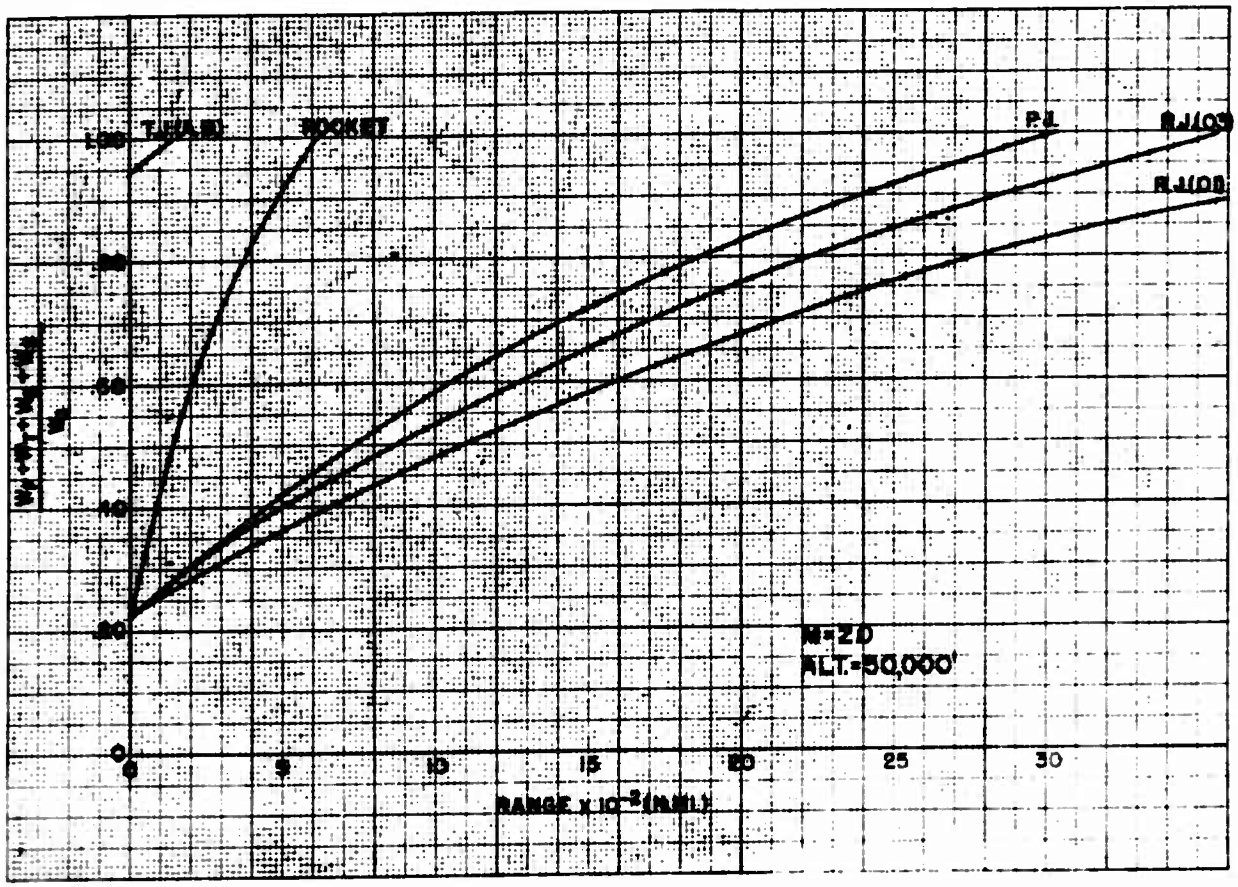

FIGURE 26 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

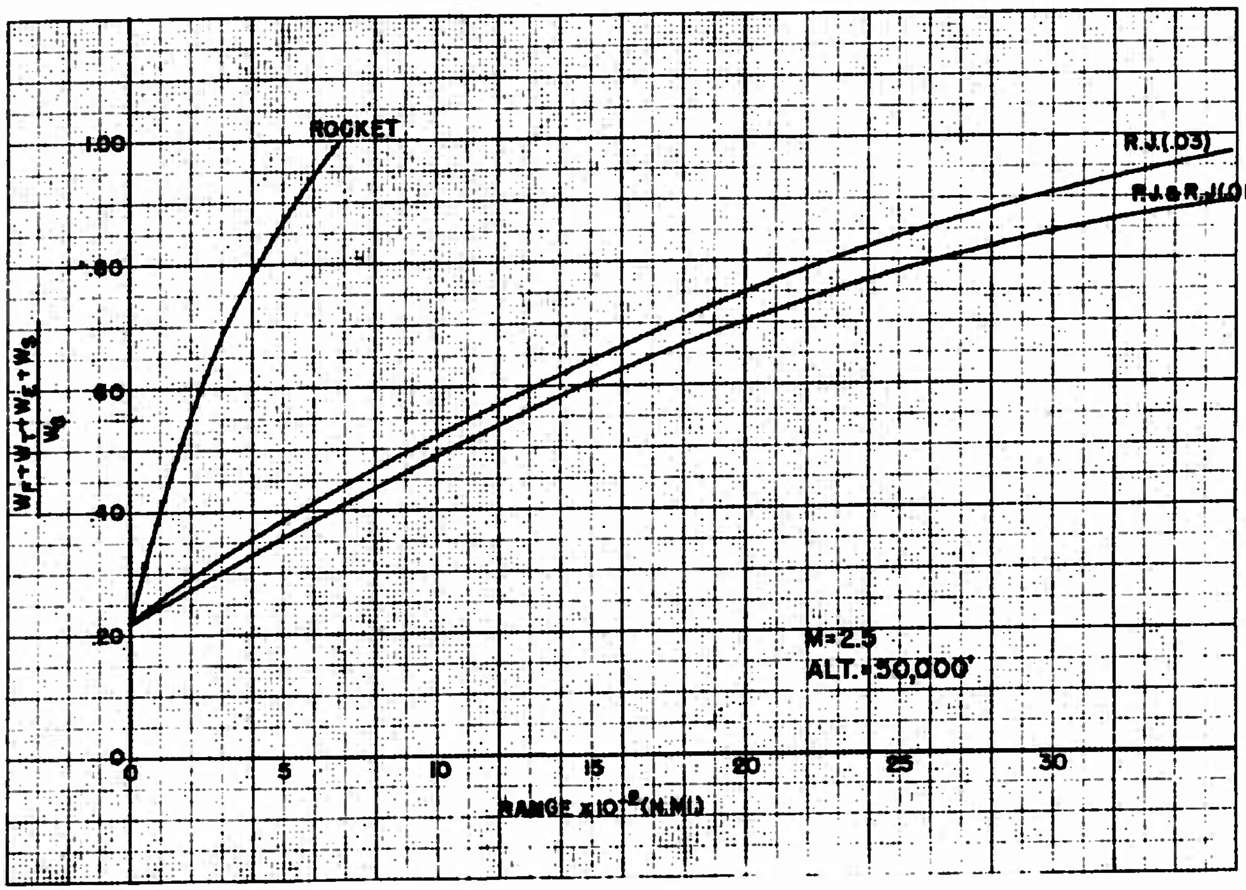

FIGURE 27 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

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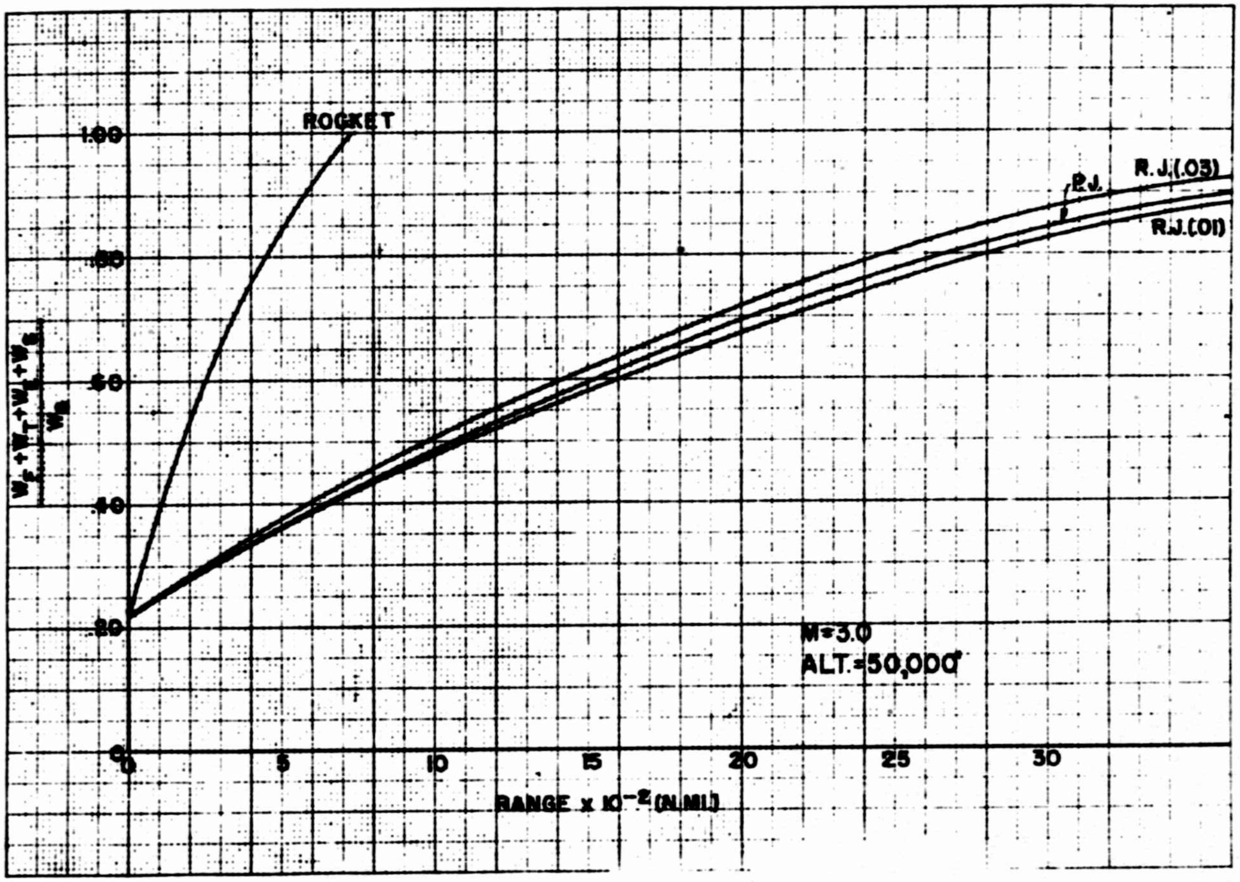

FIGURE 28 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

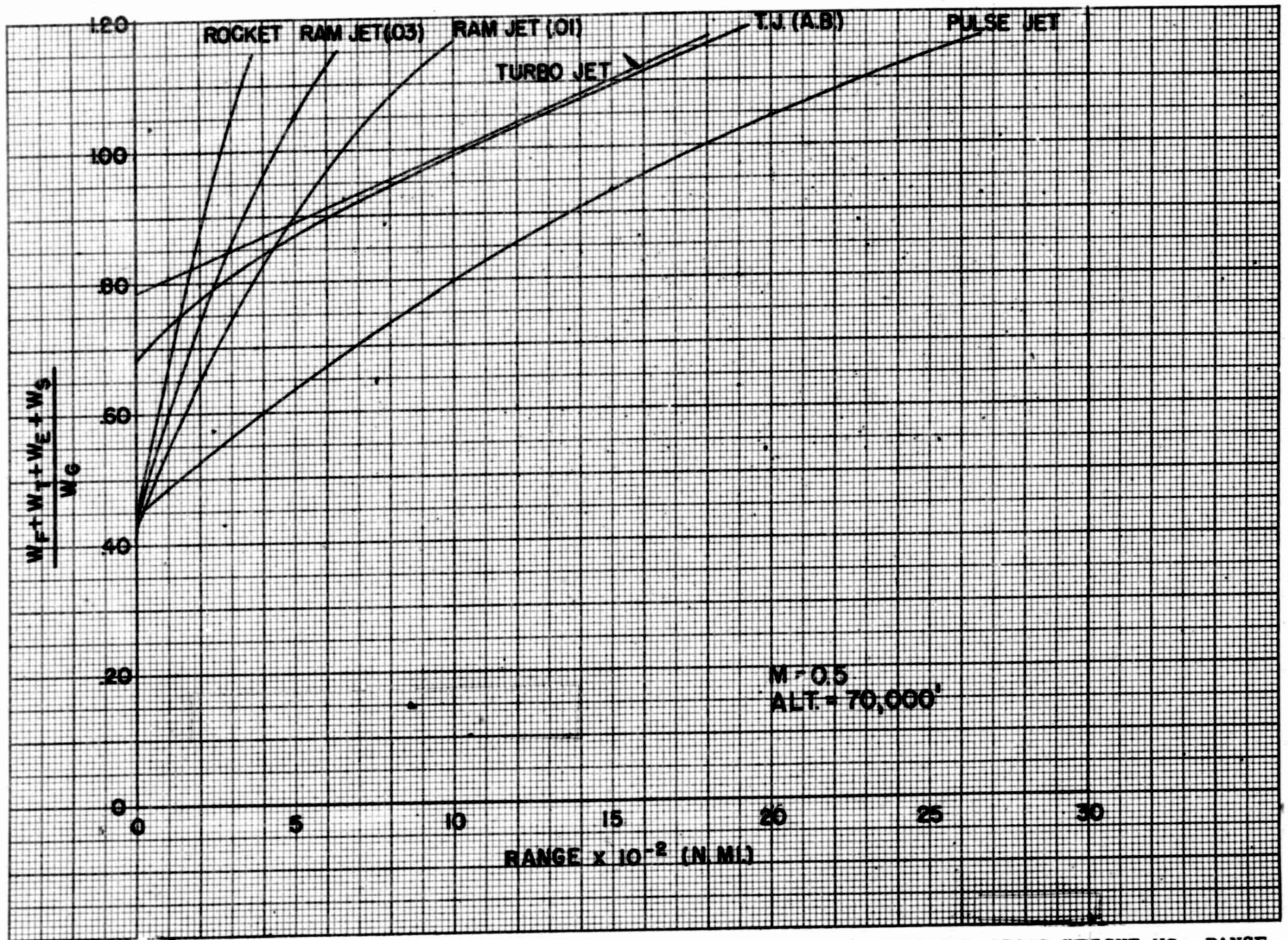

FIGURE 29 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

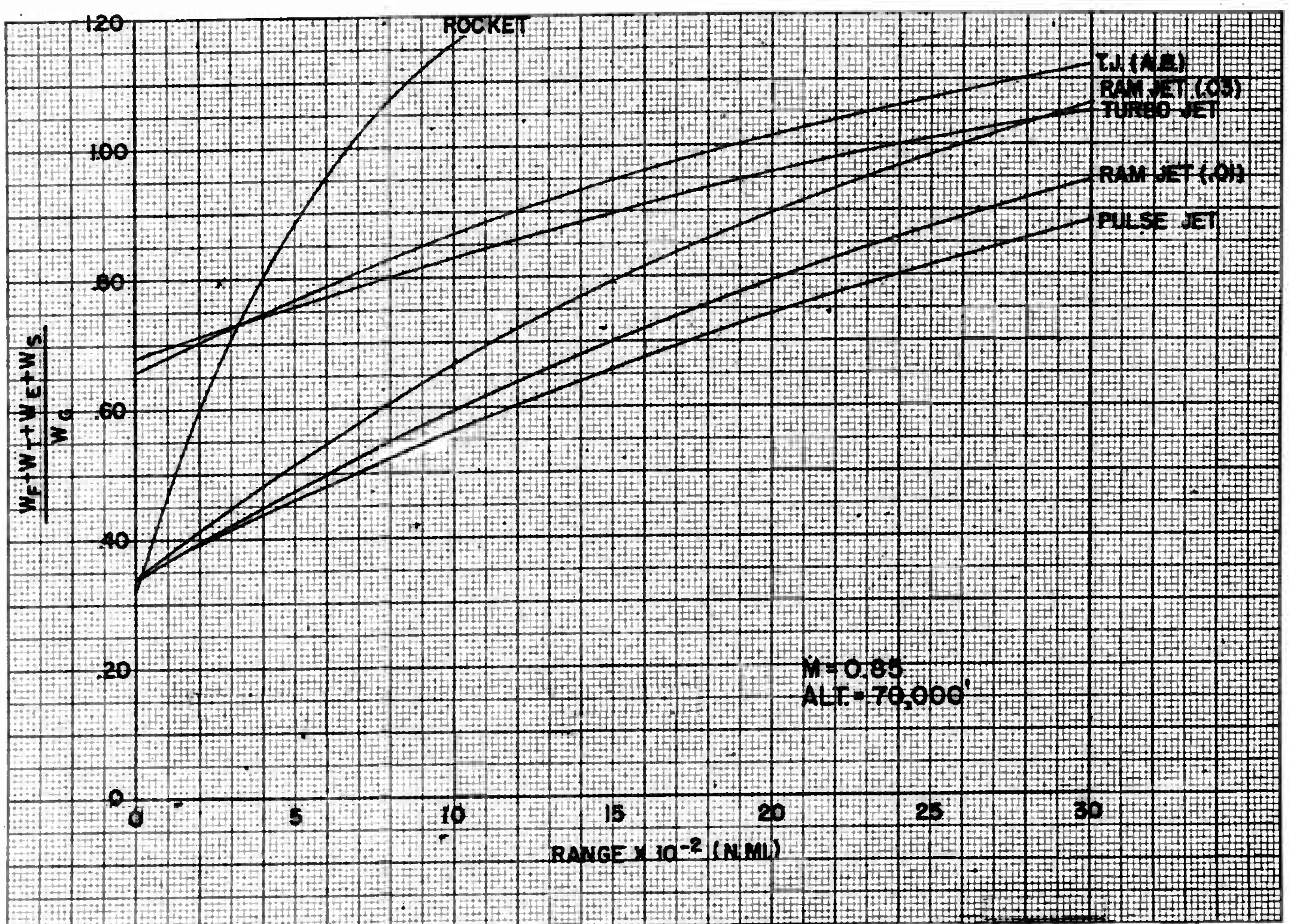

FIGURE 30 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

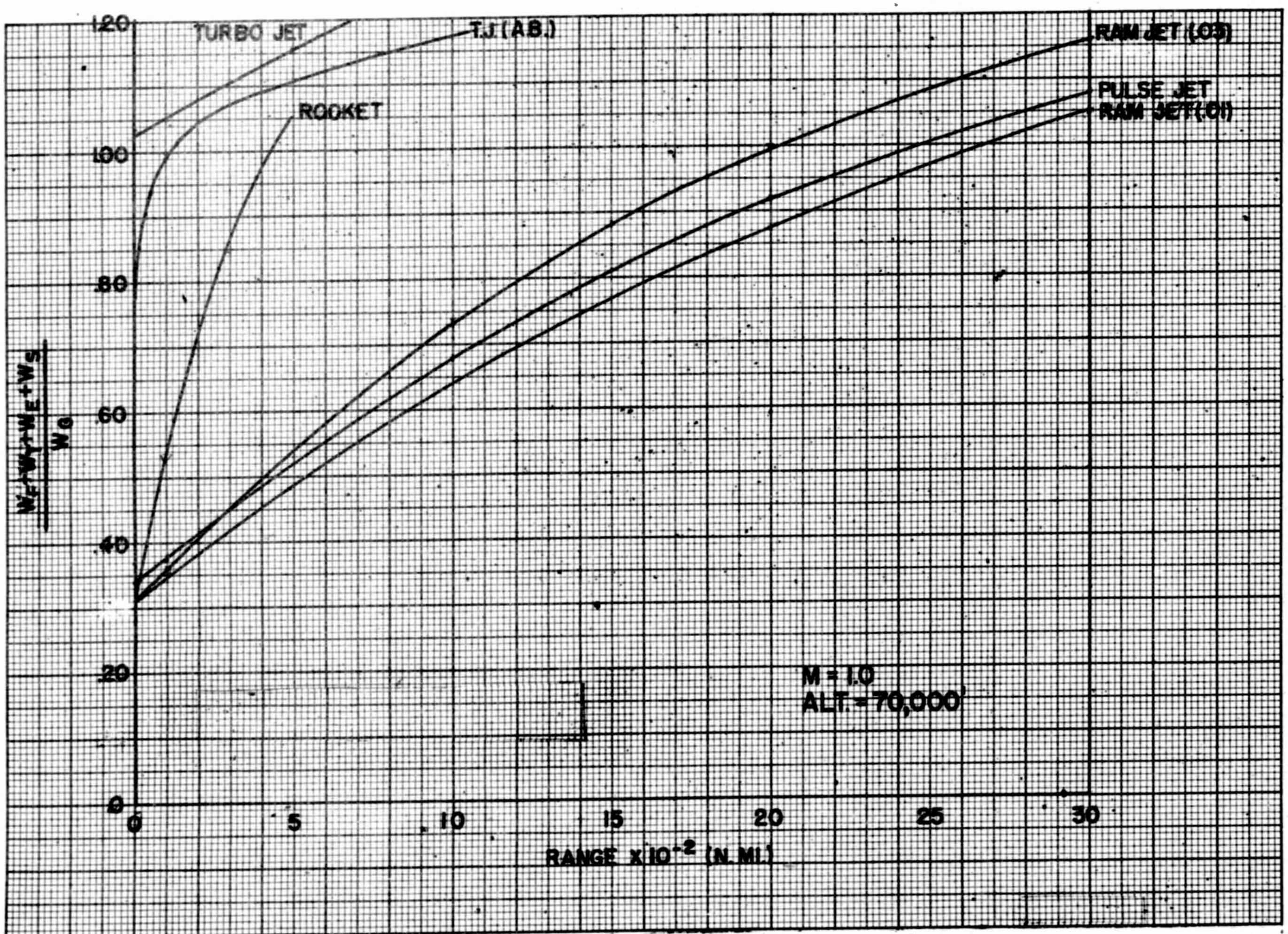

FIGURE 31 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

CONFIDENTIAL

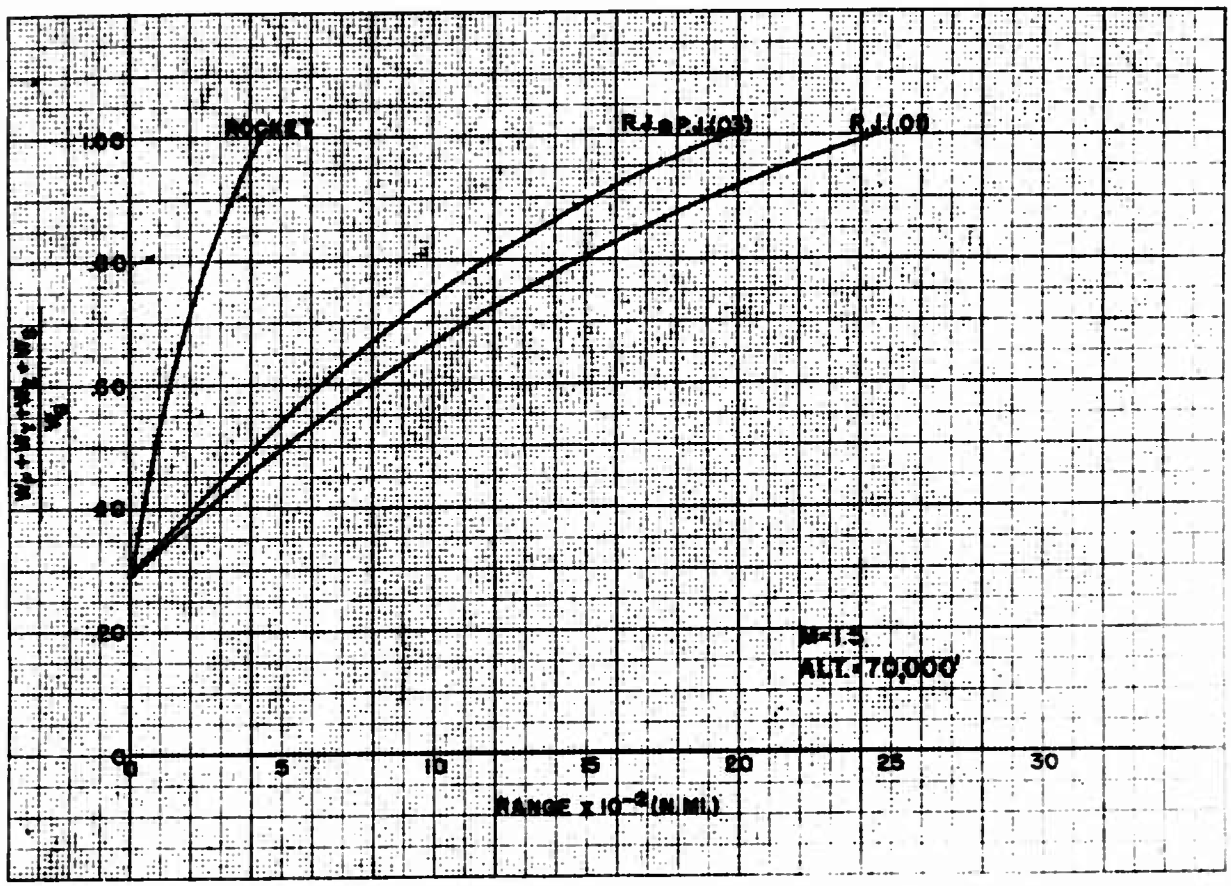

FIGURE 32 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

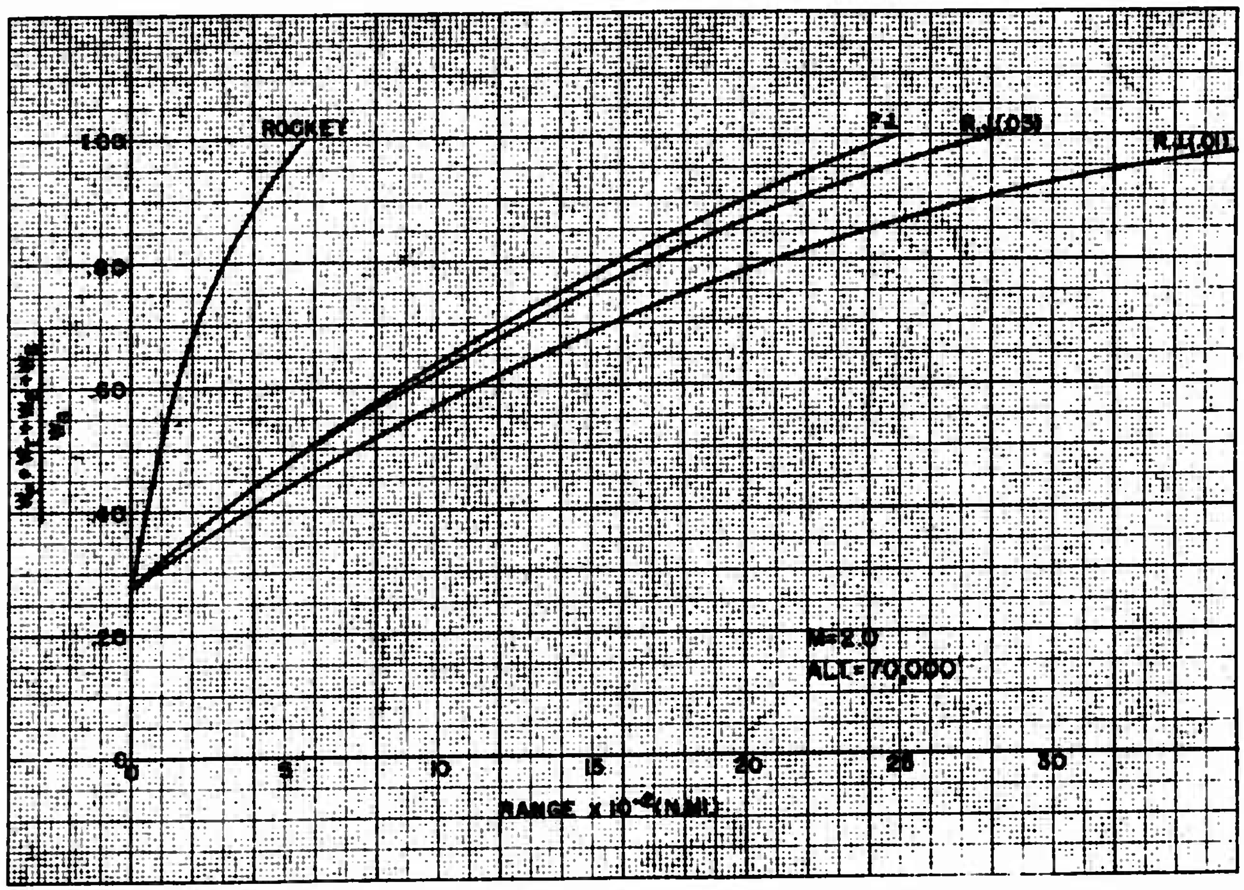

FIGURE 33 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

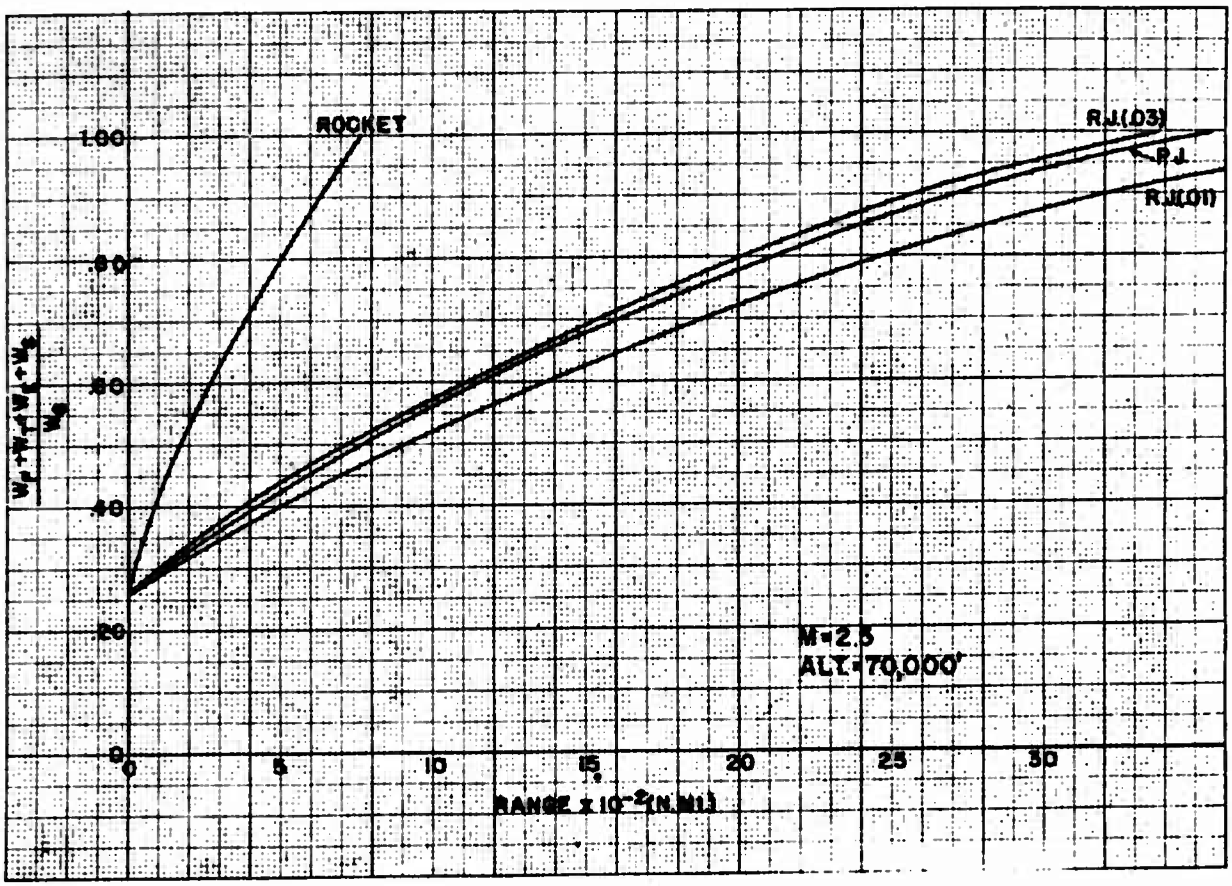

FIGURE 34 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

CONFIDENTIAL

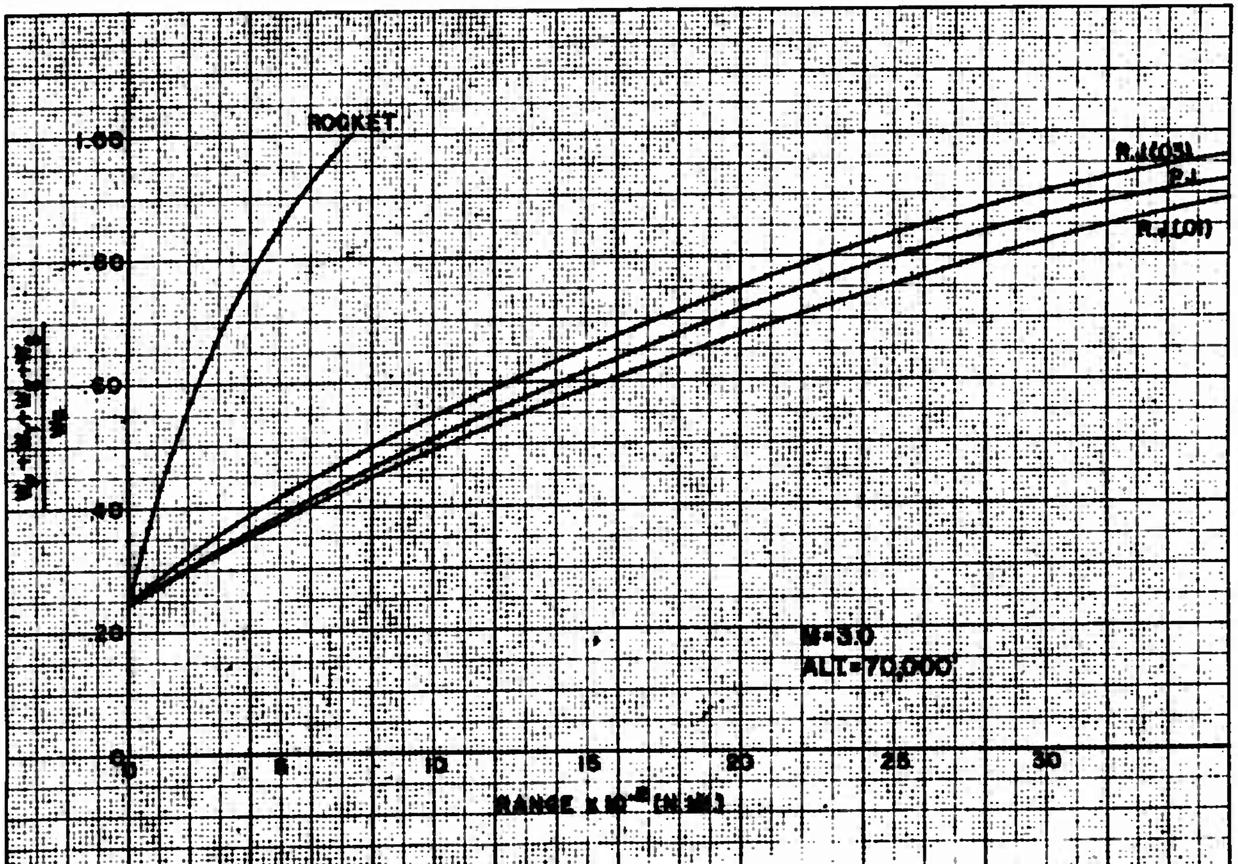

FIGURE 38 - RATIO OF WEIGHT OF ENGINE, FUEL, TANKS AND STRUCTURE TO INITIAL GROSS WEIGHT VS. RANGE

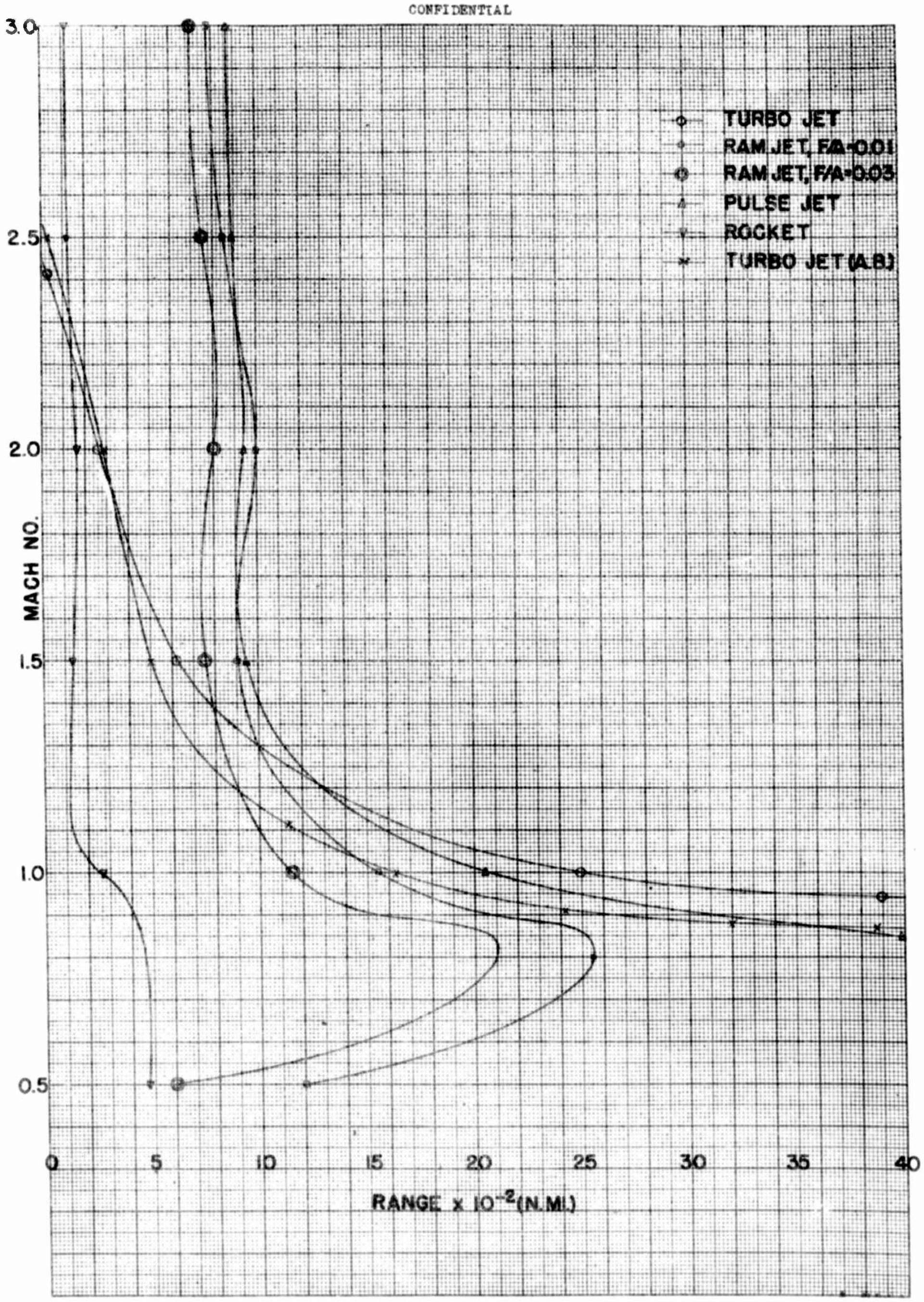

FIGURE 36 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES; ALTITUDE = SEA LEVEL, PERCENTAGE PAYLOAD = 00%

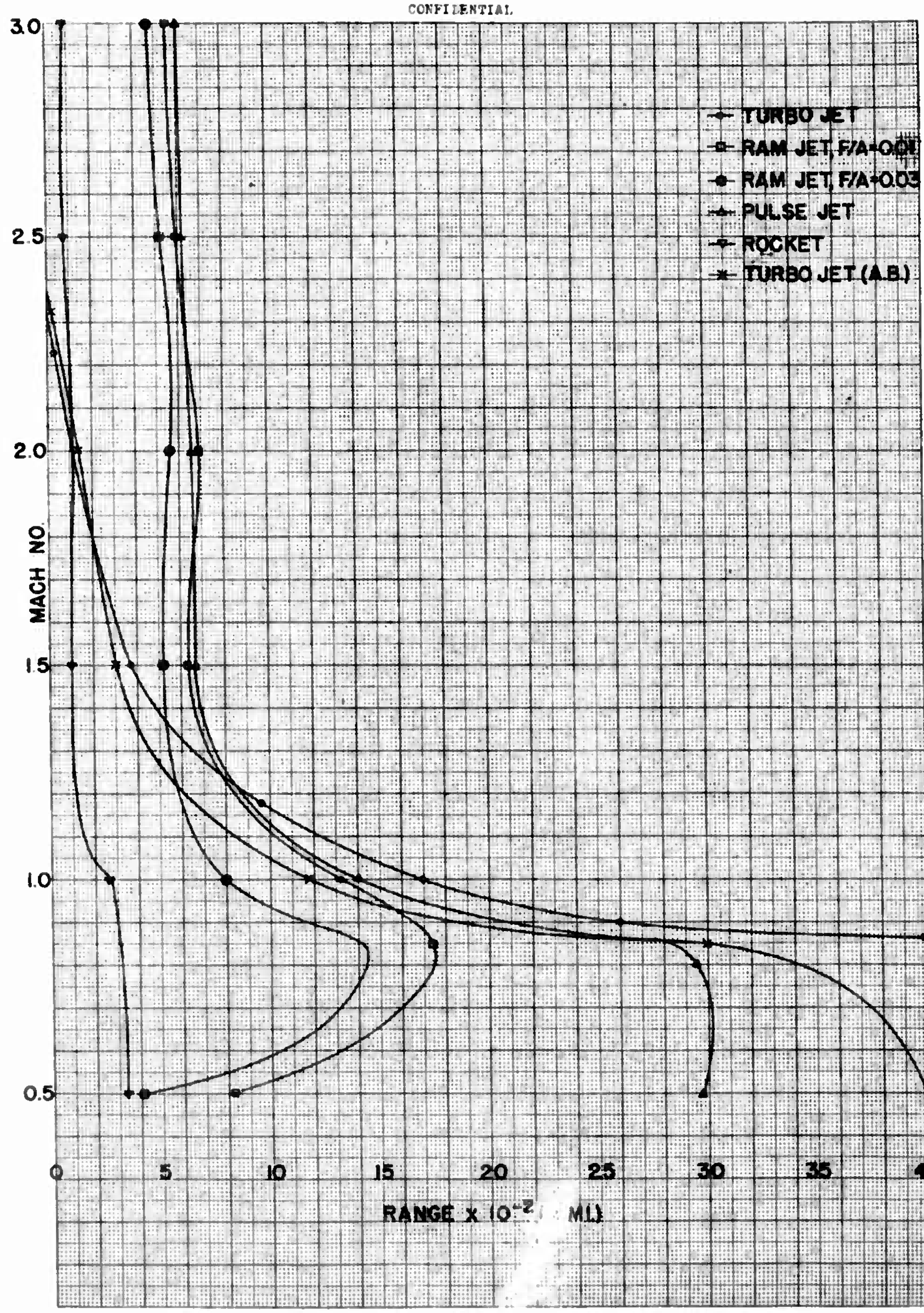

FIGURE 37 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES;
ALTITUDE = SEA LEVEL, PERCENTAGE PAYLOAD = 15%

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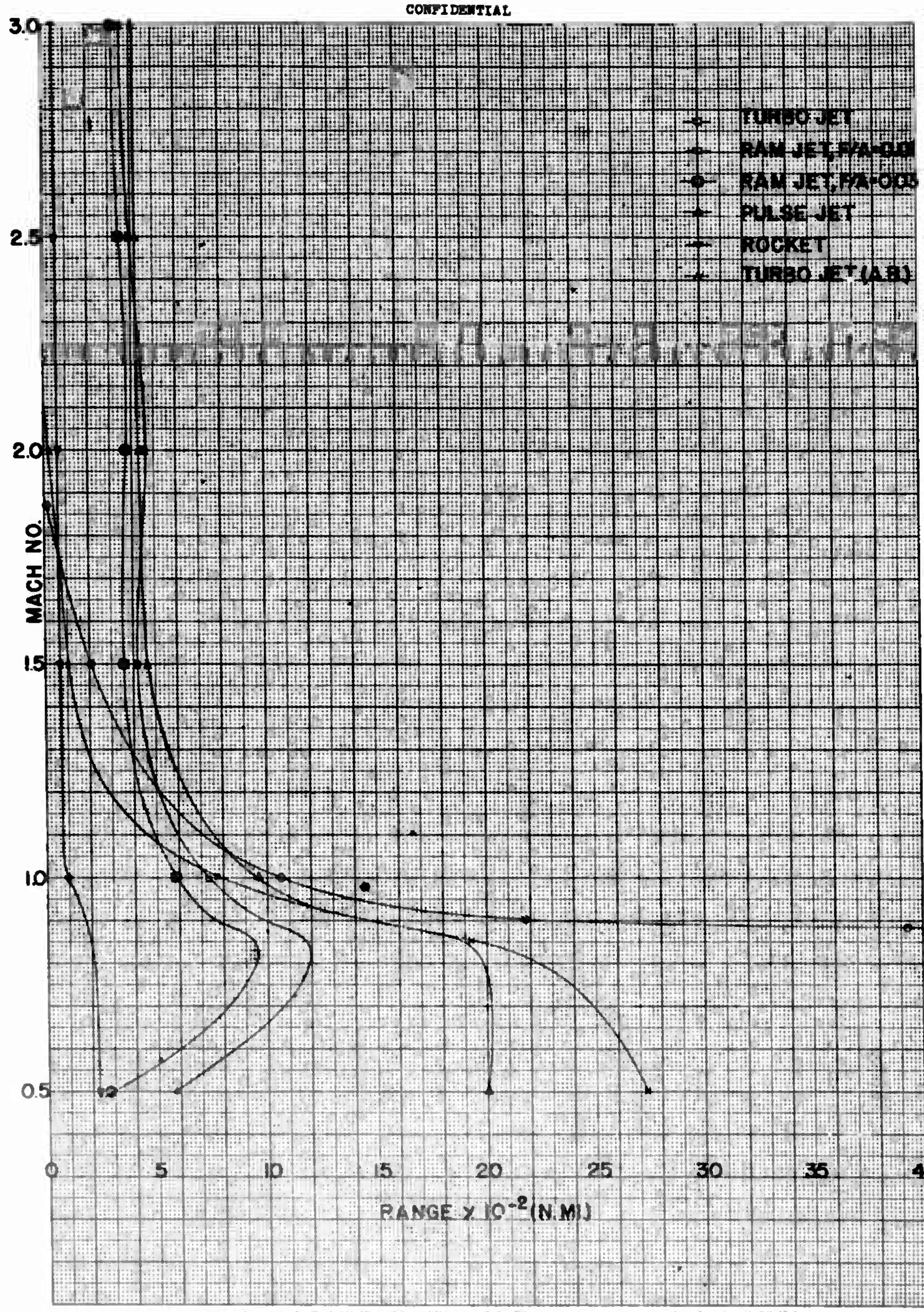

FIGURE 38 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES; ALTITUDE = SEA LEVEL, PERCENTAGE PAYLOAD = 30%

CONFIDENTIAL

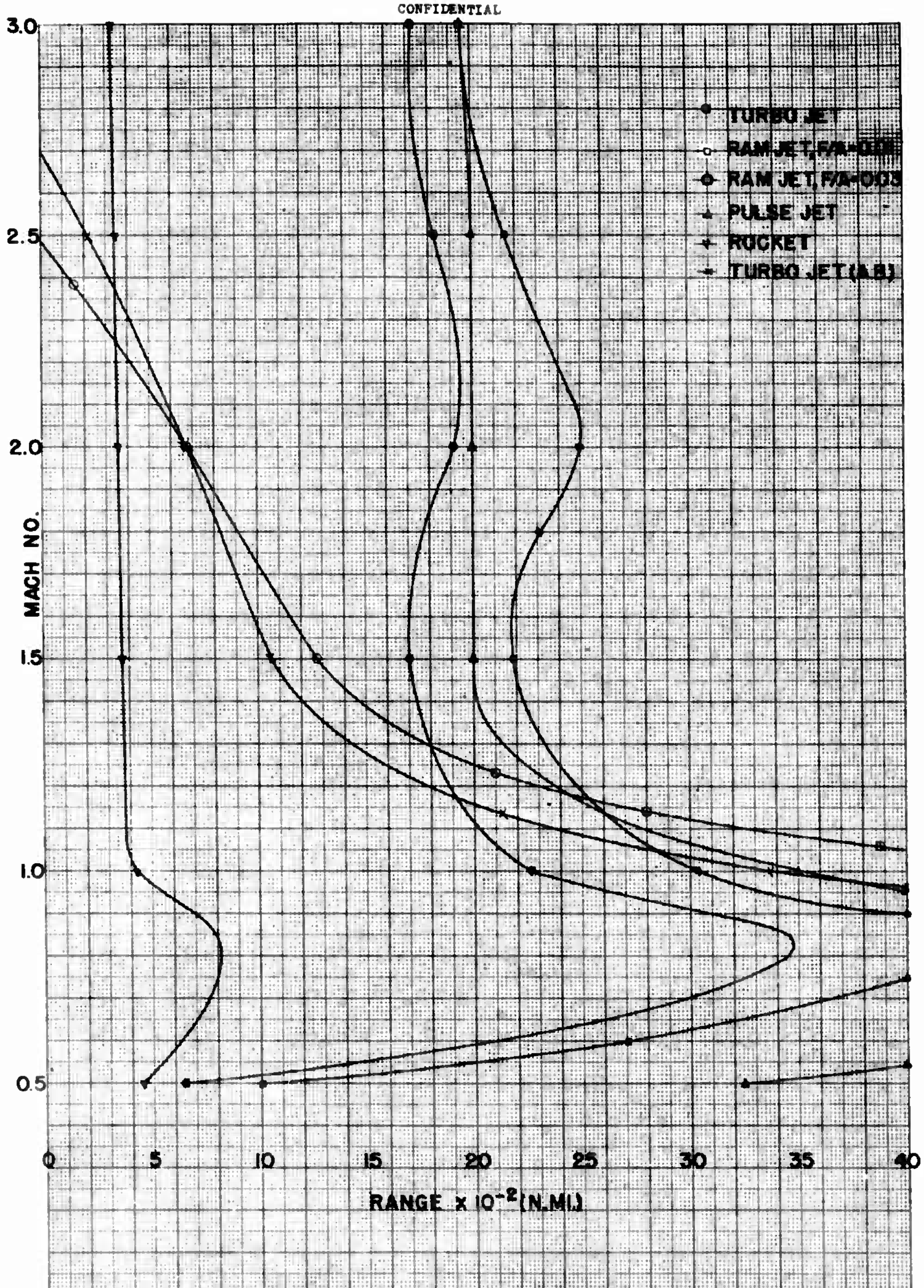

FIGURE 39 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES; ALTITUDE = 25,000 FEET, PERCENTAGE PAYLOAD = 00%

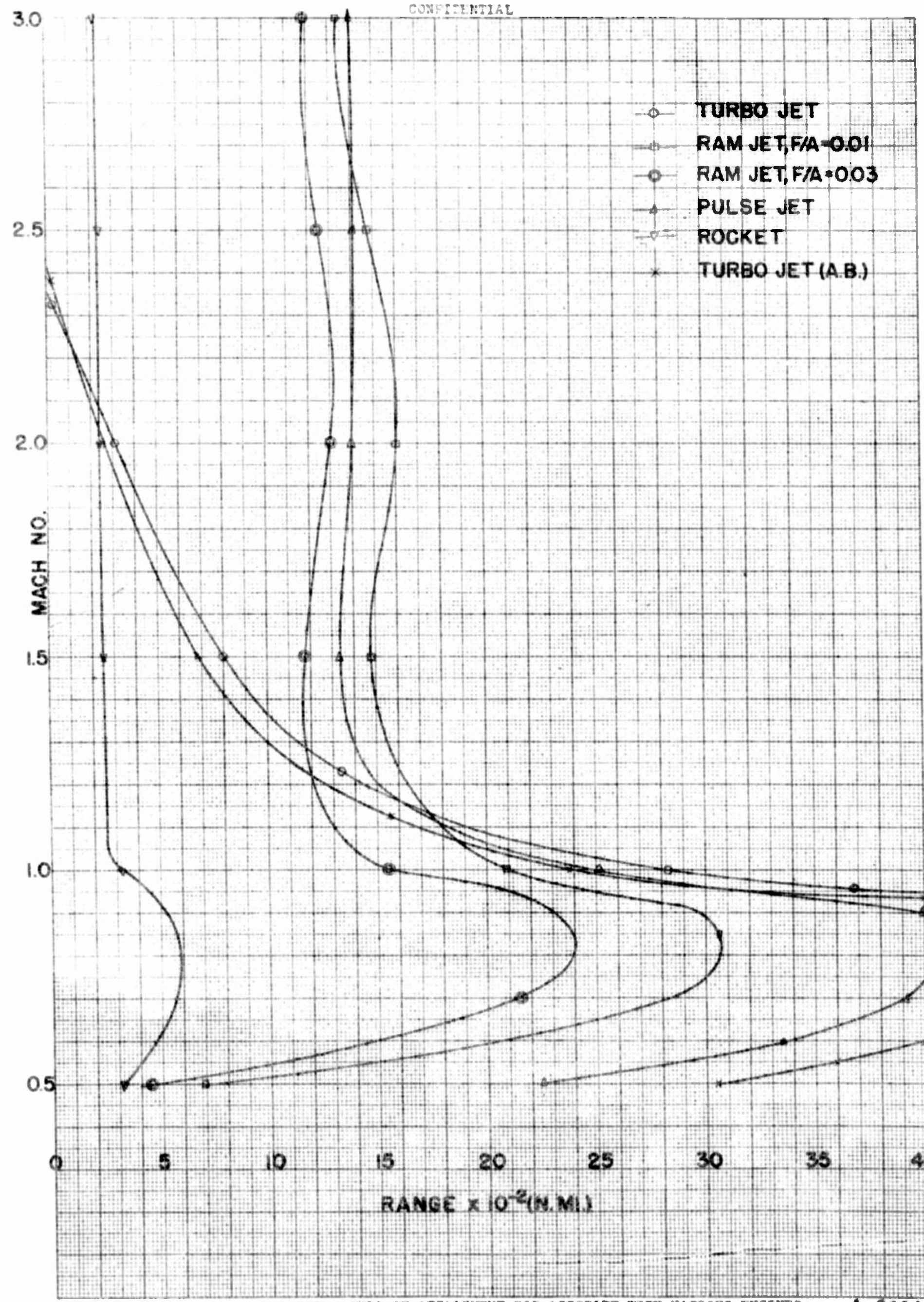

FIGURE 40 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES; ALTITUDE = 25,000 FEET, PERCENTAGE PAYLOAD = 15%

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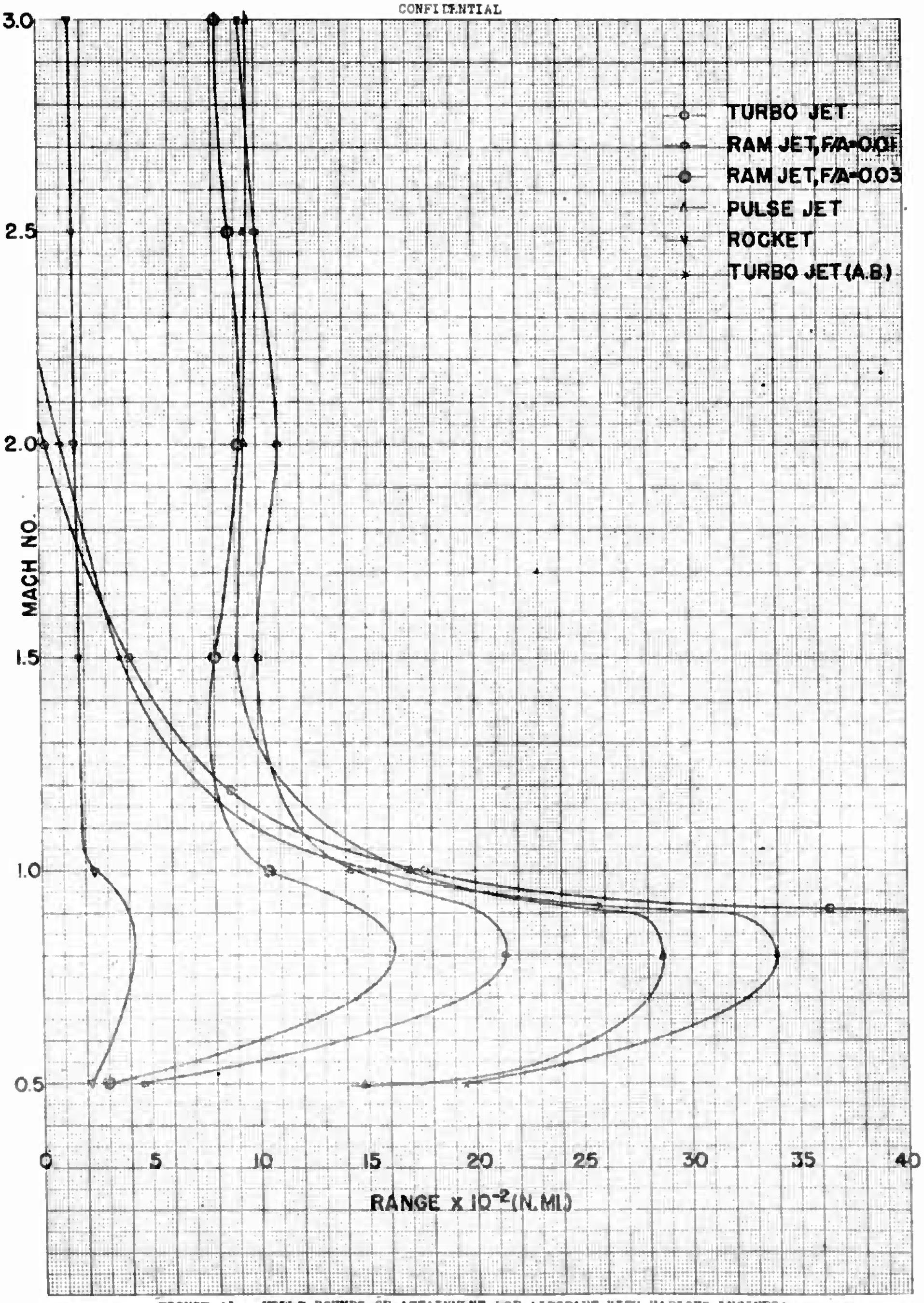

FIGURE 41 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES; ALTITUDE = 25,000 FEET, PERCENTAGE PAYLOAD = 30%

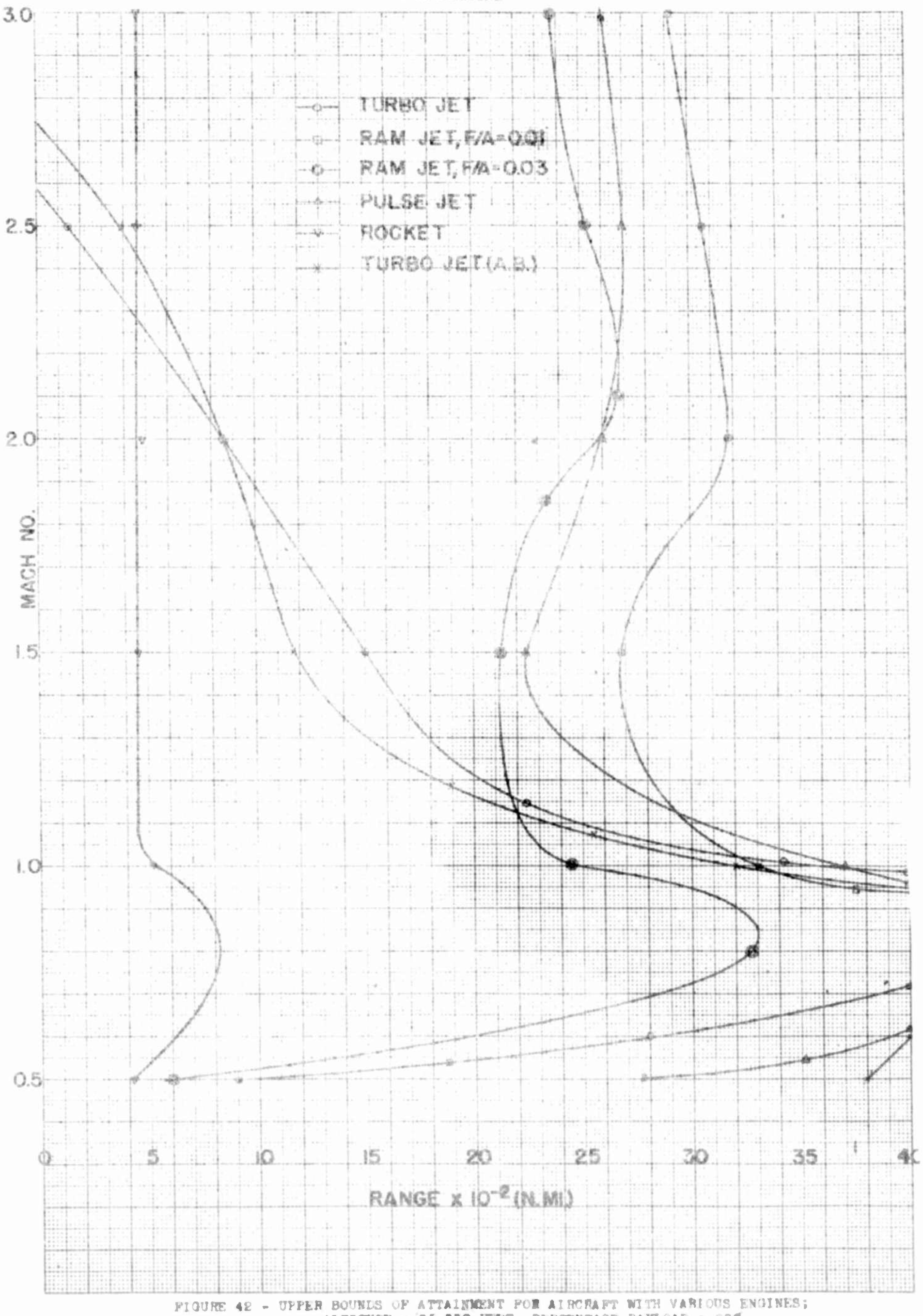

FIGURE 42 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES; ALTITUDE = 35,332 FEET, PERCENTAGE PAYLOAD = 00%

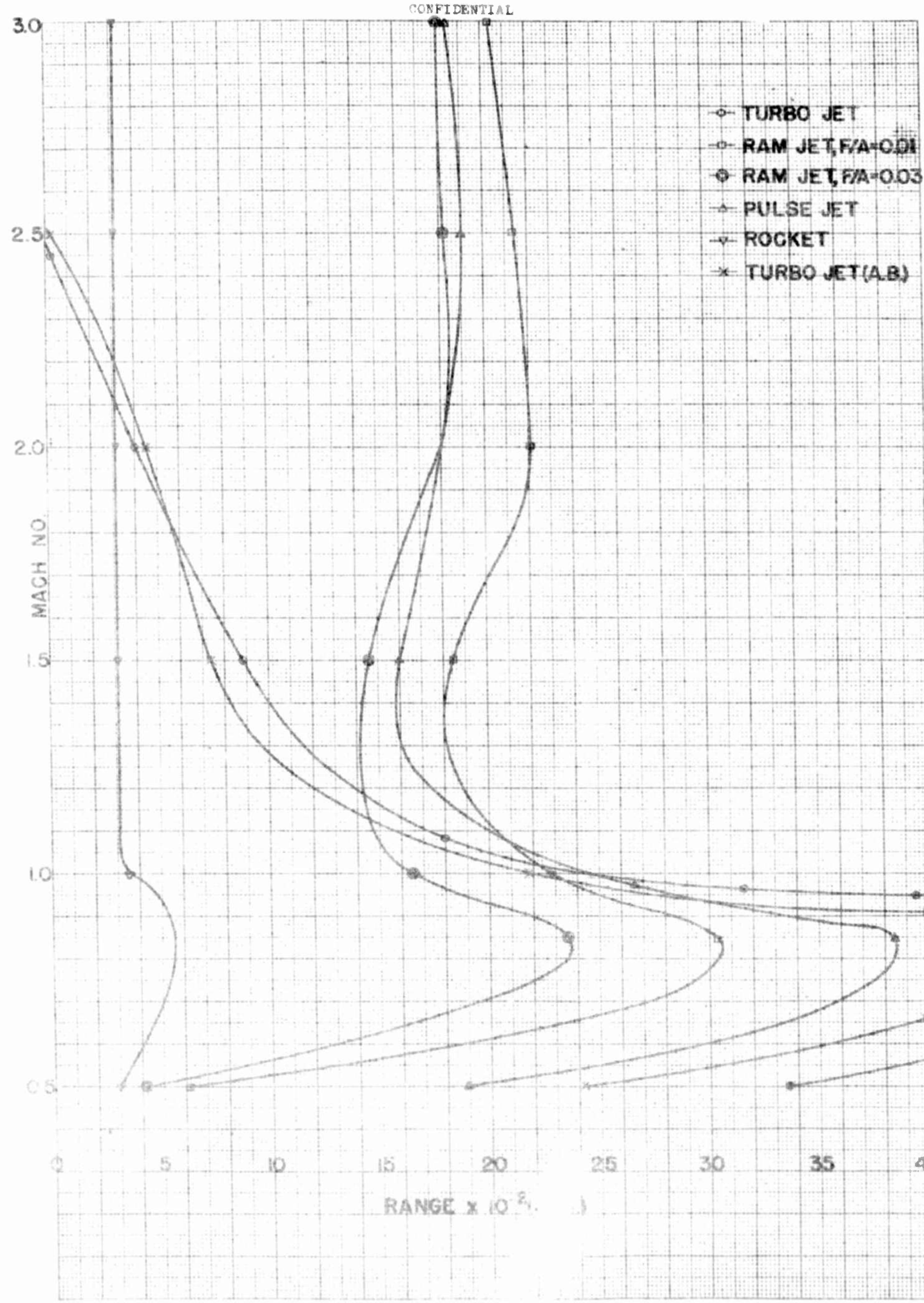

FIGURE 43 - UPPER BOUNDS OF ATTAINMENT [illegible] RCRAFT WITH VARIOUS ENGINES;
ALTITUDE = 35,[illegible] [illegible], [illegible]CENTAGE PAYLOAD = 15%

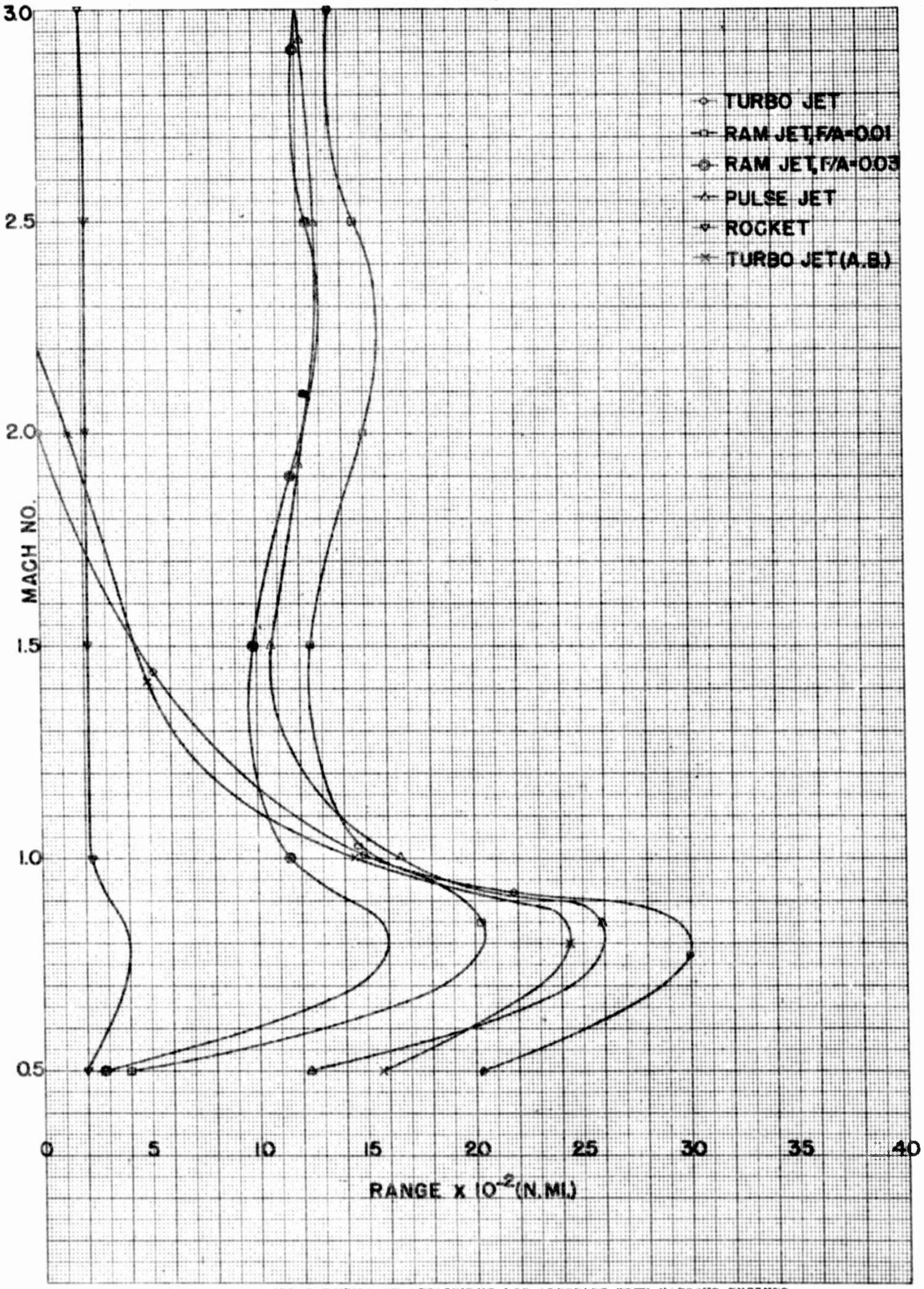

FIGURE 44 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES; ALTITUDE = 35,332 FEET, PERCENTAGE PAYLOAD = 30%

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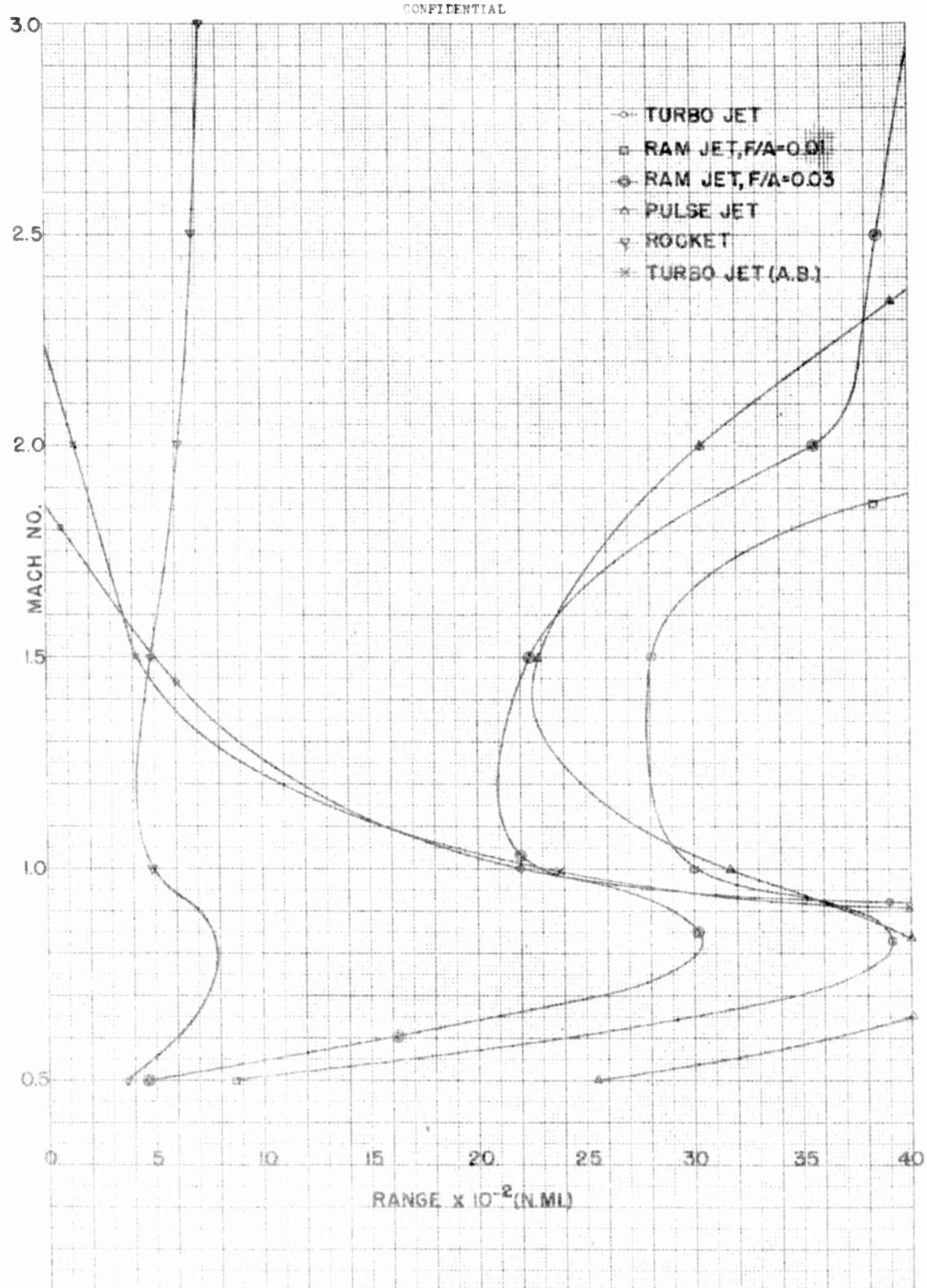

FIGURE [illegible] - [illegible] OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES;
ALTITUDE=[illegible],000 FEET, PERCENTAGE PAYLOAD = [illegible]

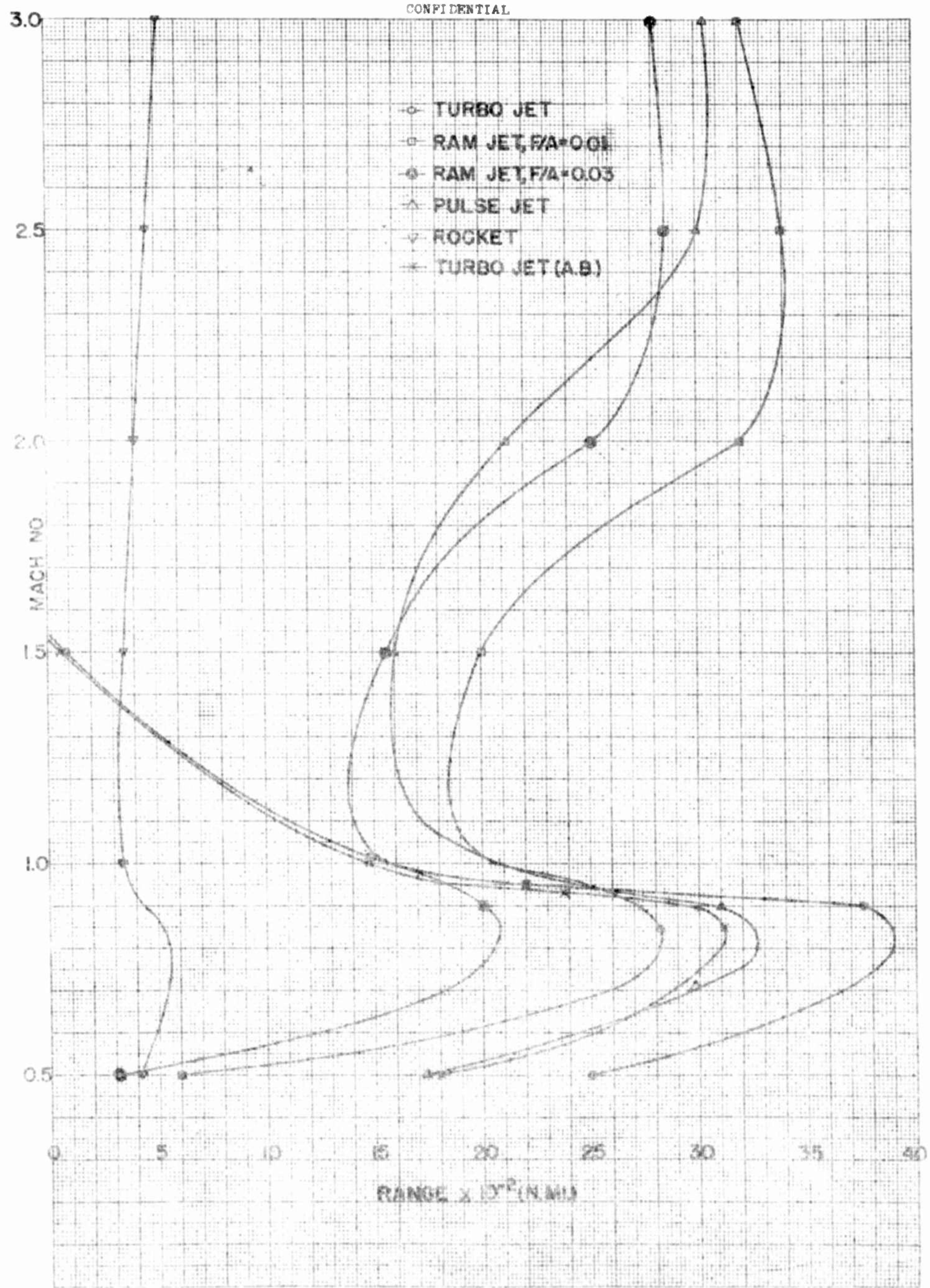

FIGURE 46 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES, ALTITUDE = [illegible] FEET, PERCENTAGE PAYLOAD = [illegible]

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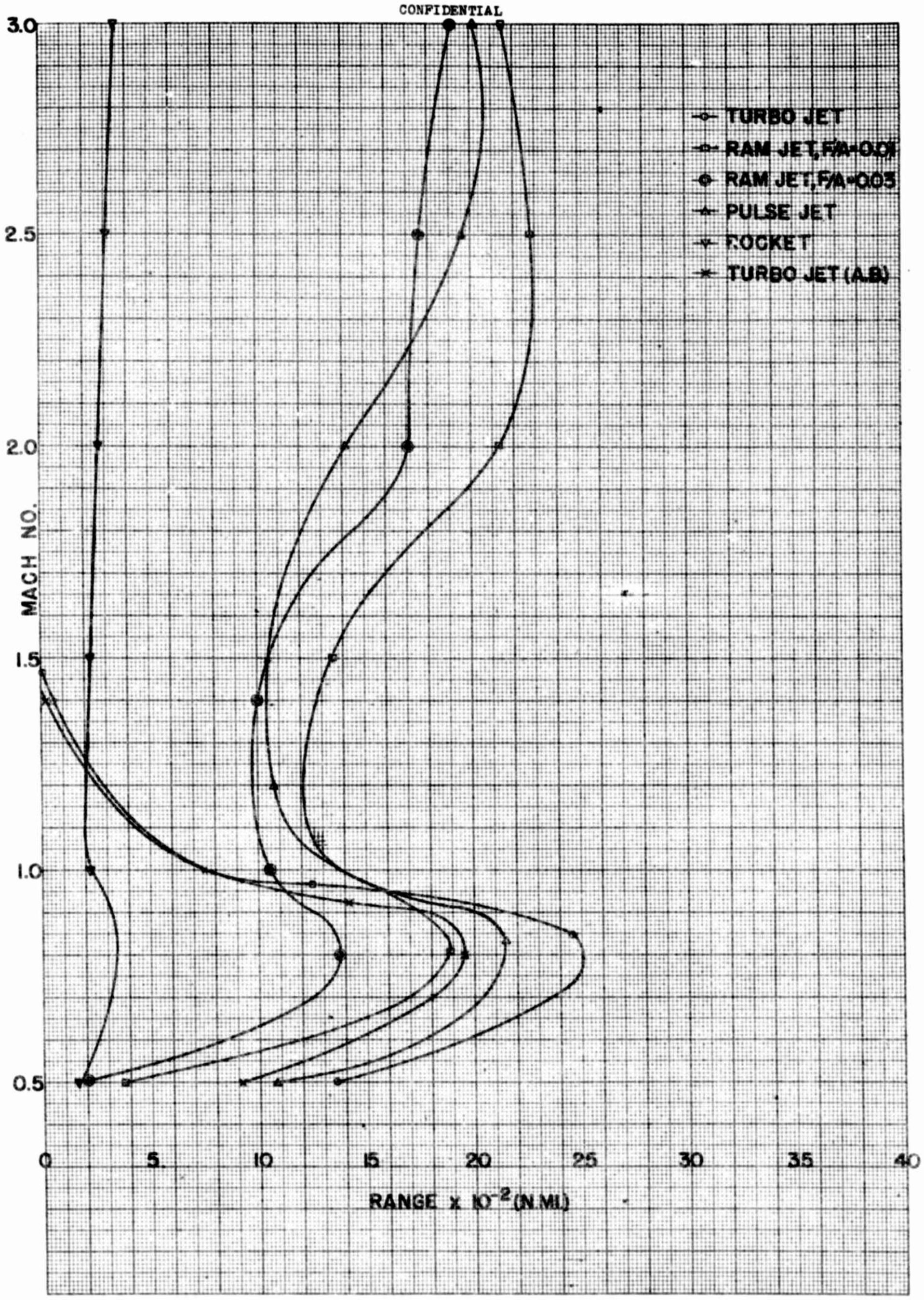

FIGURE 47 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES; ALTITUDE = 50,000 FEET, PERCENTAGE PAYLOAD = 30%

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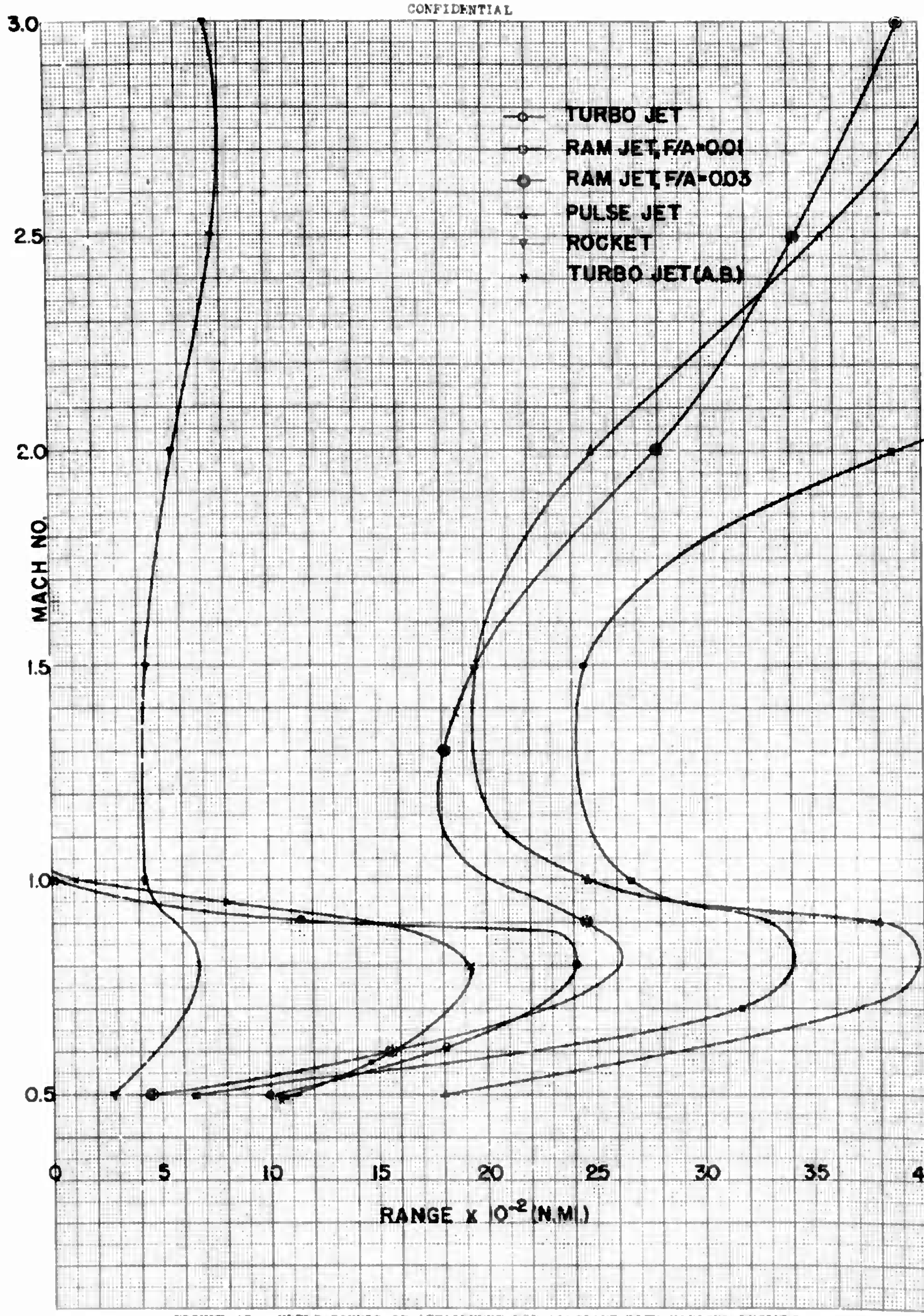

FIGURE 48 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES; ALTITUDE = 70,000 FEET, PERCENTAGE PAYLOAD = 60%

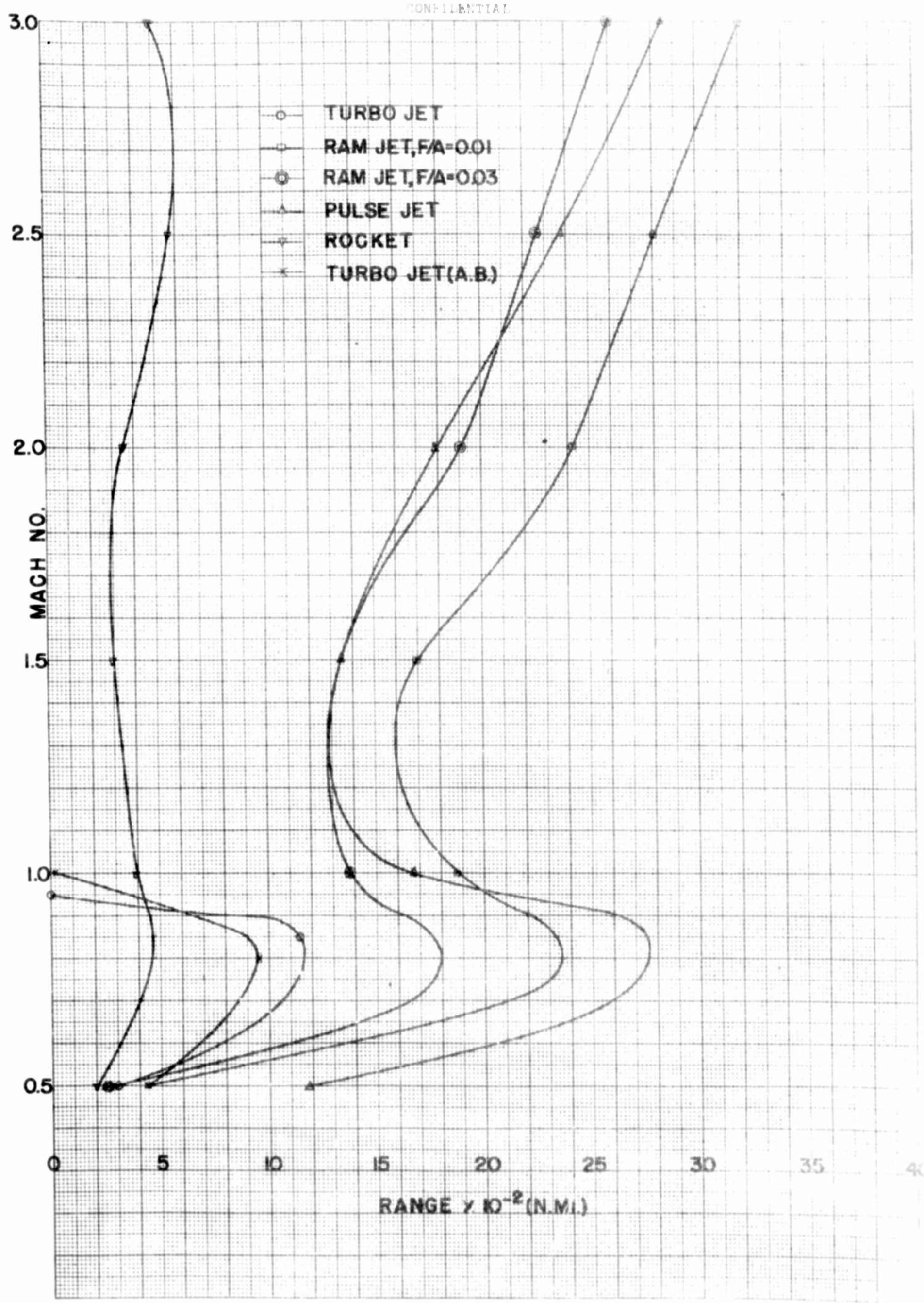

FIGURE 49 - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH [illegible];
ALTITUDE = 70,000 FEET, PERCENTAGE [illegible]

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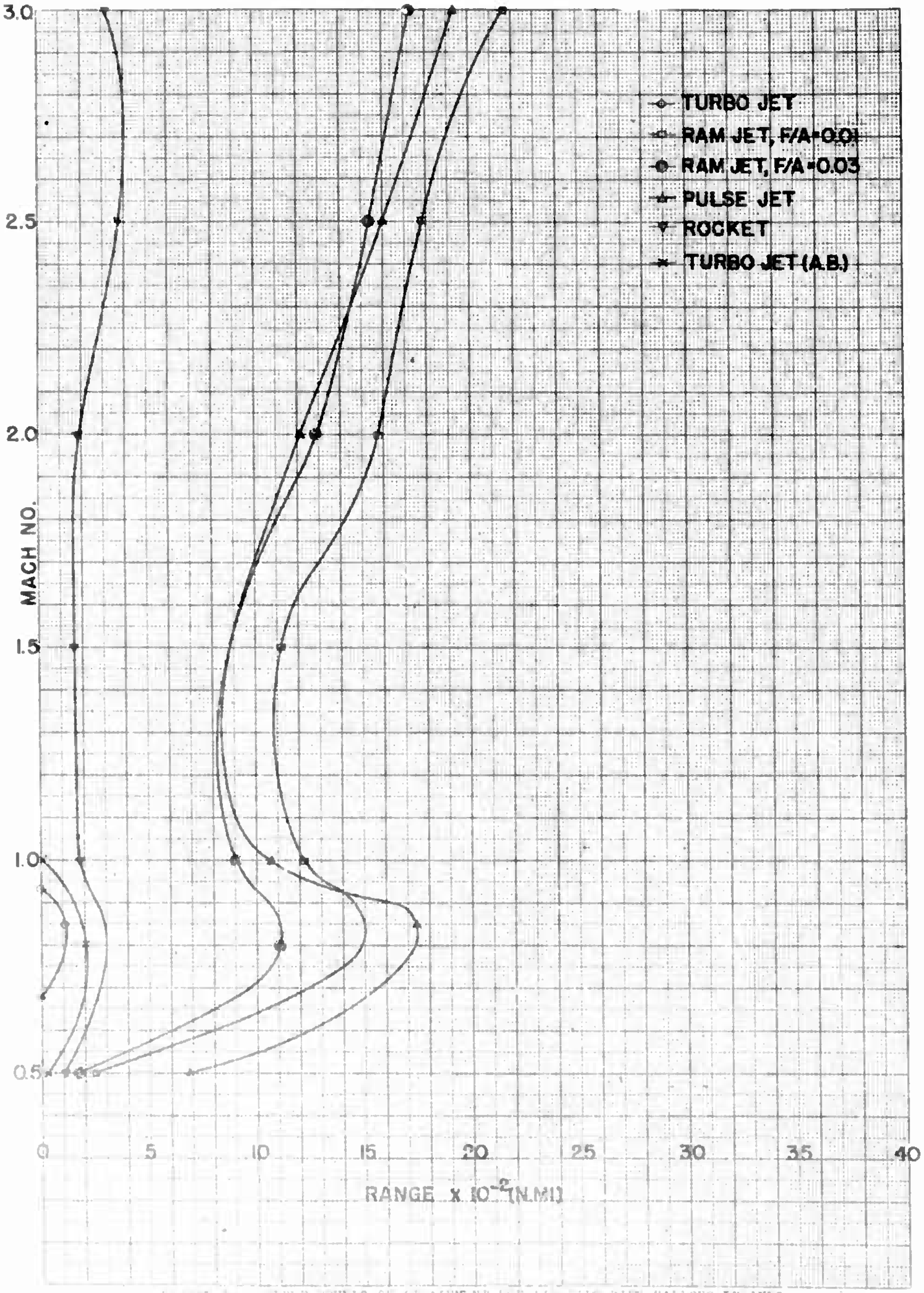

FIGURE [illegible] - UPPER BOUNDS OF ATTAINMENT FOR AIRCRAFT WITH VARIOUS ENGINES;
ALTITUDE = [illegible] FEET, PERCENTAGE PAYLOAD = [illegible]

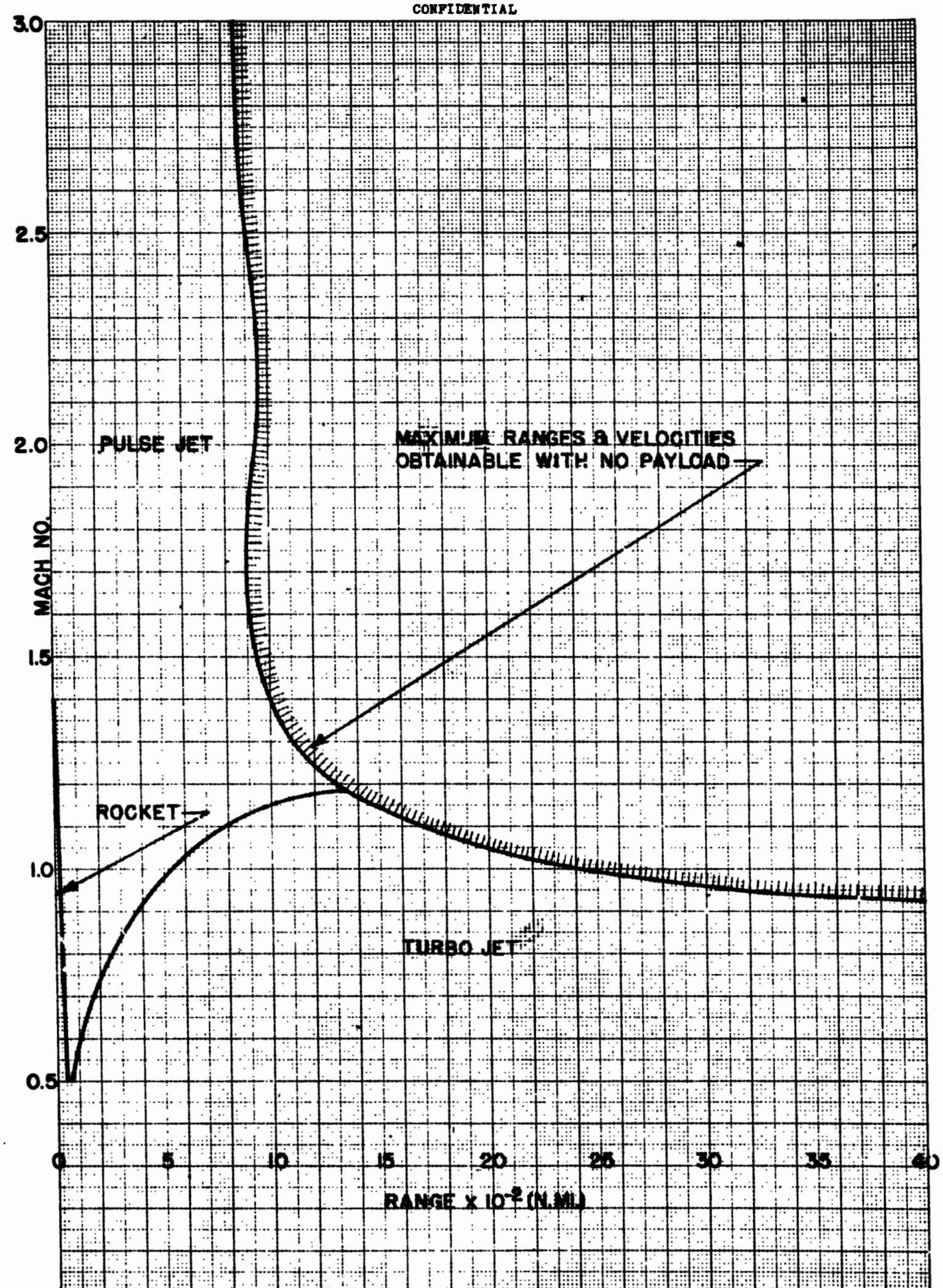

FIGURE 51 - OPTIMUM ENGINE CHOICE FOR REGIONS OF OPERATION IF RAM JET FUEL-AIR RATIO = 0.03, ALTITUDE = SEA LEVEL

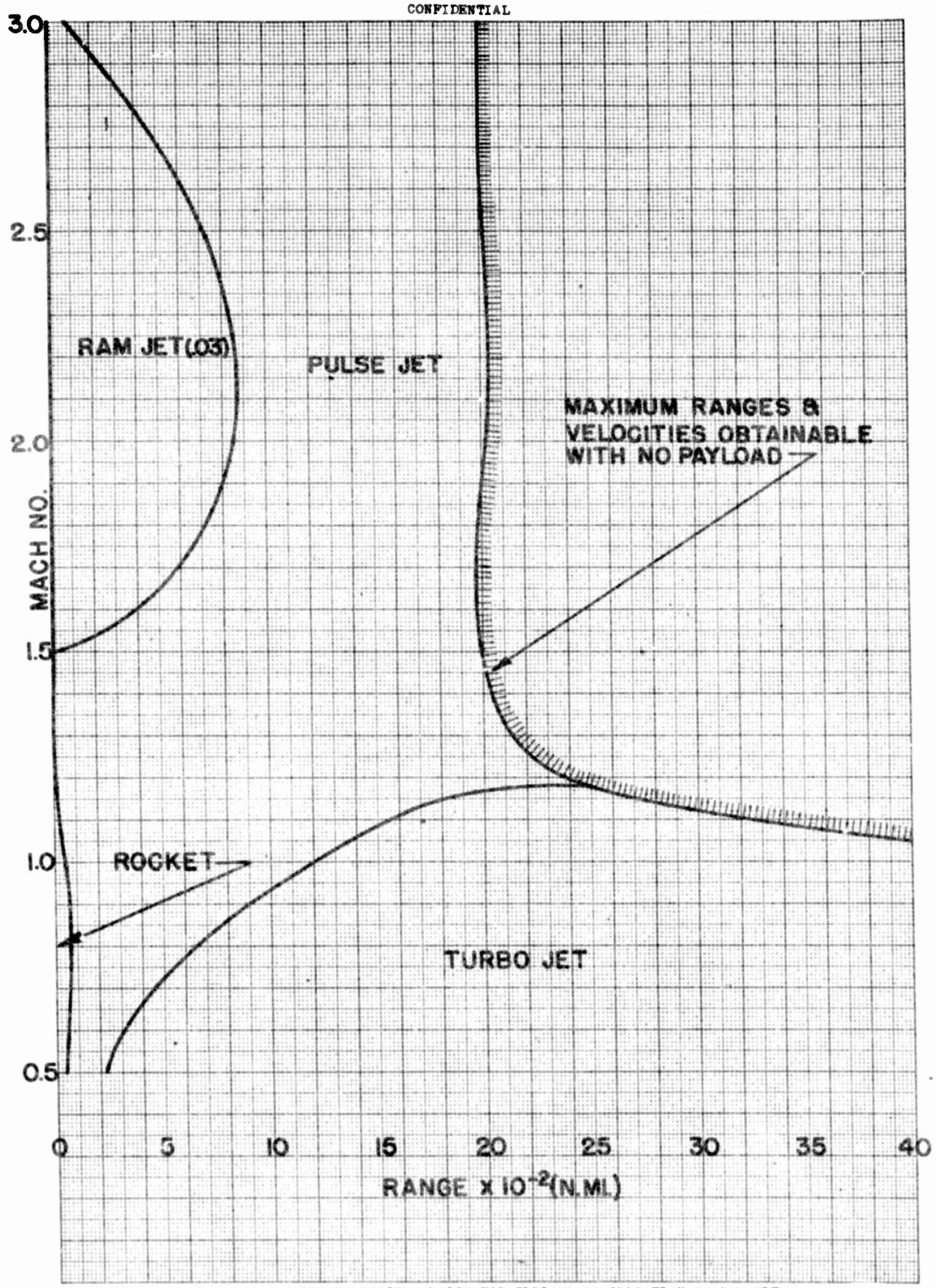

FIGURE 62 - OPTIMUM ENGINE CHOICE FOR REGIONS OF OPERATION IF RAM JET FUEL-AIR RATION = 0.03, ALTITUDE = 25,000 FEET

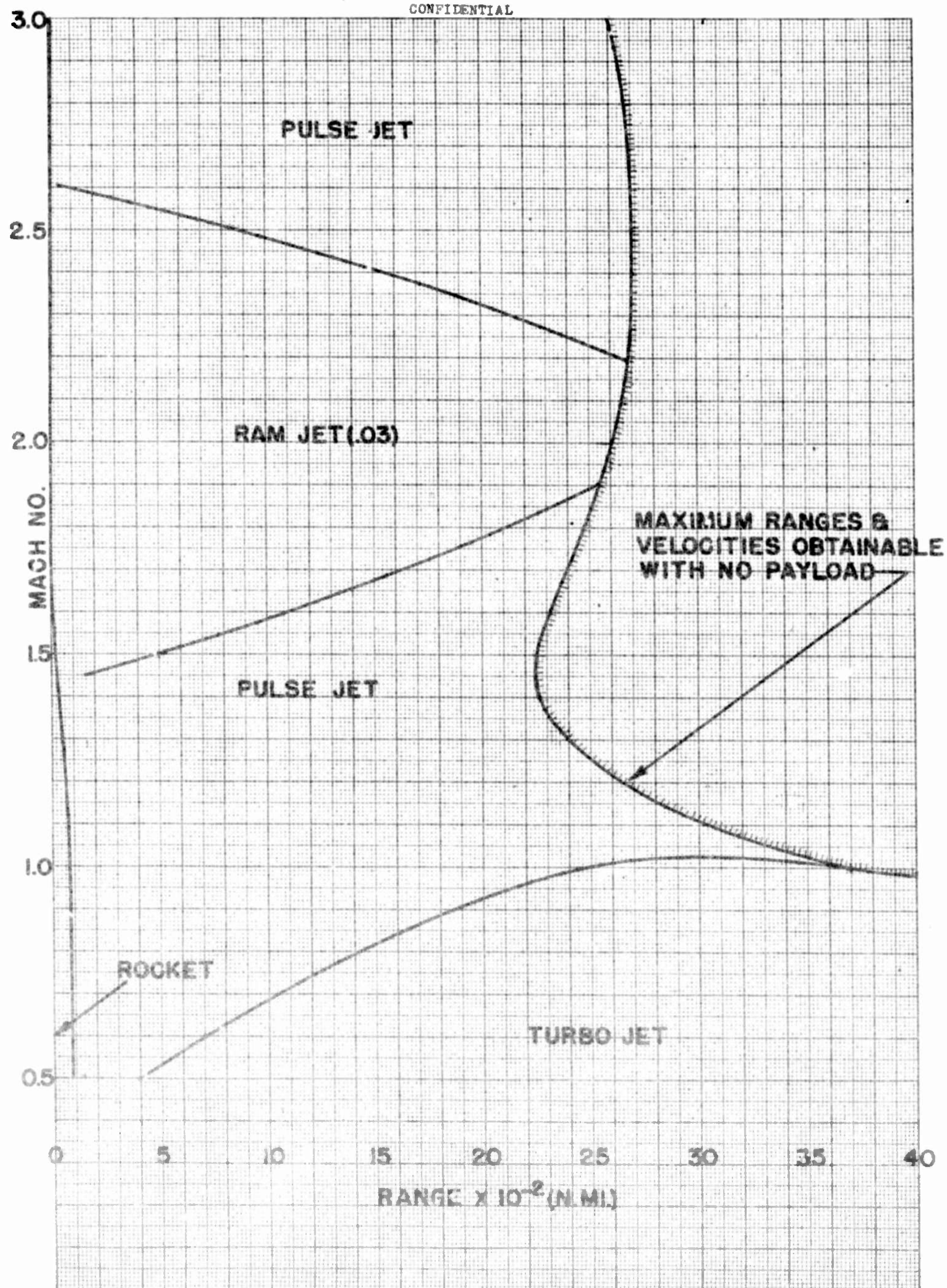

FIGURE 53 - OPTIMUM ENGINE CHOICE FOR REGIONS OF OPERATION IF RAM JET FUEL-AIR RATIO = 0.03, ALTITUDE = 35,332 FEET

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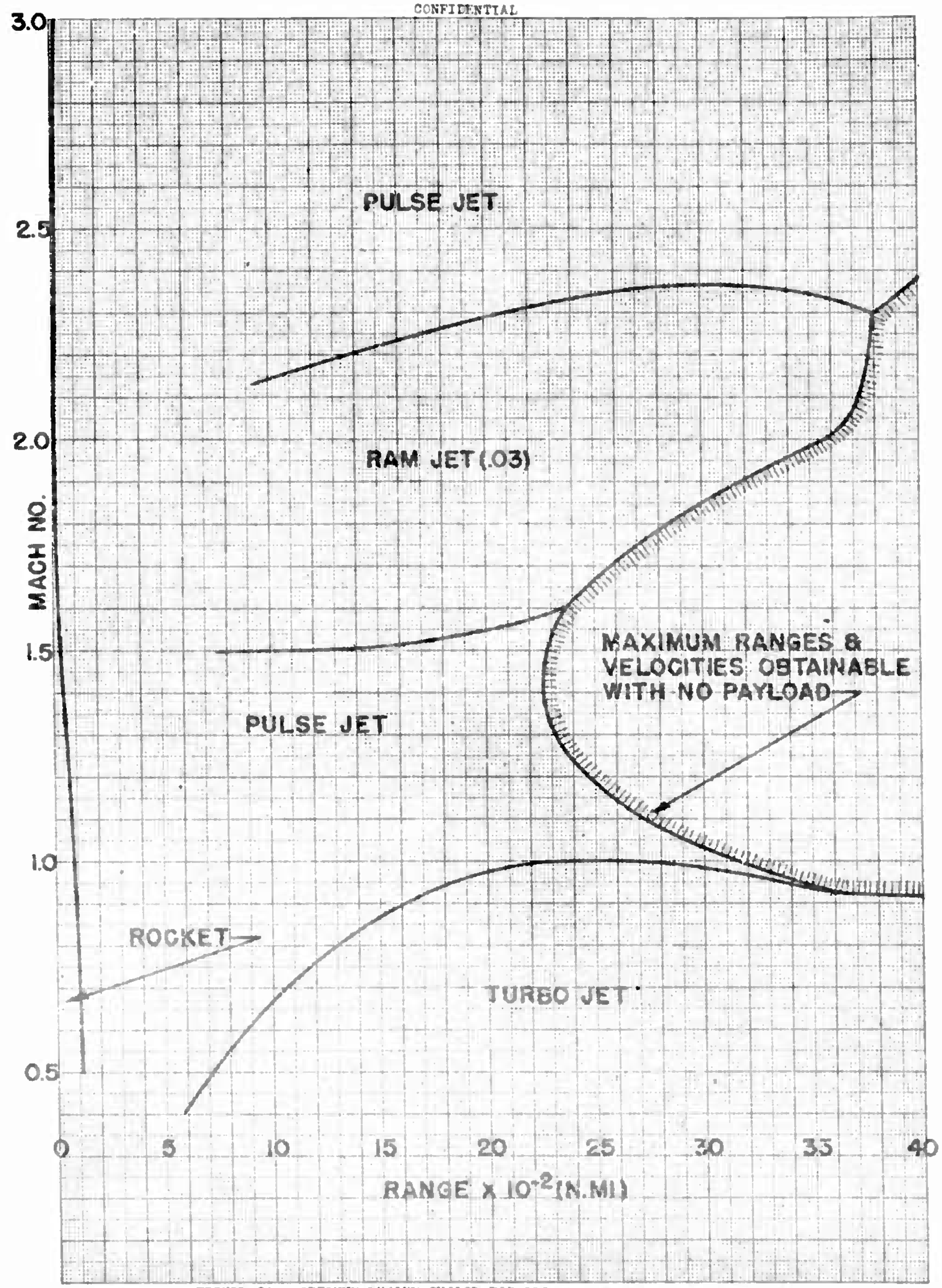

FIGURE 54 - OPTIMUM ENGINE CHOICE FOR REGIONS OF OPERATION IF RAM JET
FUEL-AIR RATIO = 0.03, ALTITUDE = 50,000 FEET

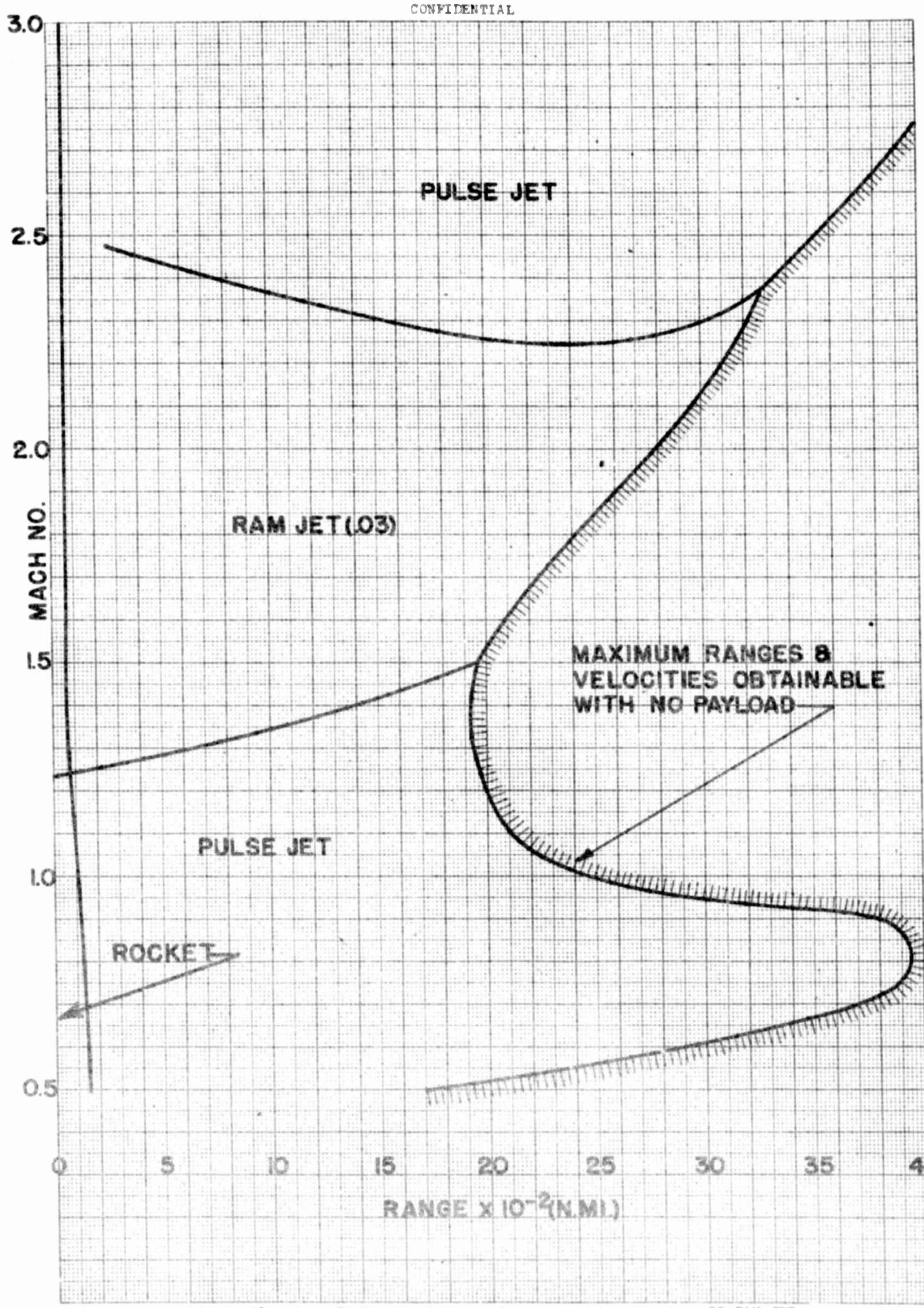

FIGURE 55 - OPTIMUM ENGINE CHOICE FOR REGIONS OF OPERATIONS IF RAM JET
FUEL-AIR RATIO = 0.03, ALTITUDE = 70,000 FEET

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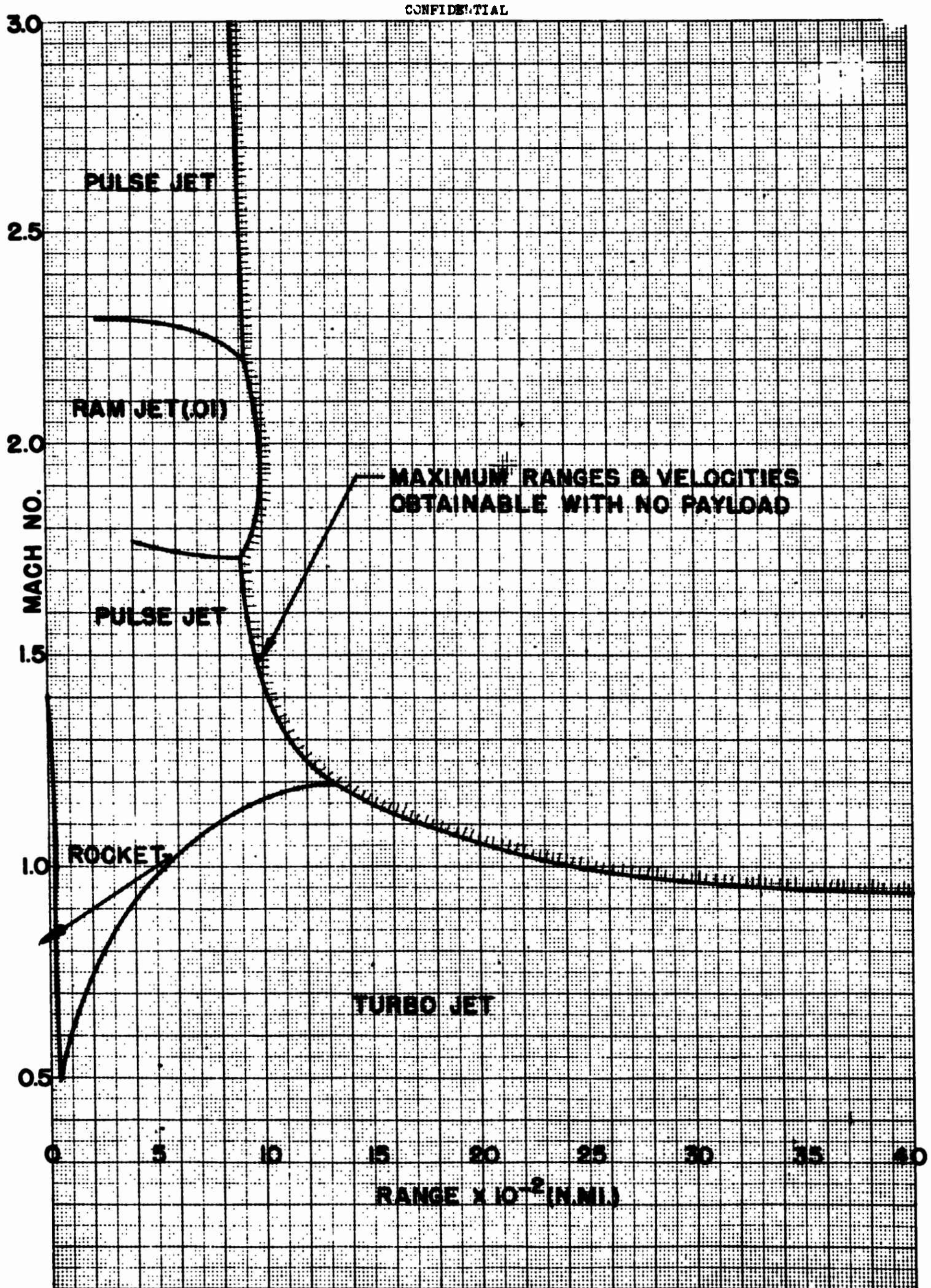

FIGURE 56 - OPTIMUM ENGINE CHOICE FOR REGIONS OF OPERATION IF RAM JET FUEL-AIR RATIO = 0.01, ALTITUDE = SEA LEVEL

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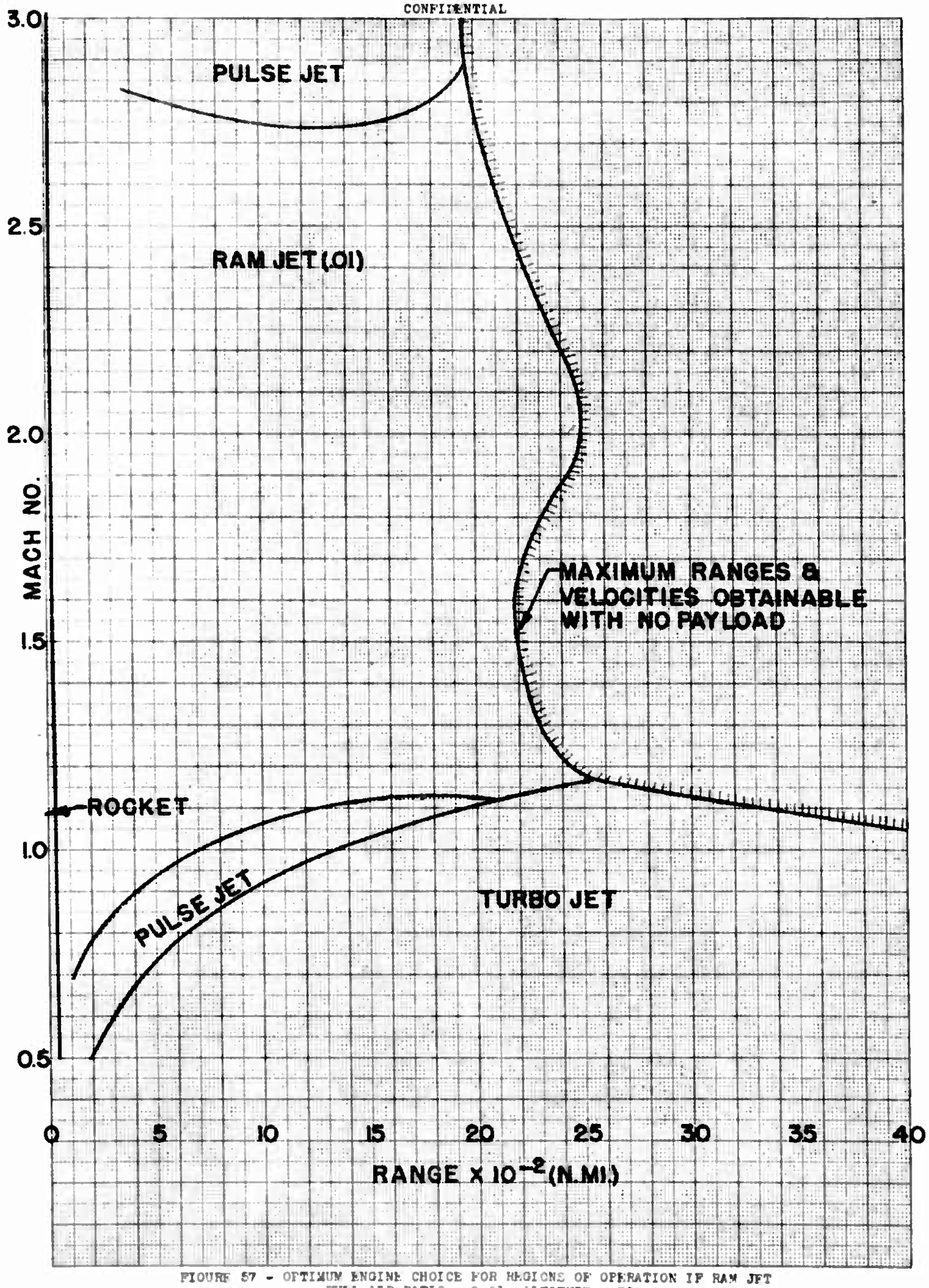

FIGURE 57 - OPTIMUM ENGINE CHOICE FOR REGIONS OF OPERATION IF RAM JET
FUEL-AIR RATIO = 0.01, ALTITUDE = 25,000 FEET

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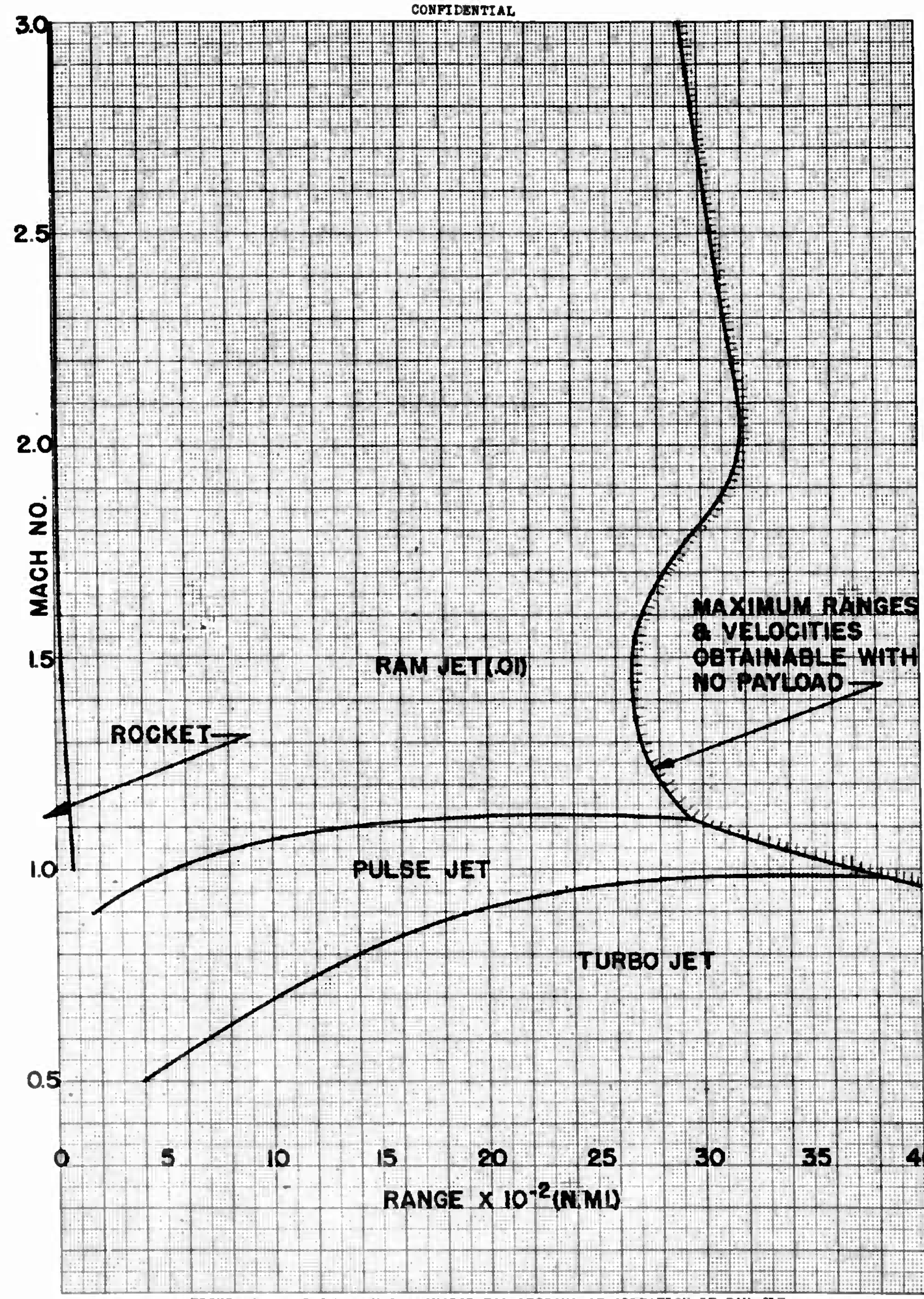

FIGURE 58 - OPTIMUM ENGINE CHOICE FOR REGIONS OF OPERATION IF RAM JET
FUEL-AIR RATIO = 0.01, ALTITUDE = 35,332 FEET

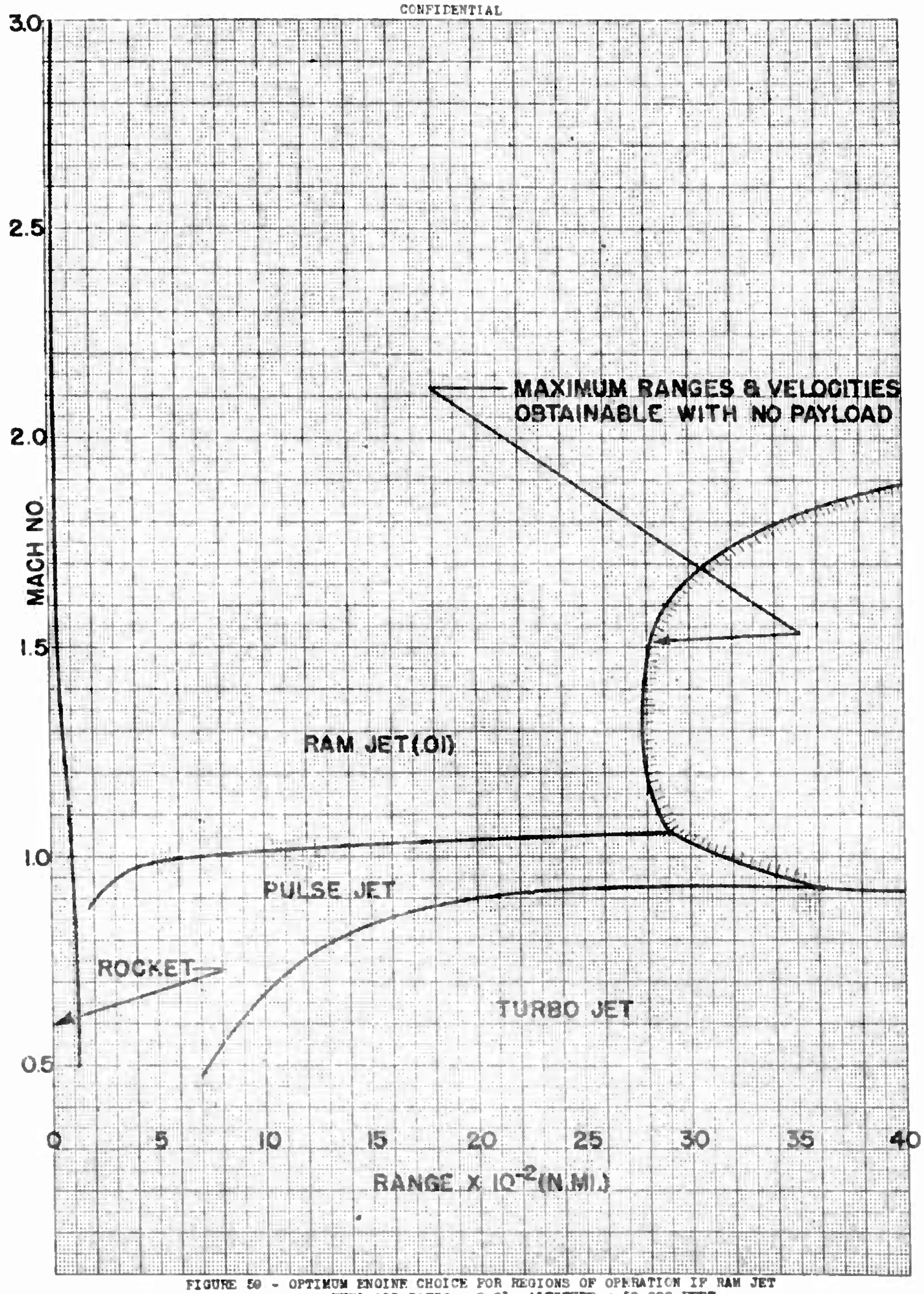

FIGURE 50 - OPTIMUM ENGINE CHOICE FOR REGIONS OF OPERATION IF RAM JET FUEL-AIR RATIO = 0.01, ALTITUDE = 50,000 FEET

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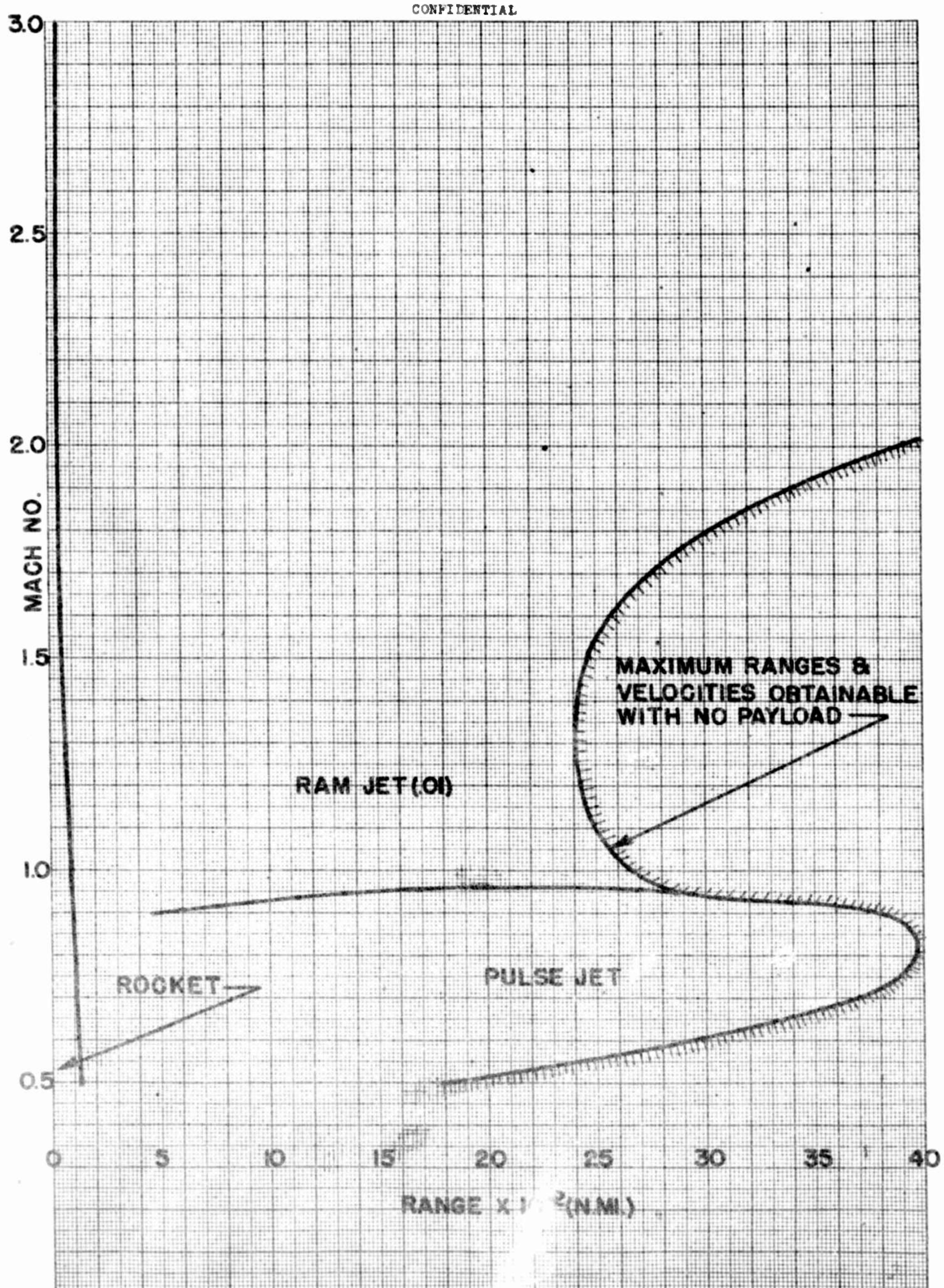

FIGURE 60 - OPTIMUM ENGINE CHOICE FOR REGIONS OF OPERATION IF RAM JET
FUEL-AIR RATIO = 0.01, ALTITUDE = 70,000 FEET

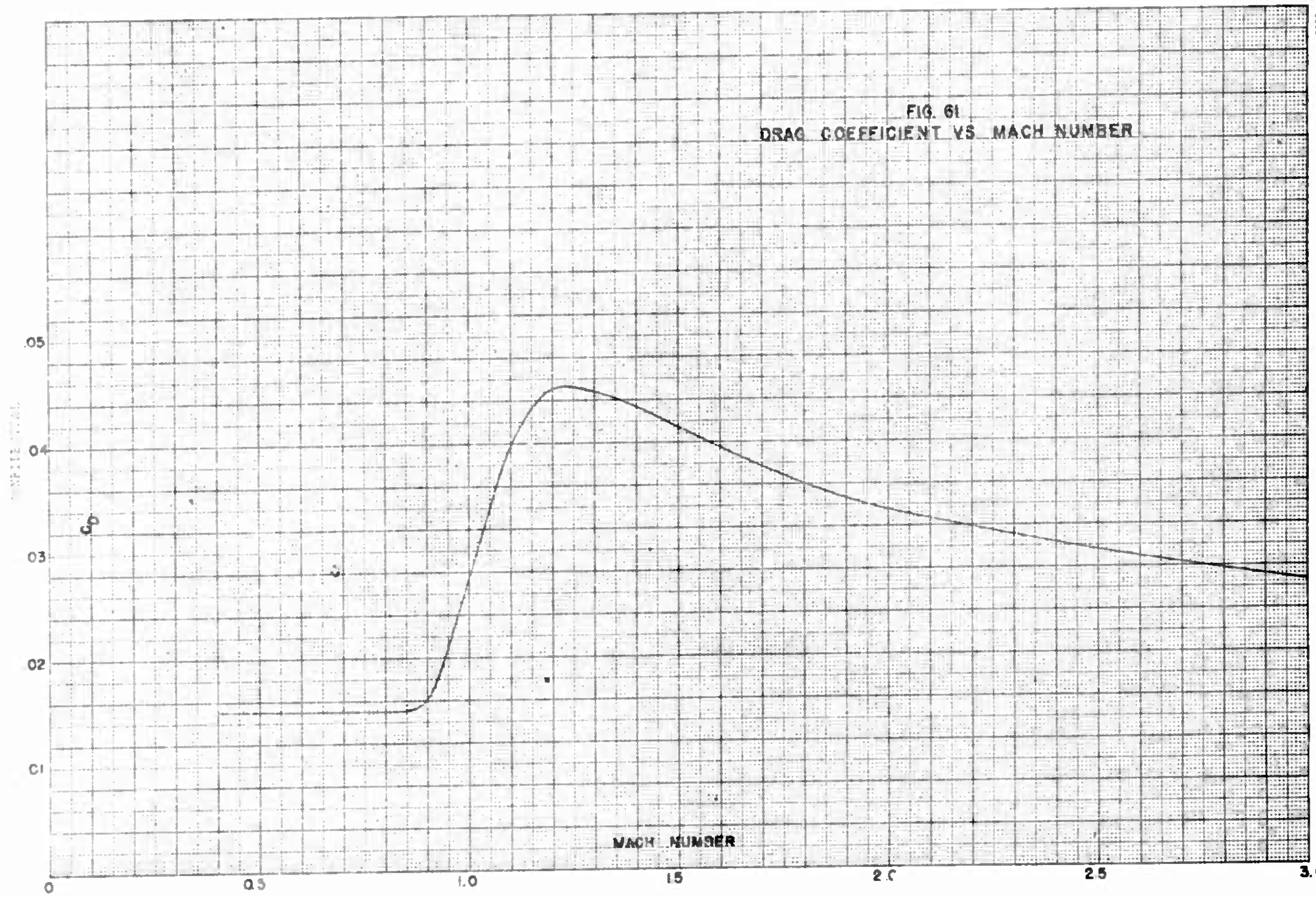

FIG. 61
DRAG COEFFICIENT VS MACH NUMBER
.05
.04
.03
.02
.01
C_D
0
0.5
1.0
1.5
2.0
2.5
3.0
MACH NUMBER
15900

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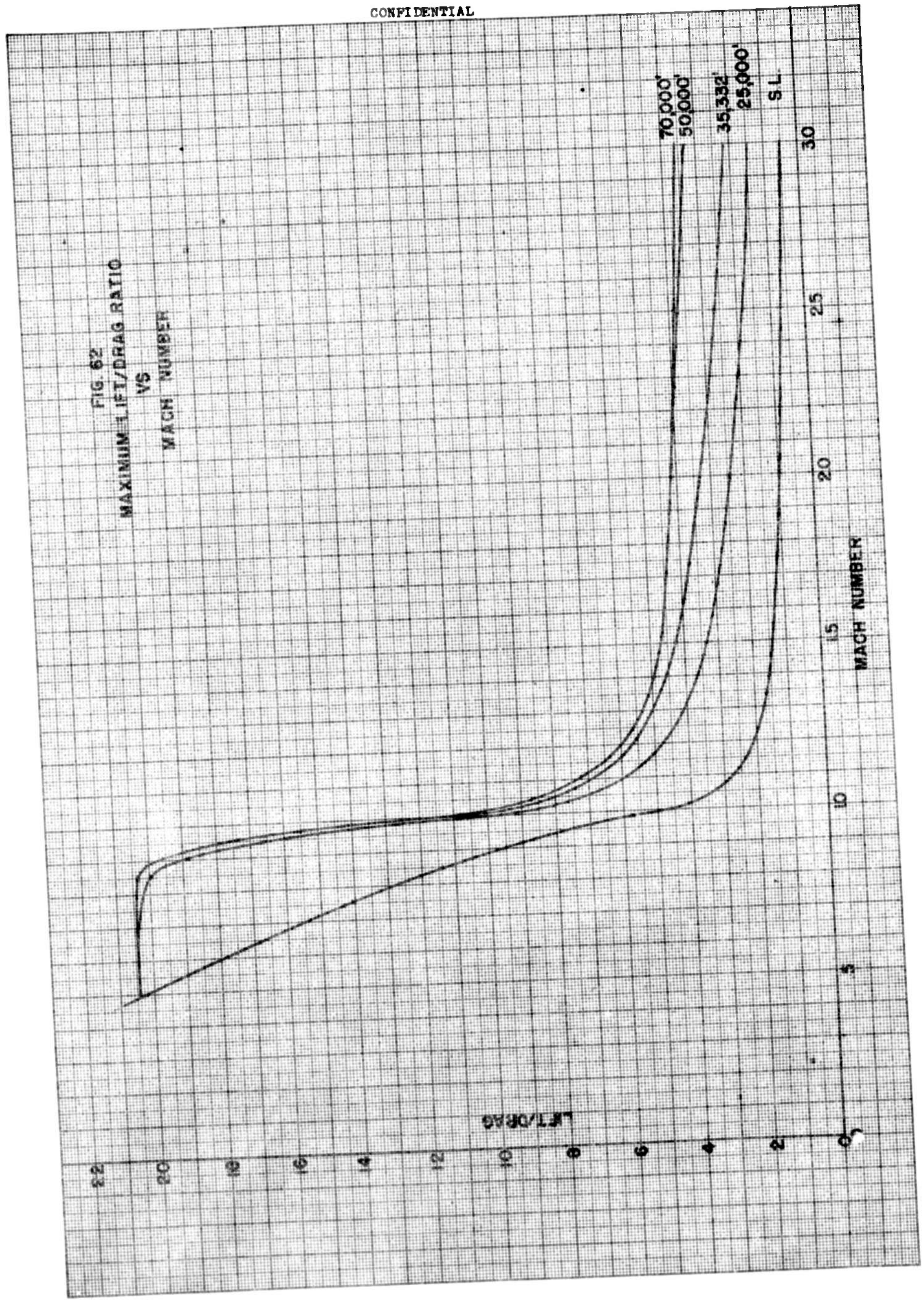
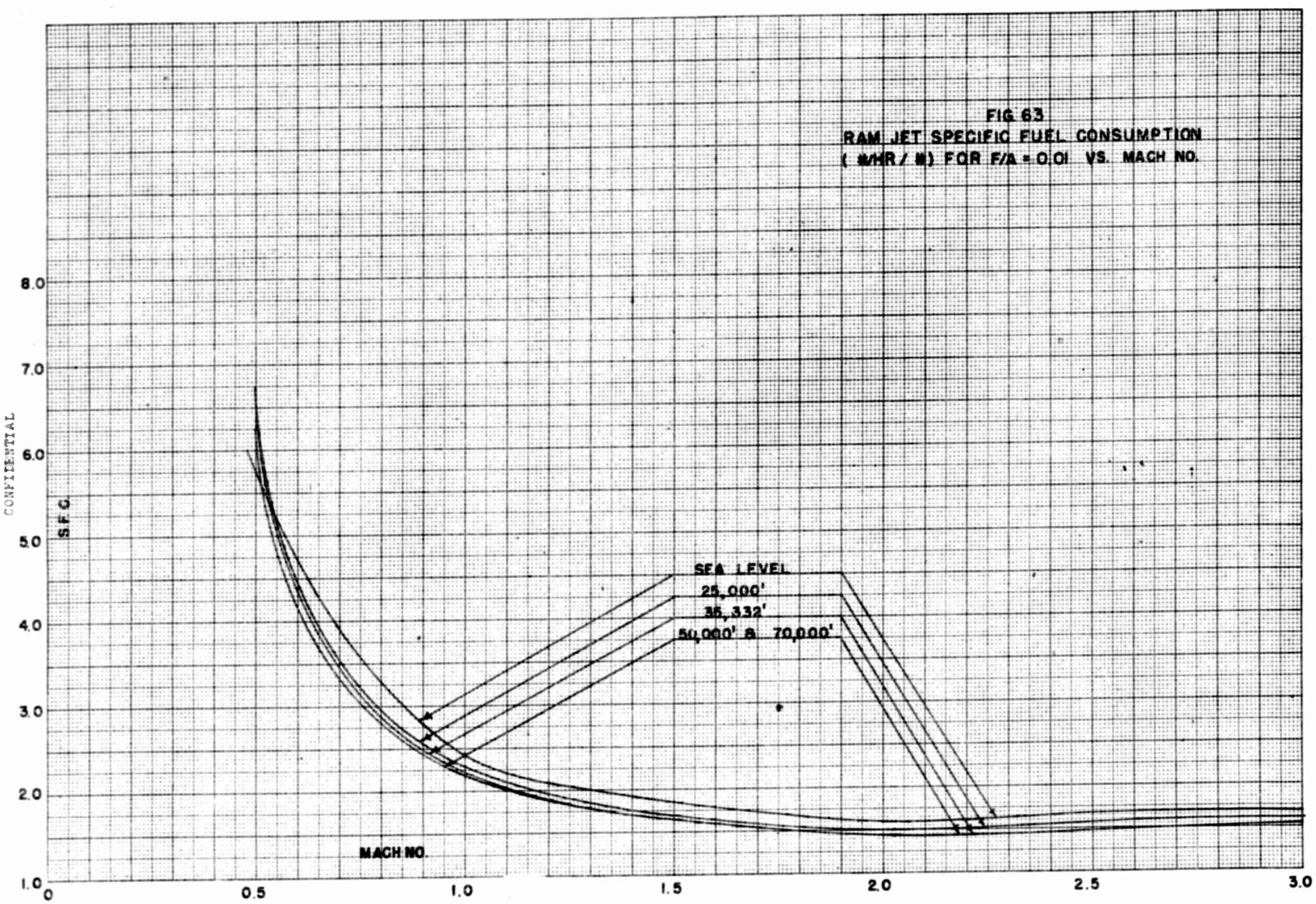


FIG. 62
MAXIMUM LIFT/DRAG RATIO
VS
MACH NUMBER



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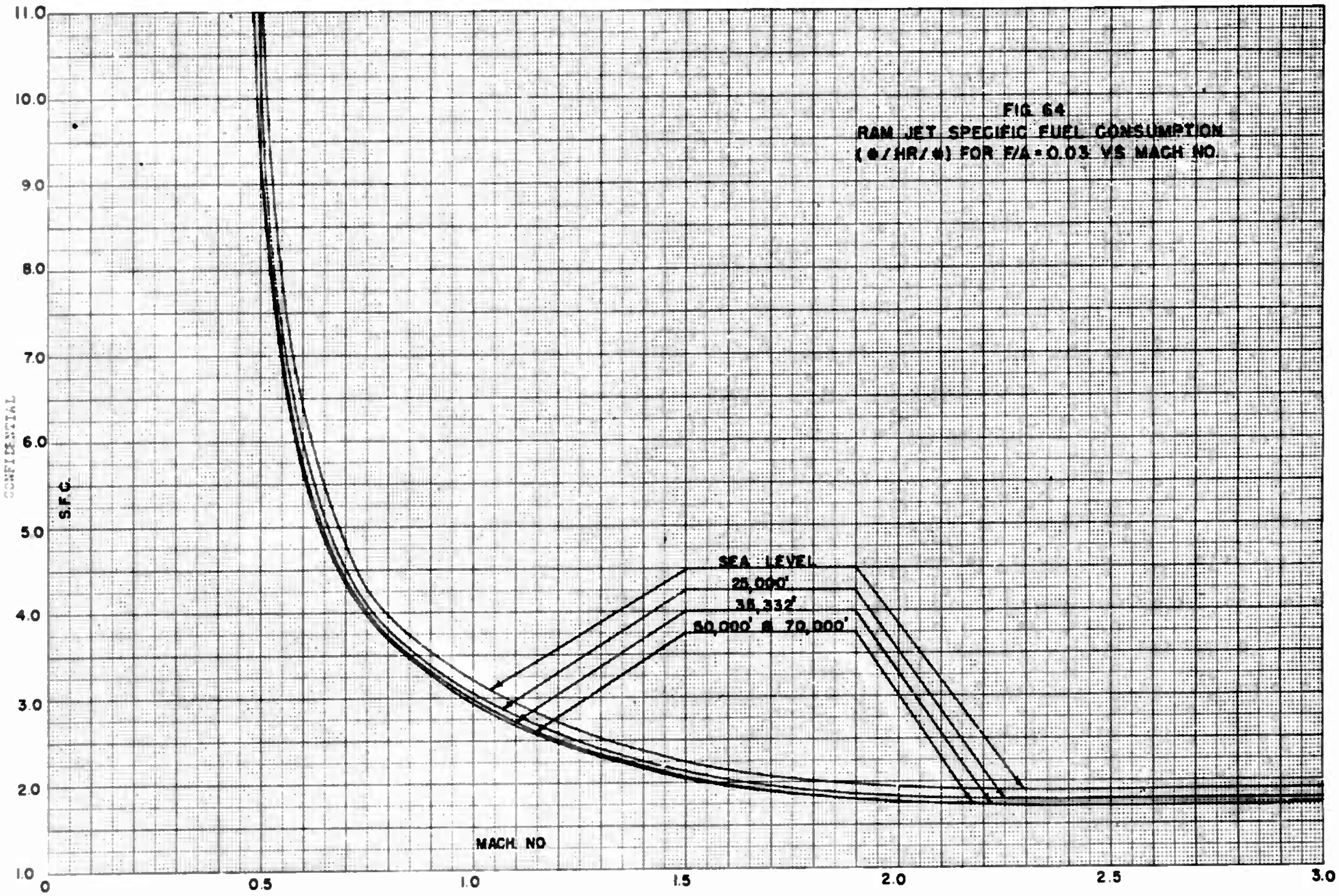

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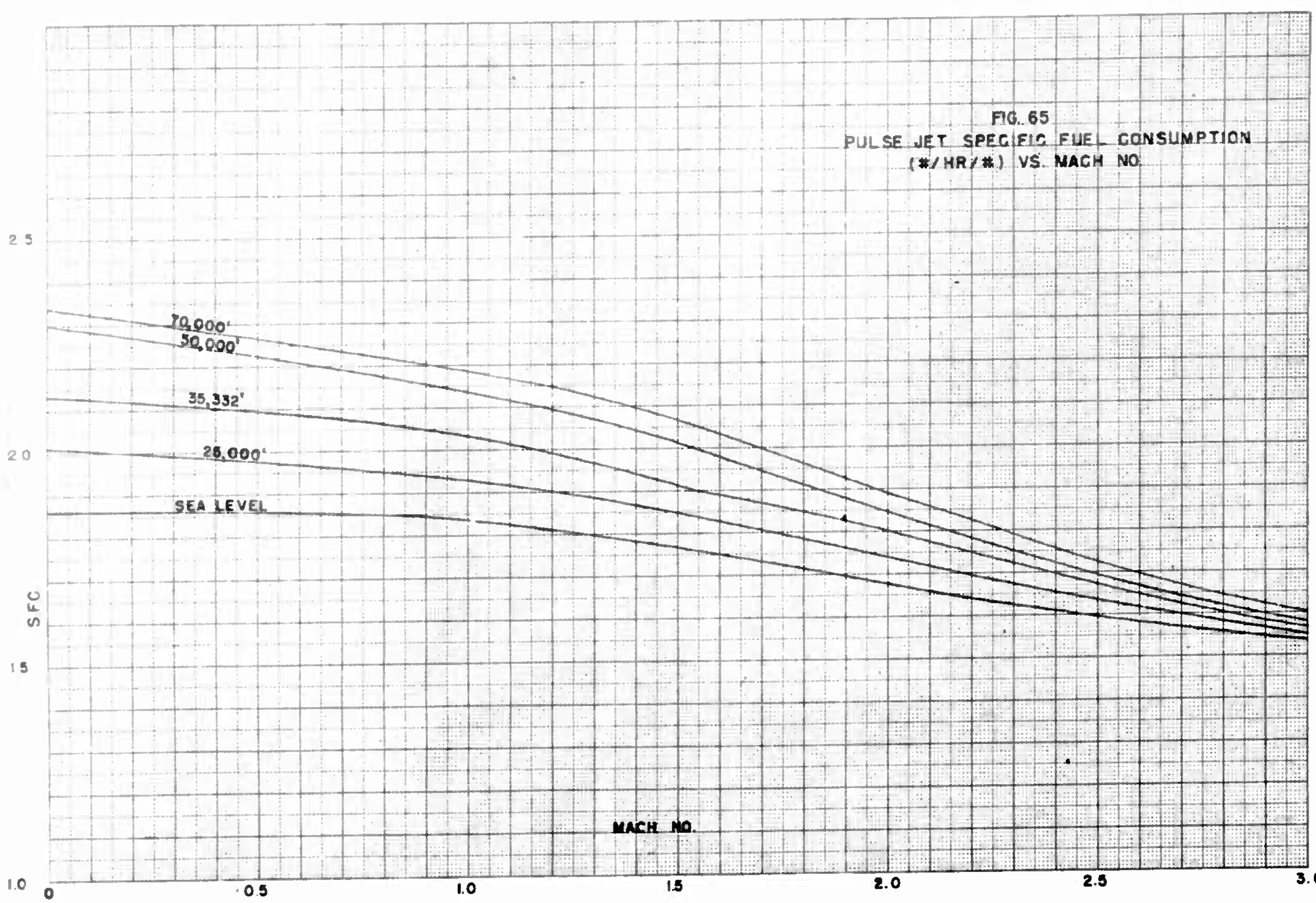
FIG. 65
PULSE JET SPECIFIC FUEL CONSUMPTION
(#/HR/#) VS. MACH NO.
SFC
1.0
1.5
2.0
2.5
MACH NO.
0
0.5
1.0
1.5
2.0
2.5
3.0
70,000'
50,000'
35,332'
25,000'
SEA LEVEL

REEL – C
1707
A.T.I.
9996

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TITLE: An Estimation of the Operational Limits of Pilotless Aircraft Using Various Jet Engines

AUTHOR(S): Perlman, Eliah P.; Zirkind, Ralph; Lancaster, O. E.

ORIGINATING AGENCY: Bureau of Aeronautics, Design Research Div., Washington, D.C.

PUBLISHED BY: (Same)

ATI- 9996

DIVISION: 8 Nov '48

ORIG. AGENCY NO.: P-1030

PUBLISHING AGENCY NO.: (Same)

DATE	DOC. CLASS.	COUNTRY	LANGUAGE	PAGES	ILLUSTRATIONS
Nov '47	Confd'l	U.S.	Eng.	94	tables, diagrs, graphs

ABSTRACT:

Comparison is made of the turbo-jet, turbo-jet with afterburning, ramjet, pulse-jet, and liquid rocket engines as applied to the special problem of propelling a missile in level flight at a constant speed. The percentage of the initial gross weight required for the engine, the airframe structure, the fuel, and the tanks were each determined for a series of Mach numbers from 0.5 to 3, range from 0 to 4000 nautical miles, and altitude from 0 to 70,000 ft. The resulting graphs were used to obtain upper bounds of attainment for the missile with various percentages of payload for each of the engines in order to determine the best engine for a given region. The turbo-jet predominates in the subsonic region with the ramjet and pulse-jet close competitors in the supersonic region. Airflow engines with a .01 fuel/air ratio show better performance than those having a ratio of .03.

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DIVISION: Guided Missiles (1)

SECTION: Performance (11)

ATI SHEET NO.: C-1-11-10 (Old C-1-3-5)

SUBJECT HEADINGS: Missiles, Guided - Operation (83137)