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THE GENERALIZED APPROACH TO THE SELECTION OF
PROPULSION SYSTEMS FOR AIRCRAFT

L. R. Woodworth and C. C. Kelber

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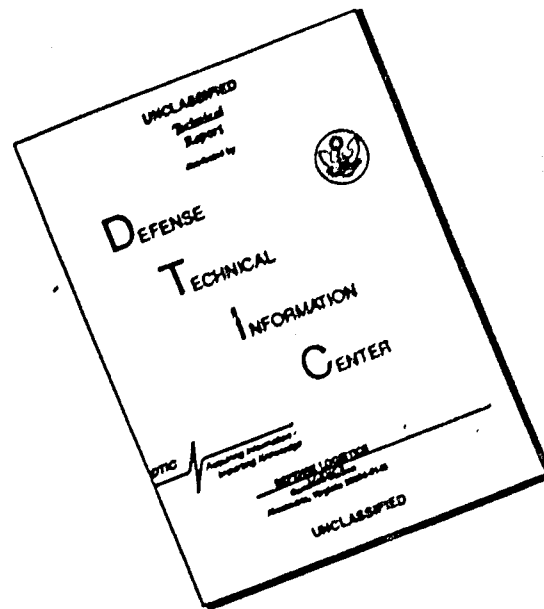
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GENERALIZED APPROACH TO THE SELECTION OF
PROPULSION SYSTEMS FOR AIRCRAFT

ABSTRACT

The generalized approach considers the powerplant in terms of the performance and application requirements of the aircraft. A powerplant choice for a given combination of these requirements represents a compromise between powerplant weight, frontal area, and fuel consumption.

Parameters relating to the performance and application requirements of aircraft are used in simplified endurance equations, which encompass a broad range of applications, in combination with powerplant characteristics. By use of electronic calculating equipment, systematic evaluation of each powerplant type and variation within type is established for each selected combination of aircraft performance and application requirements. The powerplant is selected on the basis of maximum endurance which, under the method used, is equivalent to minimum total weight of powerplant plus fuel weight.

The powerplant spectrum is represented by seven powerplant types; Rocket, Ram Jet, Afterburning Turbojet, Turbojet, Ducted Fan, Turboprop with varying power division between propeller and jet, and Reciprocating. Effects of compressor pressure ratio and turbine inlet temperature on the characteristics of the gas turbine family are considered.

LIST OF SYMBOLS

- A - area, powerplant maximum frontal, square feet
- C - net propelling specific fuel consumption $\frac{W_f}{F_n - D_p}$
- D - drag, pounds
- E_p - evaluation parameter
- F - net propelling thrust, $F_n - D_p$, pounds
- F_n - net powerplant thrust, pounds
- h - altitude
- K - ratio of weight of tanks and plumbing to fuel weight
- L - lift, pounds
- M - Mach number
- N - revolutions per minute
- n - maneuvering aerodynamic load factor $\frac{L_o}{W_o}$
- P_r - pressure ratio, compressor
- P_{T2} - compressor inlet total pressure, #/ft²
- P_{T3} - compressor outlet total pressure, #/ft²
- R - reserve fuel, percent of total
- SHP - shaft horsepower
- SFC - specific fuel consumption
- t - portion of flight time, hours
- T - total flight time, hours
- V - velocity, miles per hour

- W_a - air flow rate, lbs/second
 W_p - powerplant system weight, pounds
 W_f - fuel weight, pounds
- β - parameter
 d - ratio of ambient to sea level standard pressures
 η - efficiency
 θ - ratio of ambient to sea level standard Rankin temperatures
 λ - ratio of cruise time to total flight time
 μ - ratio of propulsion system to initial gross weights

Subscripts

- a - augmentor
f - fuel
c - climb
n - net
p - powerplant
r - cruise
t - total
o - maximum speed
1 - initial
2 - compressor inlet
3 - compressor outlet

The generalized approach as presented here is intended to examine the aircraft operational and design requirements affecting the selection of a powerplant and to determine the regions in which some of the requirements are important and require careful consideration and the regions in which some are insensitive and may be ignored. The absolute level of powerplant comparisons presented should be studied with the thought that the powerplant information used here is necessarily from unclassified sources and that airframe design and operation possibilities are still required to provide values for the airframe parameters described herein.

The primary purpose of any aircraft powerplant is to achieve a propelling force on the air frame connected to it. It would be most desirable to achieve this propelling force with no expenditure of fuel, a weightless engine, with no volume and zero frontal area. We cannot realize these conditions so we must accept penalties of powerplant weight, size and fuel consumption as necessary to achieve a propelling thrust. However, with the large variation of powerplant types that exist today and the numerous variations within each type, we do have a choice of selecting a powerplant that will minimize the magnitude of these penalties for any chosen aircraft application.

As a general rule the selection of most aircraft powerplants can be considered as representing a compromise between fuel consumption, weight and size or frontal area. The weight and fuel consumption unfortunately are generally contradictory in nature. Engineering methods of accomplishing a decrease in one usually lead to an increase in the other. This is particularly true of the gas turbine type of powerplant. A look at the propulsion spectrum with lightweight, high fuel consumption rocket

powerplants at one extreme and heavy, very low fuel consumption nuclear powerplants at the other shows the contradictory nature of fuel consumption and weight.

Any powerplant selection method must arrive at a suitable compromise of the powerplant's characteristics of thrust, fuel consumption, weight and size or frontal area. The basis for establishing this compromise is primarily dependent upon performance and design requirements of the aircraft. Once these requirements have been established then various powerplants can be studied in conjunction with these requirements and evaluated. The choice of proper powerplant evaluation factors is a difficult one; they may be factors which deal the design possibilities of the entire powerplant spectrum or the choice may be based on such factors as powerplant availability, cost or maintenance.

We will assume here that the choice of powerplant evaluation factors is dependent upon the design possibilities of the propulsion spectrum.

In a generalized approach to the selection of the powerplant we should express the operational and design requirements of the aircraft in as nearly a generalized fashion as possible. This is done by considering the aircraft operation to consist of three regimes as follows:

1. Climbing flight.
2. Cruising flight at cruising altitude.
3. Maneuvering high speed flight at cruising altitude at maximum power.

The criteria for powerplant selection shall be maximum endurance considering all flight regimes over which the powerplant must operate. Under the methods developed this is equivalent to selecting a powerplant on the

basis of minimum propulsion weight (powerplant plus fuel plus associated tankage and plumbing).

The powerplant characteristics most affecting the performance of the aircraft and therefore of primary importance in the problem of engine selection are defined as follows:

- | | | | |
|---|-------------------|---------------------------|--------------------|
| ① | $\frac{W_p}{F_n}$ | specific engine weight | $\frac{lb}{lb}$ |
| ② | $\frac{W_f}{F_n}$ | specific fuel consumption | $\frac{lb/hr}{lb}$ |
| ③ | $\frac{A}{F_n}$ | specific frontal area | $\frac{ft^2}{lb}$ |

In general the emphasis to be attached to the individual engine characteristics varies with airframe design. For instance, ① is of major importance for applications of short duration, ② for applications of long duration, and ③ for high speed applications.

The relative weight to be assigned to each engine characteristic in determining the best propulsive system for a given application is determined by the use of an evaluation equation which leads to the choice of propulsion system yielding maximum endurance when the airframe design parameters are specified. By systematic variation of these airframe design parameters many applications may be defined. The optimum propulsion system is the one whose characteristics maximize the evaluation expression to follow for any given set of airframe parameters.

Powerplant size or frontal area characteristics are combined with those of specific weight and specific fuel consumption by defining the latter characteristics on the basis of powerplant thrust minus powerplant drag:

$$\text{Net propelling specific weight} = \frac{W_p}{F_n - D_p} = \frac{W_p}{F}$$

$$\text{Net propelling specific fuel consumption} = \frac{W_f}{F_n - D_p} = C$$

The values of drag penalty associated with each powerplant are calculated with the aid of Figure 39, Appendix I. The powerplants are considered as nacelle installations in all cases except for rocket types. This assumption entails little loss in generality and greatly simplifies the problem of consideration of powerplant size or frontal area effects upon powerplant selection. Internally housed powerplants require individual consideration of co-arrangement with fuel, payload, ducting, crew quarters and allied gear and such housing is difficult of generalization in assessing size or drag penalty. Rocket installations are assumed internally housed and charged with no drag penalty.

Assuming for the most general case a composite propulsion system consisting of one type employed for cruising and another type for thrust augmentation during periods of maximum power, the fuel weight is given by

$$W_f = \frac{\mu W_1 - (W_{p_0} + W_{p_a})}{1+K} \quad (1)$$

where

W_f = fuel weight, pounds

μ = ratio of propulsion system weight (fuel, tanks, plumbing and powerplant) to initial gross weight, dimensionless

- W_1 = initial gross weight (take-off), pounds
- W_{P_o} = weight of primary (cruise) powerplant, pounds
- W_{P_a} = weight of augmenting powerplant, pounds
- K = ratio of tanks and plumbing weight to fuel weight, dimensionless

The fuel available for flight is considered as the total fuel load minus the fuel held in reserve and is approximated by the following equation:

$$(1-R) W_f = \frac{C_\ell}{V_\ell} h_o W_1 + t_o (C_o F_o + C_a F_a) + t_r C_r F_r \quad (2)$$

where

- R = percent fuel held in reserve
- C = average specific fuel consumption, $\frac{\text{lbs/hr}}{\text{lbs thrust minus drag}}$
- V_ℓ = climbing speed, miles per hour
- h_o = altitude gained during climb, miles
- W_1 = initial gross weight, pounds
- t = time, hours
- F = thrust minus nacelle drag, pounds
- subscript ℓ = climb
 - o = primary powerplant at maximum power
 - r = cruise condition of primary powerplant
 - a = augmentor powerplant

Equation 2 assumes that cruise flight begins at altitude instantly upon take-off with a sacrifice of fuel to achieve altitude. The real climb is

replaced by an imaginary vertical and instantaneous climb plus a period of cruise which approximates the time, distance and fuel used in the actual case in which best climb speed is employed. Figure 1 illustrates this concept of fuel used and distance flown during the climb portion of flight. The time and distance covered during an actual climb is credited to the cruise portion of flight in this approximation.

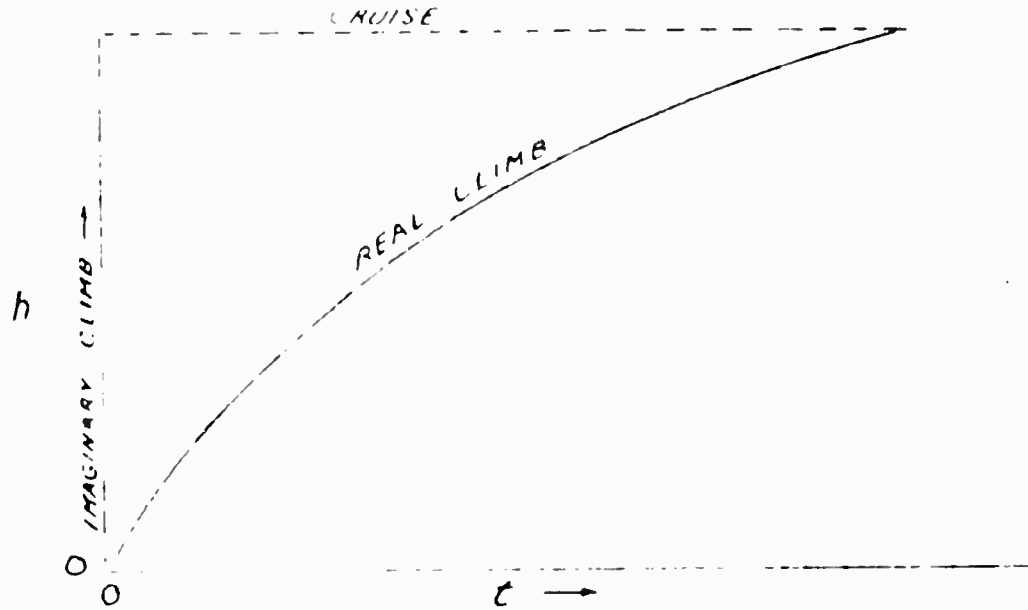


FIGURE 1

Substitution of W_f as defined in Equation 1 into Equation 2 yields:

$$\frac{1-R}{1+K} \left[\mu - \frac{W_{P_0} + W_{P_a}}{W_1} \right] - \frac{C_l}{V_\ell} h_0 = \frac{t_0 (C_o F_o + C_a F_a) + t_r C_r F_r}{W_1} \quad (3)$$

Multiplying Equation 3 by

$$\frac{W_1}{F_o + F_a}$$

and observing that

$$\frac{W_1}{F_o + F_a} = \frac{W_1}{W_o} \left(\frac{L}{D} \right)_o \frac{1}{n}$$

where

- W_o = gross weight at full power condition,
- $\left(\frac{L}{D} \right)_o$ = airframe (without nacelles) lift-drag ratio at full power condition, and
- $n = \frac{L_o}{W_o}$ = steady maneuvering aerodynamic load factor at maximum speed,

Equation 3 may be rearranged as follows:

$$\frac{1+K}{1-R} = \frac{\frac{\mu}{n} \left(\frac{L}{D} \right)_o \frac{W_1}{W_o} - \frac{\frac{W_{P_o}}{F_o} + \frac{W_{P_a}}{F_a} \frac{F_a}{F_o}}{1 + \frac{F_a}{F_o}} - \frac{C_l}{V_l} \frac{W_1}{W_o} \left(\frac{L}{D} \right)_o \frac{h_o}{n} \frac{1+K}{1-R}}{\frac{t_o \left(C_o + C_a \frac{F_a}{F_o} \right) + t_r C_r \frac{F_r}{F_o}}{1 + \frac{F_a}{F_o}}} \quad (4)$$

Further rearrangement of Equation 4 shows

$$\frac{1+k}{1-R} = \frac{\left[\frac{\mu}{n} \left(\frac{L}{D} \right)_o \frac{W_1}{W_o} - \frac{C_L}{V_L} \frac{W_1}{W_o} \left(\frac{L}{D} \right)_o \frac{h_o}{n} \frac{1+k}{1-R} \right] \left(1 + \frac{F_a}{F_o} \right) - \frac{W_{p_o}}{F_o} - \frac{W_{p_a}}{F_a} \frac{F_a}{F_o}}{t_o \left(C_o + C_a \frac{F_a}{F_o} \right) + t_r C_r \frac{F_r}{F_o}} \quad (5)$$

If this equation is multiplied by total time spent in flight and the ratio of cruise to total time be denoted by λ ,

$$\frac{1+k}{1-R} T = \frac{\left[\frac{\mu}{n} \left(\frac{L}{D} \right)_o \frac{W_1}{W_o} - \frac{C_L}{V_L} \frac{W_1}{W_o} \left(\frac{L}{D} \right)_o \frac{h_o}{n} \frac{1+k}{1-R} \right] \left(1 + \frac{F_a}{F_o} \right) - \frac{W_p}{F_o} - \frac{W_{p_a}}{F_a} \frac{F_a}{F_o}}{\lambda C_r \frac{F_r}{F_o} + (1-\lambda) \left(C_o + C_a \frac{F_a}{F_o} \right)} \quad (6)$$

Introduction of corrected quantities for the powerplant characteristics allows for altitude effects on propulsion system selection where δ and θ are the standard ambient pressure and temperature corrections respectively.

$$\frac{1+k}{1-R} \delta \sqrt{\theta} T = \frac{\left[\frac{\mu}{n} \left(\frac{L}{D} \right)_o \frac{W_1}{W_o} \delta_o - \frac{C_L}{V_L} \frac{W_1}{W_o} \left(\frac{L}{D} \right)_o \frac{h_o}{n} \frac{1+k}{1-R} \delta_o \right] \left(1 + \frac{F_a}{F_o} \right) - \frac{W_p}{F_o} \delta_o - \frac{W_{p_a}}{F_a} \delta_o \frac{F_a}{F_o}}{\lambda \frac{C_r}{\sqrt{\theta}} \frac{F_r}{F_o} + (1-\lambda) \left[\frac{C_o}{\sqrt{\theta}} + \frac{C_a}{\sqrt{\theta}} \frac{F_a}{F_o} \right]} \quad (7)$$

The term $\frac{1+k}{1-R} \delta \sqrt{\theta} T$ we shall term the evaluation parameter E_p .

Airframe design and performance factors entering the powerplant evaluation expression are:

- μ , propulsion weight to gross weight ratio
- n , high speed maneuvering load factor

$\left(\frac{L}{D}\right)_O$, high speed lift-drag ratio

$\left(\frac{L}{D}\right)_T$, cruising lift-drag ratio

h , design operating altitude

K , ratio of tanks and plumbing to fuel weight

R , percent reserve fuel requirement

λ , ratio of cruise time to total flight time

M_T , cruise Mach number

M_O , maximum flight Mach number

$\frac{W_1}{W_O}$, weight ratio, take-off to high speed condition

The engine characteristics entering the evaluation expression are net propelling specific weight at the high speed condition and the values of specific fuel consumption at cruise and at high speed condition. These are mainly functions of engine type.

Mathematically, the expression used to evaluate powerplants would be exact only if thrust and specific fuel consumption remained fixed during each separate portion of flight (climb, cruise and maximum speed). Since, for flight at constant L/D , the thrust must continuously decrease, it is apparent that the adopted expression is an approximation. However, the same powerplant characteristics which maximize the approximate expression to all practical purposes also maximize the more complicated logarithmic endurance equation. This was established by trial and is not subject to mathematical proof. Since in a practical sense the same results would be indicated by either

method, the more simple Equation 7 was adopted, for the saving in effort in handling the number of computations involved is considerable.

For the large number and ranges of variables it was desired to investigate, approximately 26,000 separate calculations were necessary. Electronic calculating machines were utilized to make the work feasible.

For punched card method of calculation on IBM machines, Equation 7 has the form

$$E_p = \frac{\beta_1 (1 + \beta_2) - \frac{W_{P_o}}{F_o} \delta_o - \frac{W_{P_a}}{F_a} \delta_o \beta_2}{\lambda \frac{C_r}{\sqrt{\theta}} \beta_3 + (1-\lambda) \left[\frac{C_o}{\sqrt{\theta}} + \frac{C_a}{\sqrt{\theta}} \beta_2 \right]} \quad (8)$$

Where E_p is the evaluation parameter, β and λ are parameters whose ranges of values are selected to cover a wide range of applications of propulsion systems to aircraft. β_1 reflects fuel carrying ability, β_2 fixes augmentation ratio and β_3 the thrust ratio of primary engine between cruise and full power conditions. Together β_2 and β_3 reflect the range of thrust over which the propulsion system is called upon to operate during flight and λ reflects the duration of maximum thrust.

(a) β_1 depends upon aerodynamic and structural capabilities of the airframe as well as performance requirements. It is a function primarily of altitude, flight speed, aerodynamic maneuvering load factor and total propulsion weight to gross weight. This parameter may have values between 1.0 and 10.0 at sea level and between 0.10 and 2.5 at 35,000 feet. This "fuel carrying ability" parameter, which strongly influences propulsion system choice, is seen to decrease with increasing altitude, with increasing

flight speed, with increasing maneuvering load factor and with increasing ratio of gross weight at maximum speed to gross weight at take-off.

Closer examination of the parameter β_1 is justified since it will be shown that it is one of the major factors in the selection of propulsion systems and is an important link between the airframe and powerplant variables. (It is always equivalent to the weight of powerplants plus fuel plus fuel installation weight remaining after climb, divided by corrected thrust required by the airframe minus nacelles at the design maximum speed.) It will be important to keep this in mind when interpreting the rather large effect of this parameter upon the relative merits of powerplants.

(b) β_2 varies between 0 and 2.0. When β_2 is zero, single propulsion systems result. This is the case in the majority of calculations.

(c) β_3 depends upon speed at cruise and speed at maximum power, upon the change in L/D with flight speed, upon steady-state maneuvering load factor and upon the capabilities of the engine type under consideration. All values of thrust ratios are covered by considering only maximum power at maximum speeds. The reciprocal of β_3 will be called the thrust ratio.

(d) The following table shows the speeds considered for each engine type:

<u>Engine Type</u>	<u>Sea Level Mach Numbers</u>					<u>Altitude Mach Numbers</u>					
Reciprocating	.25		.75			.25	.50	.75			
Turboprop	.25	.50	.75	1.0		.25	.50	.75	1.0		
Intermediate Turboprop		.50	.75	1.0		.50	.75	1.0			
Turbojet		.50	.75	1.0		.50	.75	1.0			
Ducted Fan							.75	1.0			
Afterburning Turbojet		.5	.75	1.0	1.5	2.0	.5	.75	1.0	1.5	2.0
Afterburning Turbojet with Ram Jet Augmentation									1.0	1.5	2.0
Afterburning Turbojet with Rocket Augmentation									1.0	1.5	2.0

(e) Cruising flight and high speed flight are performed at the same altitude.

(f) Cruise speeds are equal to or less than the maximum speeds in all cases but never greater than Mach 1.0.

(g) Cruise-to-total-time ratios vary between zero and unity in the following steps:

0 .6 .8 .9 .95 1.0

(h) Altitudes considered are sea level and 35,332 feet.

(i) The minimum power output of any powerplant considered is forty percent of maximum.

The simple case of finding the best propulsion system when the entire flight is at cruise speed ($\lambda = 1.0$) and cruise power ($F_o/F_r = 1.0$) for the single engine type application (no augmentation) may be illustrated graphically.

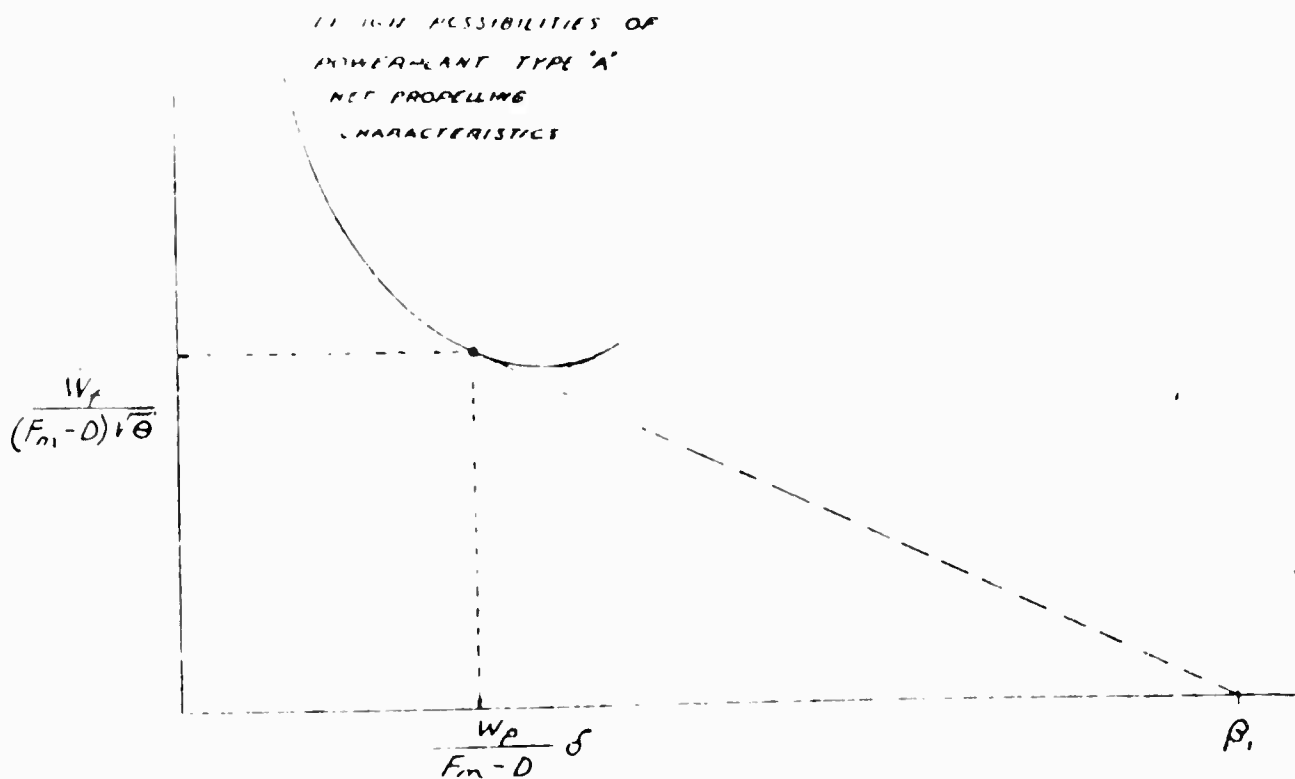


FIGURE 2.

A tangent to the curve of powerplant characteristics from a point on the abscissa equal to the value of β_1 locates the best engine and method of operation with type "A" since it maximizes the ratio

$$\frac{\beta_1 - \frac{W_P}{F_n - D} \delta}{\frac{W_f}{\sqrt{\theta} (F_n - D)}} = \frac{\text{fuel load}}{\text{fuel flow rate}} \quad (9)$$

The powerplant characteristics chosen for use in the evaluation equation were plotted on curves with net propelling specific fuel consumption as the ordinate and net propelling specific weight as the abscissae.

In cases where the gas turbine powerplant operates in more than one regime of flight it is assumed that the compressor pressure ratio is constant, that the corrected airflow at the compressor inlet is linear with corrected r.p.m. and that the actual r.p.m. of the powerplant is constant. These conditions plus a knowledge of the cycle conditions enable the computation of thrust and fuel consumption of the same powerplant at other regimes of flight to be made.

The turbojets consider compressor pressure ratios from 2 to 18 and a peak turbine inlet temperature of 1800°F. Minimum turbine inlet temperatures are as low as 800°F. The speed range considered is from a flight Mach number of .5 to 1.0 at altitudes of sea level and 35,000 feet. Figures 3, 4 and 5 illustrate typical characteristics used while Appendix I lists the methods and assumptions used to obtain the turbojet characteristics.

The afterburning turbojets also consider compressor pressure ratios from 2 to 18, 1800°F maximum turbine inlet temperature and a maximum afterburner temperature of 3200°F. The speed range is from a flight Mach number of .5 to

2.0 at altitudes of sea level and 35,000 feet. Figures 6 and 7 illustrate typical characteristics used while Appendix I lists the methods and assumptions used.

The turboprop powerplants consider compressor pressure ratios from 6 to 12 and a peak turbine inlet temperature of 1800°F. The division of power between propeller and jet is varied with the maximum portion to the propeller at the point of minimum specific fuel consumption at the flight speed considered. Figure 40 illustrates the effect of power distribution upon the turboprop characteristics.

The speed range for the turboprop powerplants is from a flight Mach number of .25 to 1.0 at altitudes of sea level and 35,000 feet. Figures 8, 9 and 10 illustrate typical characteristics of this type of engine. The methods and assumptions used in compiling these characteristics and obtaining the weight of the propeller and reduction gear are presented in Appendix I.

The characteristics assigned to reciprocating engines are:

- a. $\frac{W}{SHP}$ = 1.2 at 100 percent power
- b. $\frac{W}{A}$ = 225 $\frac{lbs}{ft^2}$
- c. SFC = .75 $\frac{lb/hr}{SHP}$ at 100 percent power
.60 at 80 percent power
.45 at 60 percent power
.60 at 40 percent power
- d. supercharged to 35,332 feet.

These assumptions are simple but thought reasonable and adequate for the purpose. Figure 11 illustrates the characteristics of the reciprocating powerplant.

For the rocket powerplant, weight per unit thrust is assumed to be .03 and specific fuel consumption 18 pounds per hour per pound of thrust. No drag penalty for the rocket powerplant has been considered.

Ducted fan powerplant characteristics were obtained for overall compressor ratios of 6, 10 and 16 with a constant turbine inlet temperature of 1600°F. The ratio of fan flow to turbine flow is 2:1. All ducted fan powerplants are considered to afterburn the fan air at the maximum power condition. Empirical relations based on dimensional analysis fix the weights of components not common to the turbojet engine. The speed range covered is from a flight Mach number of .75 to 1.0 at 35,000 feet altitude.

The characteristics of ramjet powerplants were established by cycle calculation using assumed component efficiencies and a combustion chamber outlet temperature of 3540°F. Computations were made for five Mach numbers from 1.0 to 3.0 at two altitudes, 0 and 35,000 feet. Appendix I lists the assumptions used in obtaining the ramjet characteristics illustrated in Figure 12.

The nacelle drag and frontal area methods and assumptions of all powerplants are also presented in Appendix I.

Figures 13 to 38 illustrate the results obtained. Figures 13 to 20 are concerned with the optimum choice of pressure ratios of the gas turbine powerplants, while Figures 21 to 38 are concerned with comparison of the optimum of each type of powerplant.

For the turboprop engine, Figures 13 and 14 show the major effects on choice of pressure ratio for this type powerplant to be β_1 and the high flight speed requirement. At values of β_1 above .5 at 35,000 feet and 5 at sea level we can say that the high flight speed requirement is the sole major effect on choice of the optimum compressor pressure ratio. As the high

flight speed requirement is increased, which means M_0 progresses from .75 to 1.0, we see that the optimum compressor ratio drops. This is understandable since the high compression ratio turboprops in this region have a more rapid increase in fuel consumption and decrease in thrust than the lower compression ratios as either flight speed is increased or turbine inlet temperature is decreased. This will be more pronounced at sea level than at high altitudes which accounts for higher optimum pressure ratios at altitude. It is significant to note that in the choice of optimum compression ratios for turboprop engines the value of λ , thrust ratio and cruising Mach number within the limits considered have no effect.

Figures 15 and 16 for turbojet types at sea level and 35,332 feet show that optimum pressure ratio is dependent on λ at high thrust ratios and must decrease with increasing ratios of cruise to total flight time. This is explained by the relatively better specific fuel consumption of lower pressure ratio powerplants when operated at low throttle which characteristic is emphasized by longer periods at cruise power. At high flight speed at sea level the 18:1 turbojet is inferior because of the high compressor outlet temperature which limits the amount of fuel that can be burned and increases specific fuel consumption. Turbojets at altitude show a similar trend to lower pressure ratios when required to produce a large thrust ratio and cruise power is maintained for almost the entire flight.

Figures 17 and 18 illustrate choice of P_r for the afterburning turbojet at sea level. Higher flight speeds are shown to demand lower pressure ratios. For high thrust ratios and low cruising Mach the high pressure ratio powerplant has slightly better cruise specific fuel consumption at the low power condition. Therefore, for high values of λ

and high thrust ratio, a high pressure ratio is superior. At low values of λ , somewhat lower values of pressure ratio may be superior because of lower specific weight and de-emphasis on cruise time.

The choice of optimum P_r for afterburning turbojet at altitude is dictated largely by β_1 and M_0 as shown by Figure 19. Increasing flight speed has the general effect of lowering pressure ratio as shown in previous curves. This is due to improving characteristics of the lower pressure ratio engines as flight speed is increased. Neither λ nor F_r/F_0 have much effect on the choice of optimum pressure ratio. At flight speeds up to and including Mach = 1.0 the high pressure ratio turbojets have definitely better fuel consumption characteristics both at full and throttled power. It is interesting to note that even at the highest flight speed the optimum pressure ratio does not drop much below 6:1. This is because lower pressure ratios have larger diameter combustion chambers resulting in greater frontal area and thus the increased drag offsets any potential gain from using a lower pressure ratio.

Figure 20 illustrates the choice of optimum pressure ratio for the afterburning ducted fan and reveals it to be only a function of β_1 . However, the scope of study of the ducted fan presented here is too brief to say that this is the general rule. A variation of air flow ratios of fan flow to turbine flow might have led to different conclusions.

Figures 21, 22, 23, 24 and 25 illustrate the comparison of the turbojet and turboprop at flight speeds from .75 to 1.0 Mach number and altitudes of sea level and 35,332 feet. These results show that there is no single "crossover" point at which one or the other is preferred. Instead the choice of turboprop or turbojet depends to a great extent upon the application. For example, Figure 23 which is drawn for a cruising and maximum flight Mach

number of .75, we from past experience might be inclined to favor the turboprop engine; however, at low values of β_1 where powerplant specific weight is important, we find the turbojet superior. If we consider the case where the high flight speed required is at $M_0 = 1$, as in Figure 22, we can still find regions where the turboprop is superior even when the cruising flight is also at Mach 1. Similar trends are noted at both sea level and 35,000 feet. High values of maximum Mach number, thrust ratio, λ and low values of β_1 all tend to favor the turbojet powerplant.

Figures 26, 27, 28 and 29 illustrate the comparison between turbojets and turbojets with afterburners installed. As would be expected the afterburning turbojets are superior at low values of β_1 where powerplant specific weight is of major importance. The afterburning turbojet is also superior when high values of λ and thrust ratio are required. This is explained by the fact that at these conditions the afterburner is turned off for a major portion of the flight and thus the afterburning turbojet will cruise at a higher turbine inlet temperature than the non-afterburning and therefore will have a lower installed weight of powerplant with a fuel consumption in the cruising condition that is nearly compatible with that of the non-afterburning turbojet. When high values of β_1 are used with low thrust ratios or low λ values, the tendency is to favor the non-afterburning turbojet. In studying these comparison curves the thought should be borne in mind that the high flight speed condition utilizes the maximum power output of the powerplant. If we chose to operate the afterburning turbojet at less than this value, then at the high values of β_1 the superiority of the non-afterburning turbojet would only reflect the penalties of the afterburner installation on powerplant specific weight and a slight loss in specific fuel

consumption. Comparisons have only been drawn up to a maximum flight Mach number of 1. Above this point the drag penalties effect the non-afterburning turbojet severely. At a maximum flight Mach number of 1.5 the afterburning turbojet will be superior to the non-afterburning turbojet in nearly all regions.

Figures 30 and 31 illustrate a comparison between the afterburning ducted fan and the afterburning turbojet. The ducted fan is superior at low values of β_1 where its low specific weight with the afterburner on can be utilized. Since the ducted fan also has a higher afterburning thrust augmentation ratio than the afterburning turbojet we can expect that high values of thrust ratio and low values of β_1 to favor the ducted fan as is shown. The decreasing superiority of the ducted fan at high values of β_1 can be explained by its inferior specific weight non-afterburning and also that within the range of thrust ratios considered the afterburner may be always lit. It is felt that the ducted fan characteristics may be optimistic and therefore attention is invited to the shape of the curves and not the absolute levels of comparison.

Figure 32 compares a combined propulsion system (afterburning turbojets with rocket augmentation) with the afterburning turbojet alone. Only the largest cruise to total flight time ratio is illustrated since smaller values of λ are still more to the disadvantage of the composite system. This comparison is insensitive to thrust ratio in the regions indicated but shows the interesting effect of decreasing worth of the rocket augmentation with increasing flight speed and increasing fuel carrying ability, β_1 . Higher supersonic flight speeds decrease the specific weight of the afterburning turbojet but have no effect upon the rocket specific weight. Only at low values of β_1 is the rocket-afterburning turbojet combination

superior to the afterburning turbojet alone. This curve is valid at all altitudes above 35,000 feet at which the turbojet is capable of efficient operation.

Figure 33 makes a similar comparison using ramjet augmentation. In this case the effect of high flight Mach number is reversed from the rocket case since the ramjet specific weight is decreasing more rapidly than that of the afterburning turbojet with increasing flight speed while its specific fuel consumption shows a decrease against an increase for the afterburning turbojet. Again, this composite system is superior only at low β_1 values (short duration flights) although sensibly independent of λ .

Figures 34 to 38 show relative merit of a number of propulsion systems when the entire flight is at the cruise condition. Figures 22 and 23 are for sea level, Mach .25 to .50 and .75 to 1.0 respectively. At sea level the reciprocating engine can match the turboprop only at low speed and large β_1 and cannot approach it at all at Mach .5. Also at Mach .5 the turboprop is clearly superior at every point to the turbojet and the intermediate-turboprop.

At higher cruise flight speeds the advantage of the turboprop disappears and at Mach 1.0 is superior to the turbojet and intermediate-turboprop only at values of $\beta_1 > 10.0$ as shown in Figure 35. However, the turboprop is still the best type considered at Mach .75 at values of β_1 greater than 2.5.

Going to altitude in Figures 36 to 38 it is seen that at low cruise flight speed (Mach .25) both turboprop and reciprocating types enjoy regions of superiority. In general, if β_1 is less than 1.0 the turboprop will give the better performance and if β_1 is greater than 1.0 the reciprocating engine type will be superior. Explanation of the peculiar behavior of the reciprocating-turboprop comparison curve with respect to β_1 lies in the change in optimum powerplant operating characteristics as β_1 changes. At

altitude the reciprocating engine is lighter for the same thrust (due to supercharging) than the turboprop when both are operated at 100 percent power. At minimum specific fuel consumption cruise, the reciprocating engine specific weight is much greater than that of the turboprop. This fact allows superiority for the reciprocating engine only at high values of β_1 and at extremely low values of β_1 — in the first case due to lower specific fuel consumption and in the second case due to lower specific weight when specific fuel consumption is not important (short time flights).

Figure 37 shows the turboprop at altitude, as at sea level, has no competitors at Mach .5 except at very low β_1 values when flight is entirely cruise and no maximum speed thrust ratio requirements exist.

Figure 38 shows relative worth of four gas turbine type engines at altitude at Mach numbers from .75 to 1.0.

At Mach .75 the ducted fan is best at β_1 values up to .5 above which it yields to turboprops. At Mach 1.0 the ducted fan appears to have wide application.

The method presented here affords a means for evaluating aircraft powerplants and to provide information as to the significance of various design and operational requirements. As the state of the powerplant design art progresses insertion of new powerplant characteristics into the evaluation equation will provide a more valid level of comparison.

Since the powerplants selected are optimums, it is of some interest to know the sacrifice involved when other than optimums are chosen due to non-availability, cost or similar reasons. A brief study showed that no general conclusions could be drawn; that the sensitivity of the powerplant choice within a type could vary widely. It would therefore seem necessary

to consider each individual case. However, the method presented here would still permit a quick determination of the areas of likely powerplants.

The authors extend their thanks and appreciation to Mr. L. B. Rumph, Head of the Aircraft Design Section of the Aircraft Division, who has contributed many of the ideas and methods presented here, and to Messrs. W. B. Gist, C. K. Sturdevant and M. H. Thurlo of the Propulsion Group, Aircraft Division, who have expended much of their own time and effort on this paper.

APPENDIX IMETHODS AND ASSUMPTIONS, POWERPLANT CHARACTERISTICS

The turbojets ram pressure ratio recovery is given in the following table.

<u>Mach</u>	<u>P_{T2}/P_{T0}</u>
0	1.00
.5	.97
.75	.965
1.00	.957
1.50	.933
2.00	.870

The compressor efficiency is eighty-five percent based on compressor inlet static pressure and compressor outlet total pressure, the combustion chamber is ninety-five percent efficient with a four percent total pressure loss. The turbine is ninety percent efficient, the exhaust nozzle is ninety-five percent. Cycle calculations were made for turbine inlet temperatures from 800°F to 1800°F at a flight Mach number of .5 to 1.0 at an altitude of sea level and 35,332 feet.

The weight of the turbojet per pound of corrected air at the compressor inlet is

$$\frac{W}{\frac{w_a \sqrt{\theta}}{\delta_{T2}}} = 7.30 \log e^{\frac{P_{T3}}{P_{T2}}} + 7.57 \quad (10)$$

For drag calculations the maximum frontal area is assumed to be 1.5 times the frontal area at the maximum diameter of the engine. At the compressor inlet the corrected air flow per square foot of frontal area is 28.9 lb/sec while at the combustion chamber maximum diameter the corrected

airflow per square foot of frontal area is 5.22 pounds per second. These assumptions establish the following table of nacelle frontal area to compressor inlet area:

<u>Nacelle Frontal Area</u> <u>Compressor Inlet Area</u>	P_{T3}/P_{T2} (Compressor Pressure Ratio)
4.63	2
2.62	4
1.87	6
1.50	10
1.50	12
1.50	18

The assumed variation of nacelle drag coefficient is shown in Figure 39 for a d/D of .7.

The corrected airflow at the compressor inlet is assumed to vary linearly with turbojet corrected rpm. The rpm is assumed constant under all flight conditions. This results in a variation of corrected airflow as flight speed and altitude are changed. Since the inlet area is a fixed constant for the powerplant and expressed in terms of corrected airflow, then it is necessary to fix the corrected airflow at some operating point of flight speed and altitude in order to establish the frontal area. The inlet area for turbojets was established at a flight Mach number of 1 at 35,332 feet.

The afterburning turbojet was based on the same components as the turbojets; the weight was increased fifteen percent to allow for the afterburner weight. When the afterburner is lit the turbine inlet temperature is 1800°F and the afterburner top temperature is 3200°F . The afterburner and the exhaust nozzle are both assumed ninety percent efficient.

The turboprops use the same gas turbine components as the turbojets. The inlet area is based on the corrected flow at 35,332 feet altitude, flight

Mach number of .75. The weight equation minus reduction gear and propeller is

$$\frac{W}{W_a \frac{\sqrt{\theta_{T2}}}{\Delta \zeta}} = 7.57 + 7.30 \log \frac{P_{T3}}{P_{T2}} + .0553 h_t \quad (11)$$

where h_t is the propeller driving turbine enthalpy drop, BTU per pound of air and combustion products.

The propellers are all considered to be four bladed single rotation supersonic and the weight is based on the power requirements at $M = .75$ at 35,332 feet and is computed for this point by the equation

$$W_{\text{propeller}} = .309 \text{ SHP} \quad (12)$$

where SHP = shaft horsepower.

The assumed propeller efficiency schedule versus flight Mach number is as follows:

<u>Mach</u>	<u>η</u>
0	.85
.5	.85
.75	.82
1.00	.75
1.50	.67

The reduction gear weight equation used

$$W_{\text{gear}} = \frac{100 \text{ SHP}_{\text{max}}}{N} \quad (13)$$

where

SHP_{max} = maximum shaft horsepower to be transmitted

N = propeller rpm

The value of N is estimated at the propeller design point and is

$$N = \frac{128,000}{(\text{SHP})^{1/2}} \quad (14)$$

where the SHP term is the same as that of equation (12).

The weight of the propeller driving turbine, reduction gear and propeller are all dependent upon the shaft horsepower. Therefore, it was possible to determine the effect of power division between propeller and jet upon the turboprop characteristics as illustrated by Figure 40.*

It is assumed that ramjets can be built to weigh eighty pounds per square foot of frontal area when required to operate at sea level at no greater than Mach 1.0. This value is the minimum used in determining ramjet characteristics and is scaled upward for higher flight speed requirements in direct proportion to the internal pressure encountered.

Cycle calculations were made to establish values of thrust per pound of air and fuel flow per pound of thrust. Combustor efficiency is .90, nozzle efficiency .95, diffuser efficiency a function of Mach number. Burner temperature is 3540°F, burner velocity 200 feet per second.

Figure 12 illustrates ramjet net propelling characteristics based upon drag coefficients according to Figure 39 with $d/D = 0.9$.

* Intermediate turboprops roughly take half the heat-to-work in the propeller driving turbine of the turboprop and utilize it in the jet nozzle. This leads to higher specific fuel consumption but a reduction in specific weight through a saving in weight of propeller, reduction gear, and turbine.

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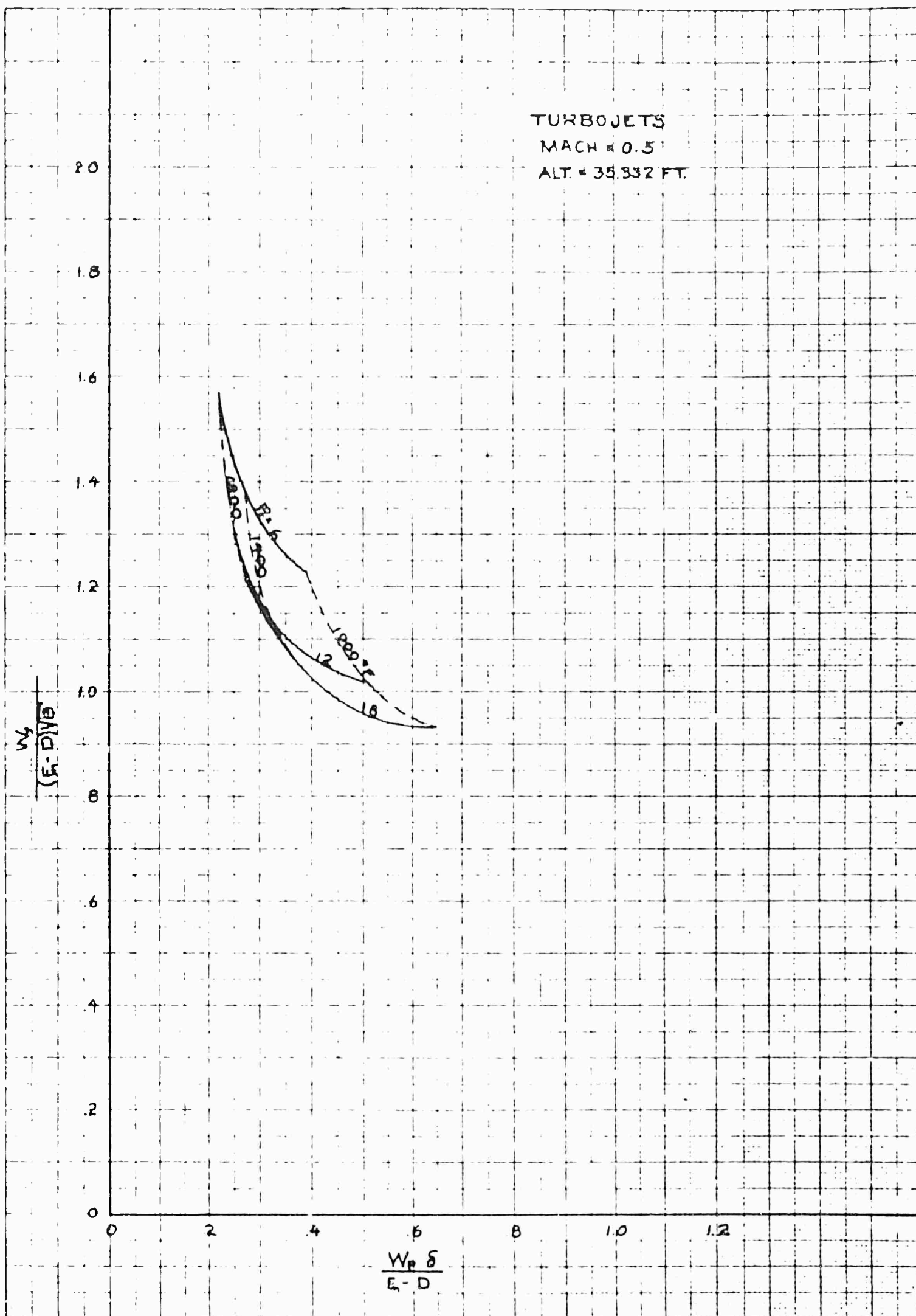


FIGURE 3

TURBOJET

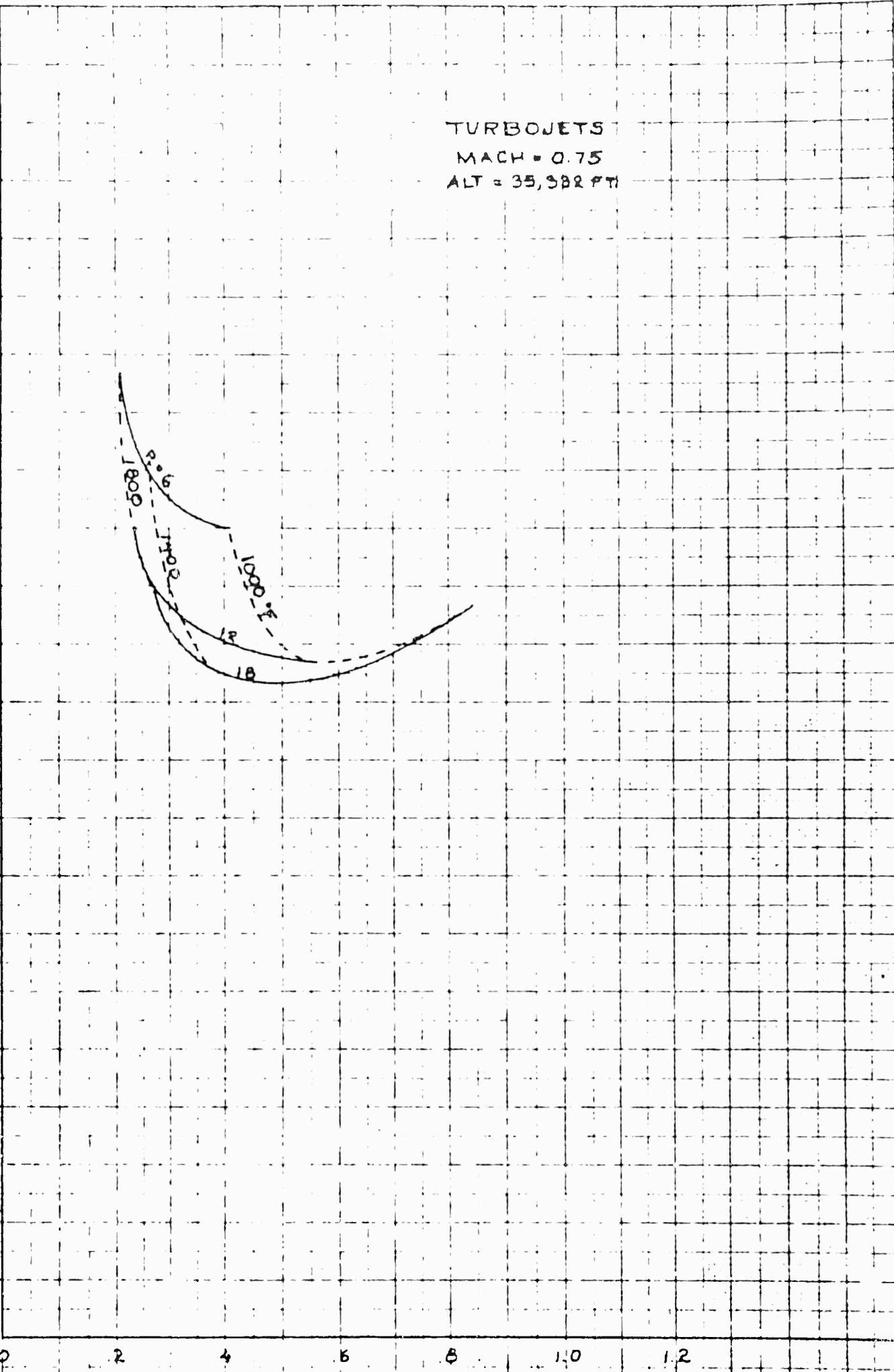
MACH = 0.75

ALT = 35,982 FT

20
18
16
14
12
10
8
6
4
2
0

$$\frac{W_f}{F_m - D}$$

$$\frac{W_p \delta}{F_m - D}$$



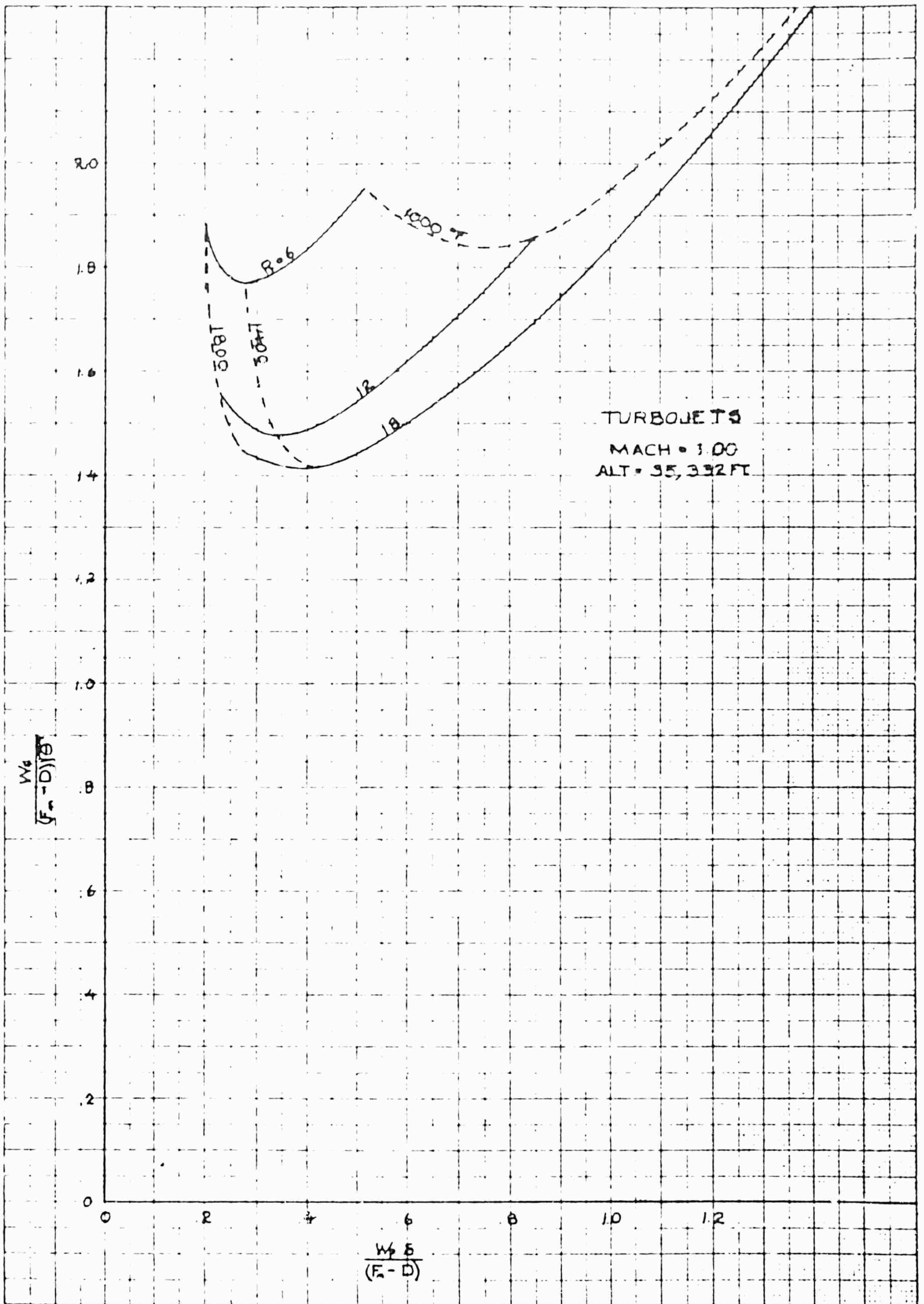
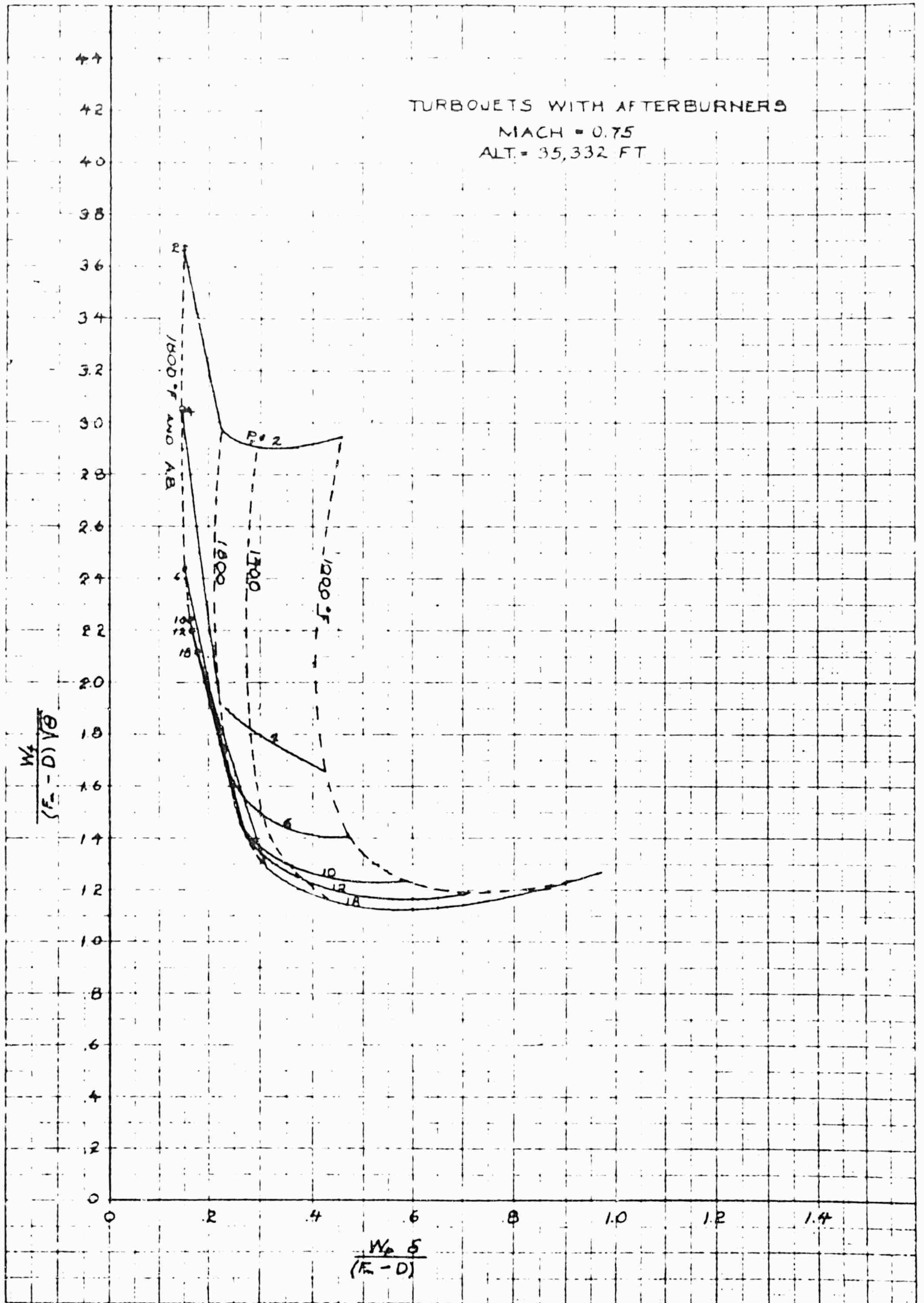
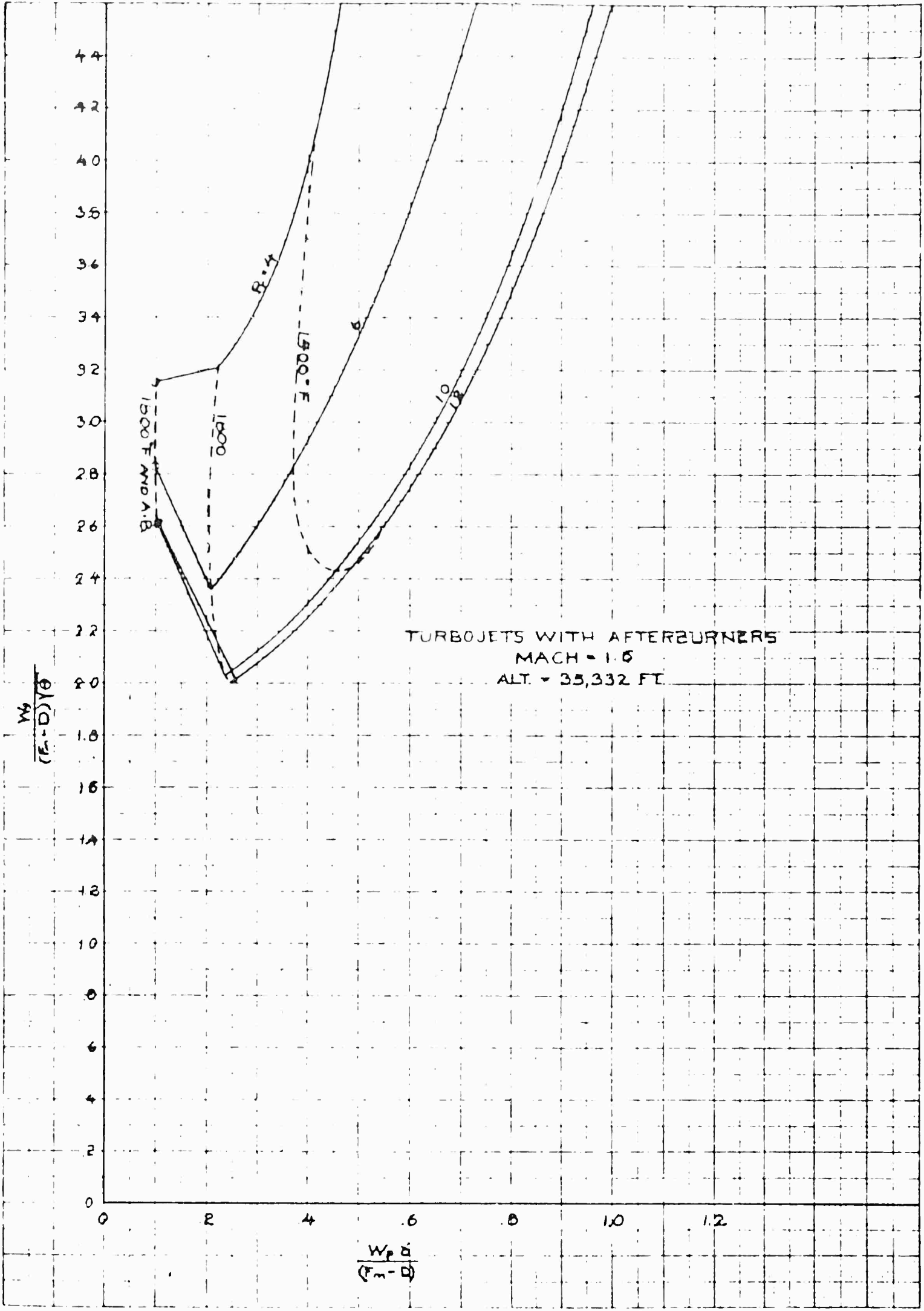


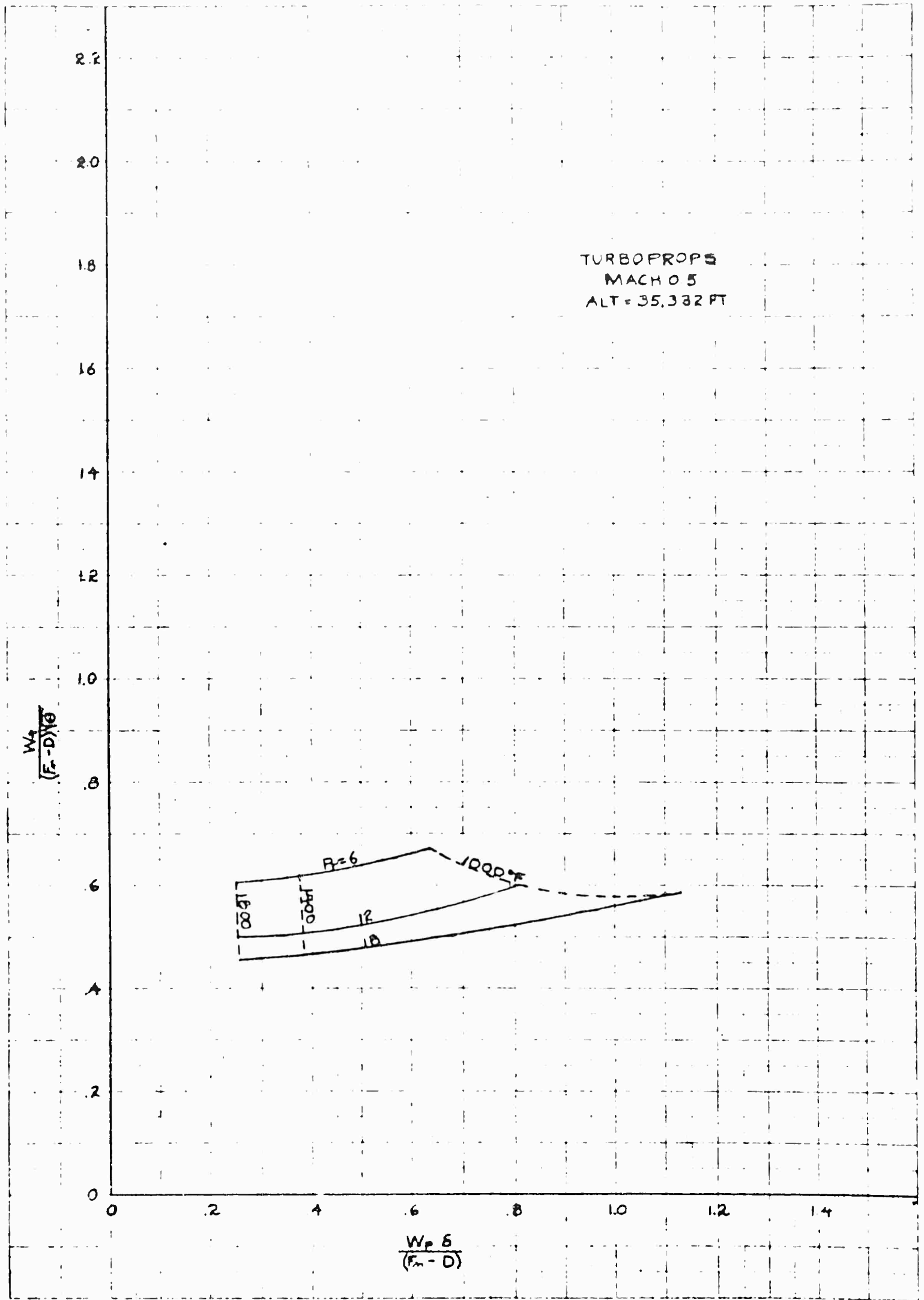
FIGURE 5

TURBOJETTS WITH AFTERBURNERS

MACH = 0.75
 ALT. = 35,332 FT





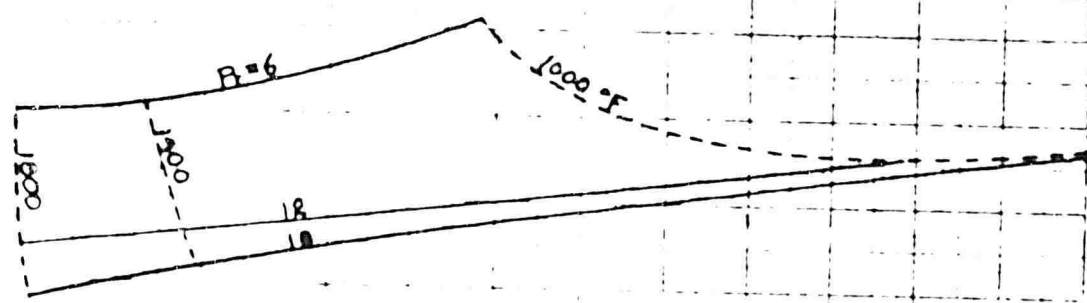


TURBOPROPS
MACH 0.75
ALT = 35,332 FT

$\frac{W_P}{F - D}$

2.2
2.0
1.8
1.6
1.4
1.2
1.0
0.8
0.6
0.4
0.2
0

$\frac{W_P}{F - D}$



TURBOPROPS
MACH 1.0
ALT = 35,832 FT.

$$\frac{W_p}{(F_m - D)} \sqrt{10}$$

2.0

1.8

1.6

1.4

1.2

1.0

.8

.6

.4

.2

0

0

.2

.4

.6

.8

1.0

1.2

1.4

$$\frac{W_p 5}{(F_m - D)}$$

1800

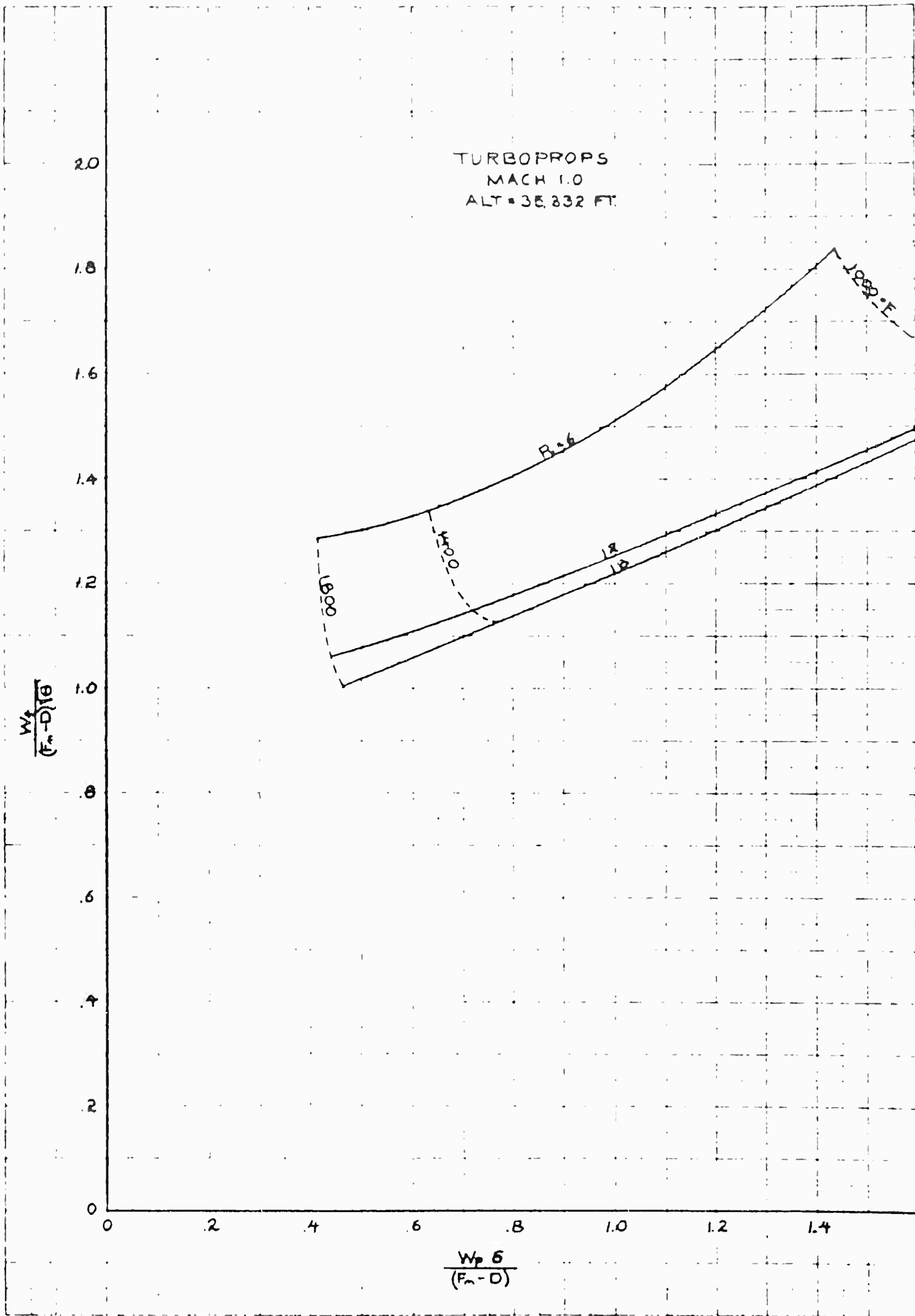
1700

B=6

12

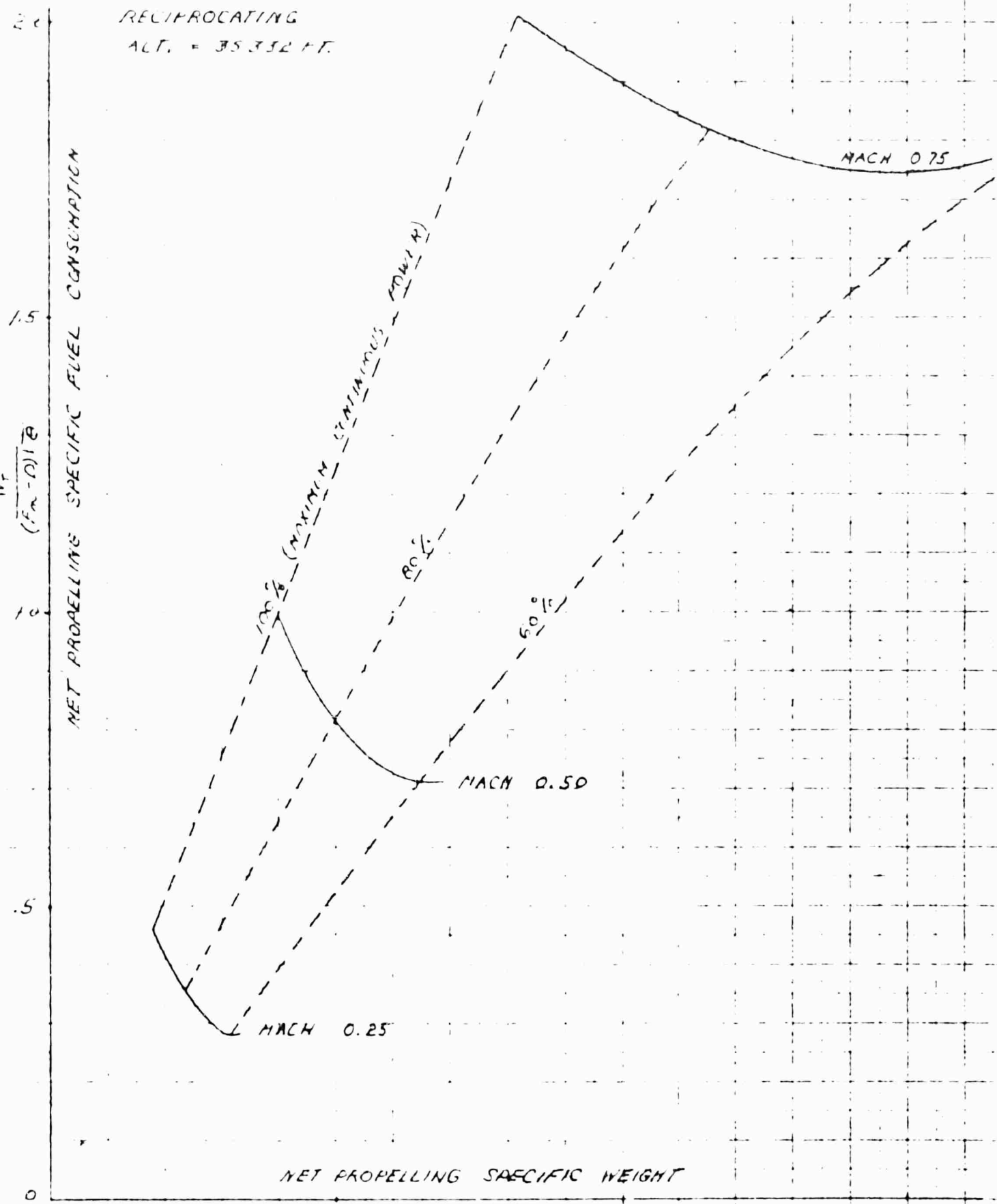
10

1000-E



SUPERCHARGED
 RECIPROCATING
 ALT. = 35,352 FT.

NET PROPELLING SPECIFIC FUEL CONSUMPTION
 $\frac{W_p}{(F_m - D) \rho}$

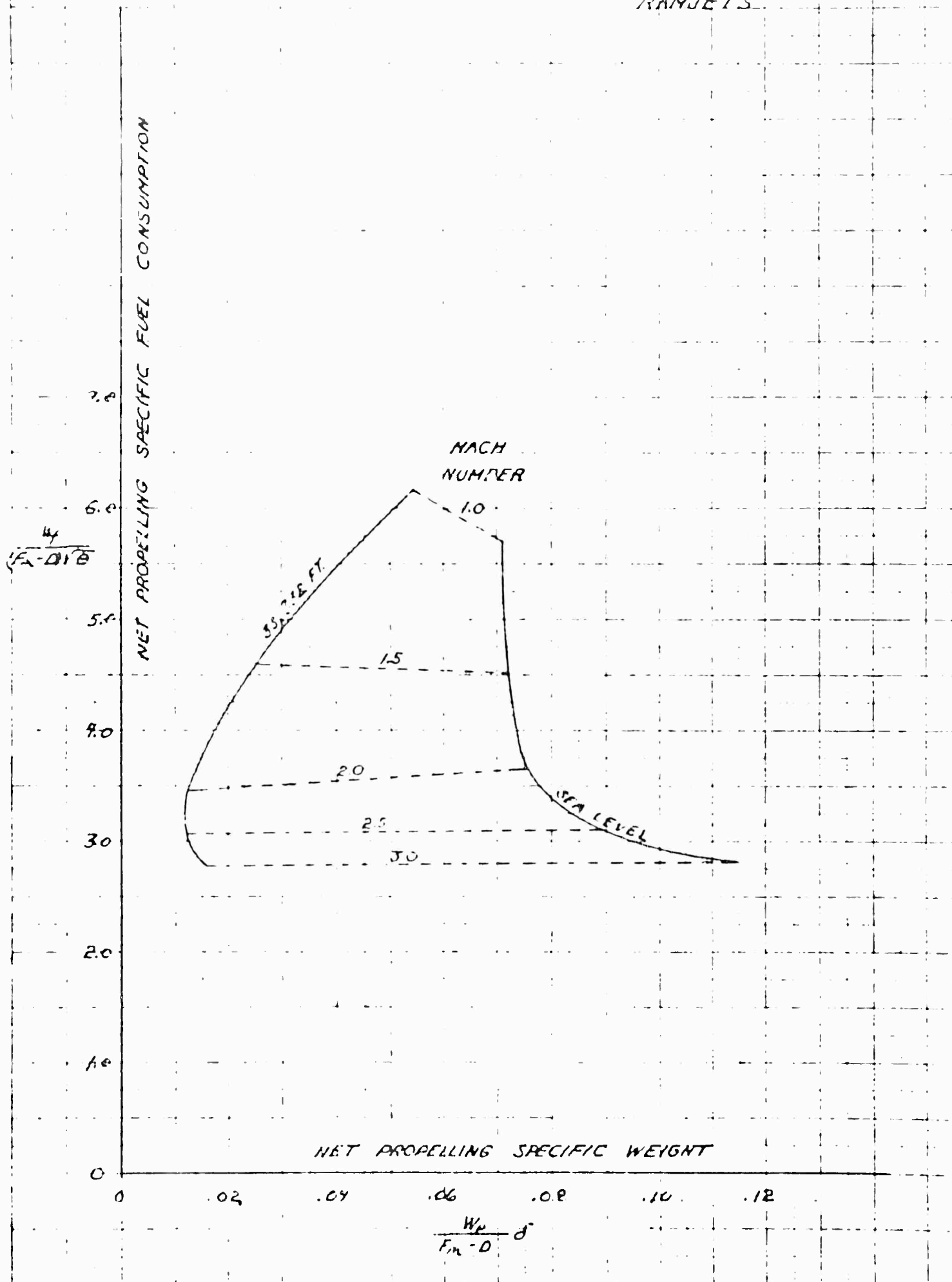


NET PROPELLING SPECIFIC WEIGHT

$$\frac{W_p}{F_m - D} \rho$$

FIGURE 11

RANJETS



TURBOPROP
SEA LEVEL

$\lambda = .20 \text{ to } .95$
 $1.0 \leq \frac{5}{F} \leq 1.667$ when $M_r \neq M_0$
 $1.0 \leq \frac{5}{F} \leq 2.5$ when $M_r = M_0$

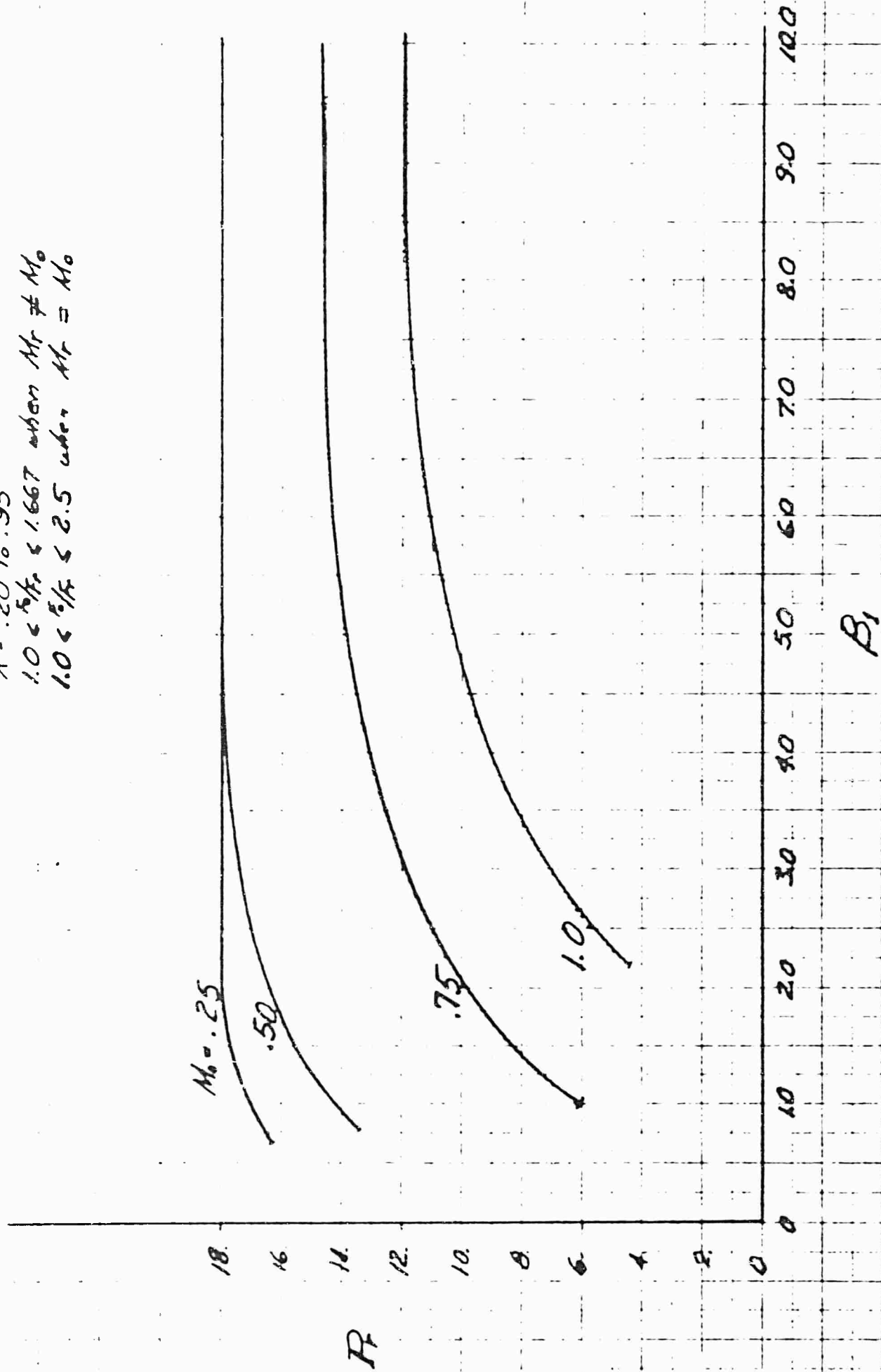


FIGURE 13

TURBOPROP
 ALTITUDE - 35,332 ft.

$\lambda = 0.20$ to 0.95

$1.0 < \frac{P}{P_0} < 1.667$ when $M_0 \neq M_0$

$1.0 < \frac{P}{P_0} < 2.5$ when $M_0 = M_0$

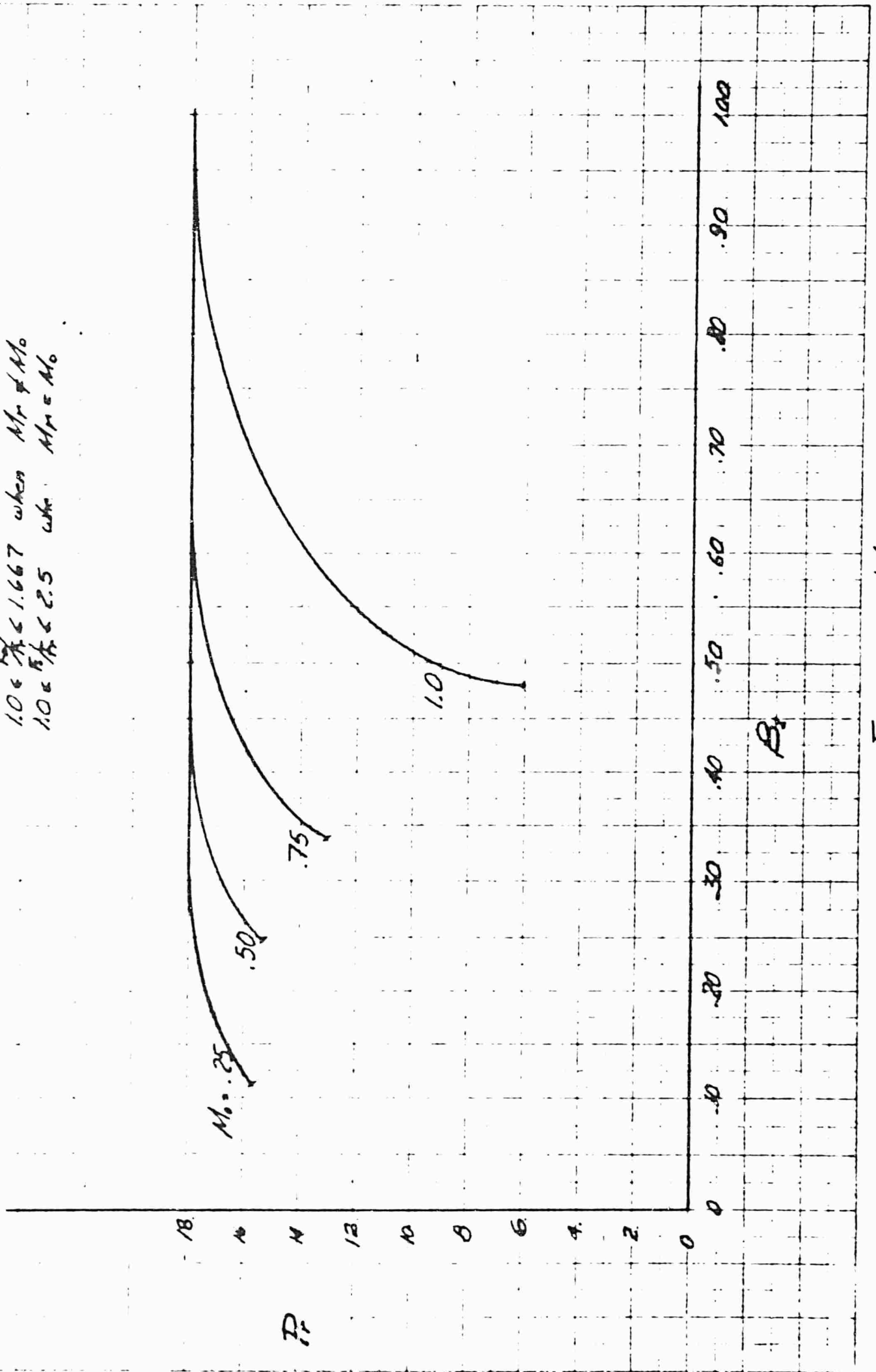


FIGURE 14

TURBOJET
SEA LEVEL

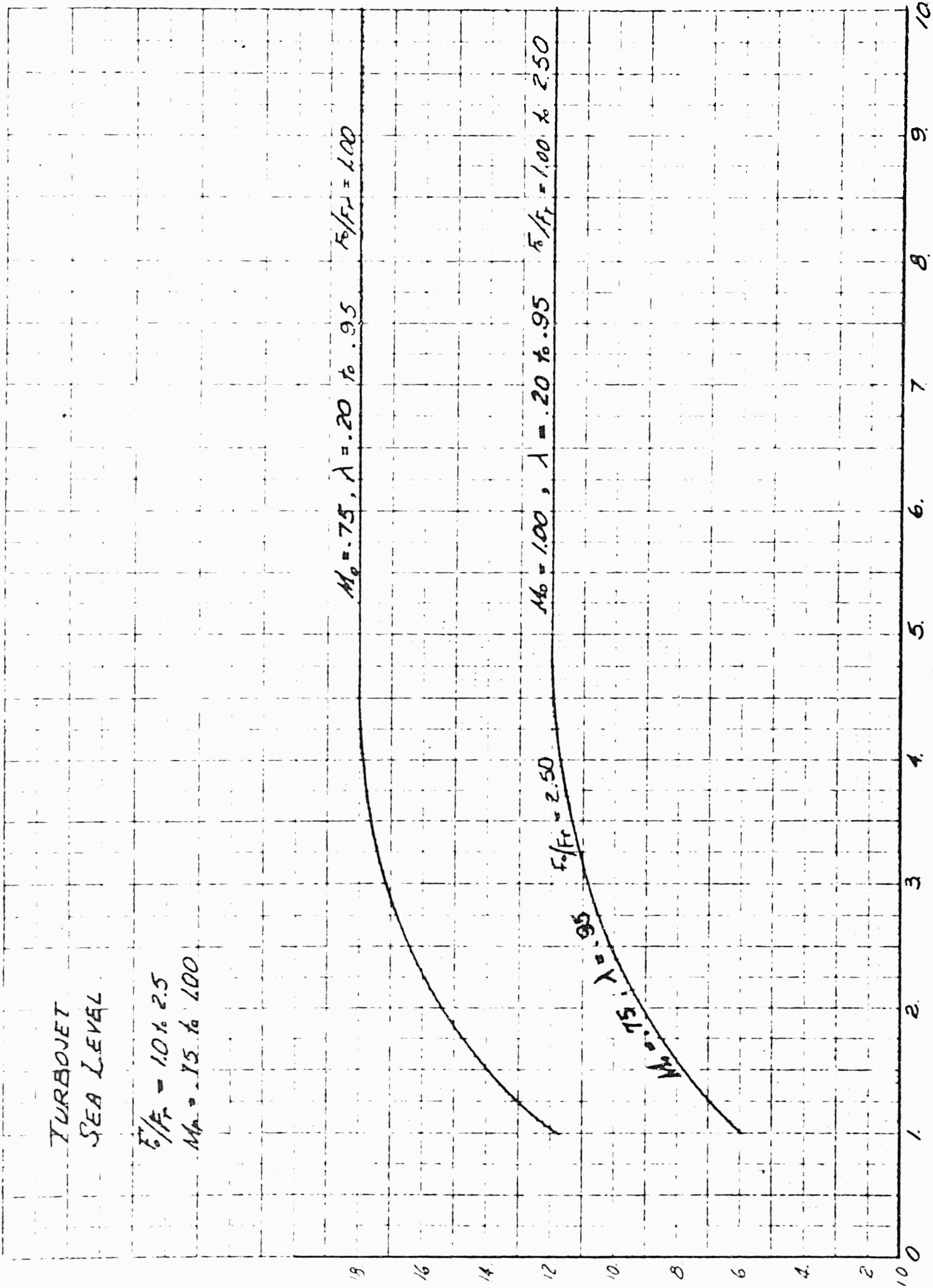
$F_0/F_T = 1.0$ to 2.5
 $M_0 = .75$ to 1.00

$M_0 = .75, \lambda = .20$ to $.95$ $F_0/F_T = 1.00$

$M_0 = 1.00, \lambda = .20$ to $.95$ $F_0/F_T = 1.00$ to 2.50

$M_0 = .75, \lambda = .95$

$F_0/F_T = 2.50$



B₁

FIGURE 15

TURBOJET
ALTITUDE - 35,332 ft.

$M_6 = .75$ to 1.0
 $M_4 = .75$ to 1.0

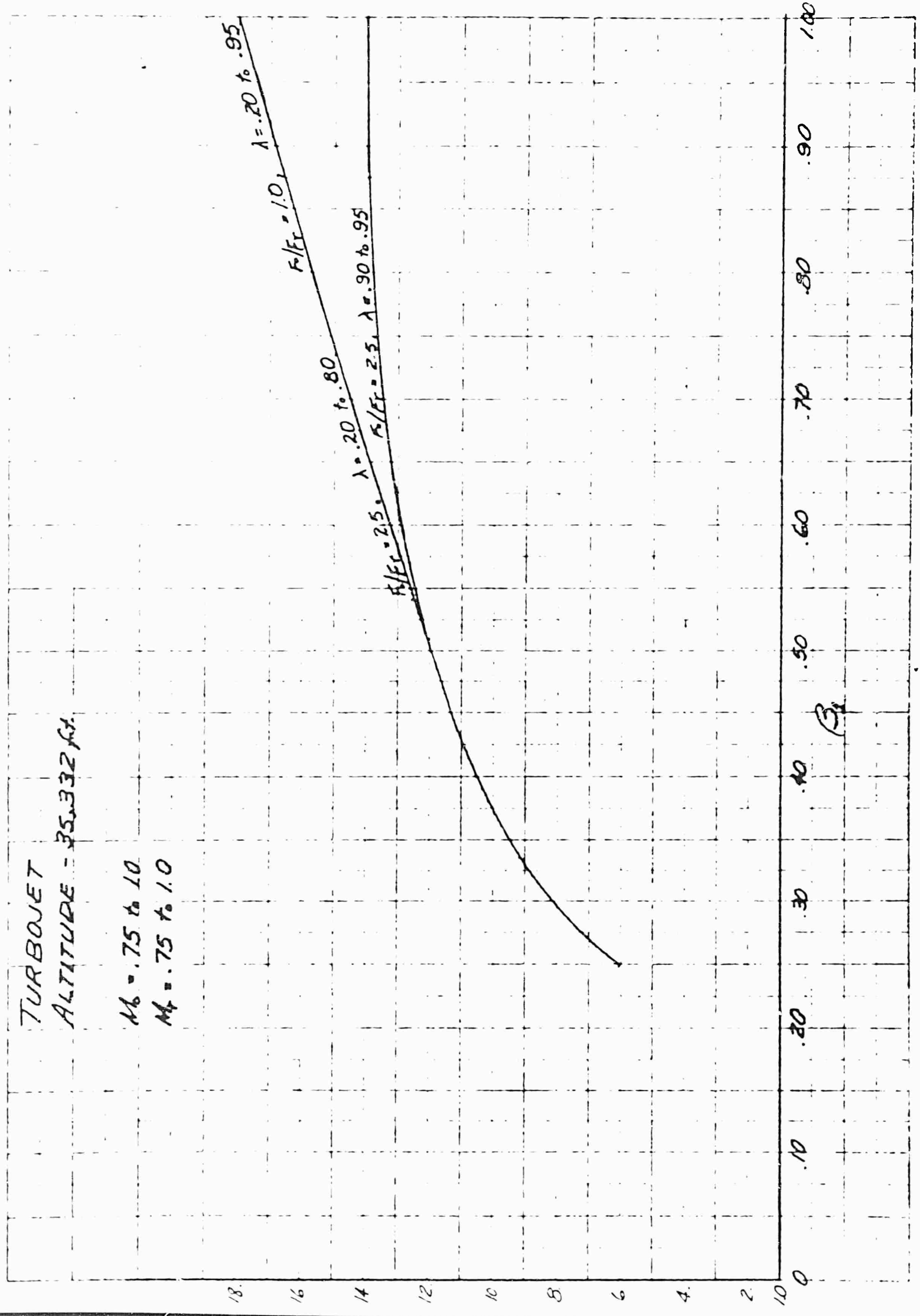


FIGURE 16

AFTERBURNING TURBOJET
SEA LEVEL

$F/F_0 = 1.0$ to 1.667

$\lambda = .20$ to $.95$

$M_T = .75$ to 1.00

$M_0 = .75, F/F_0 = 1.667$

$M_0 = .75, F/F_0 = 1.00$

$M_0 = 1.50, F/F_0 = 1.0$ to 2.5

$M_0 = 1.00$

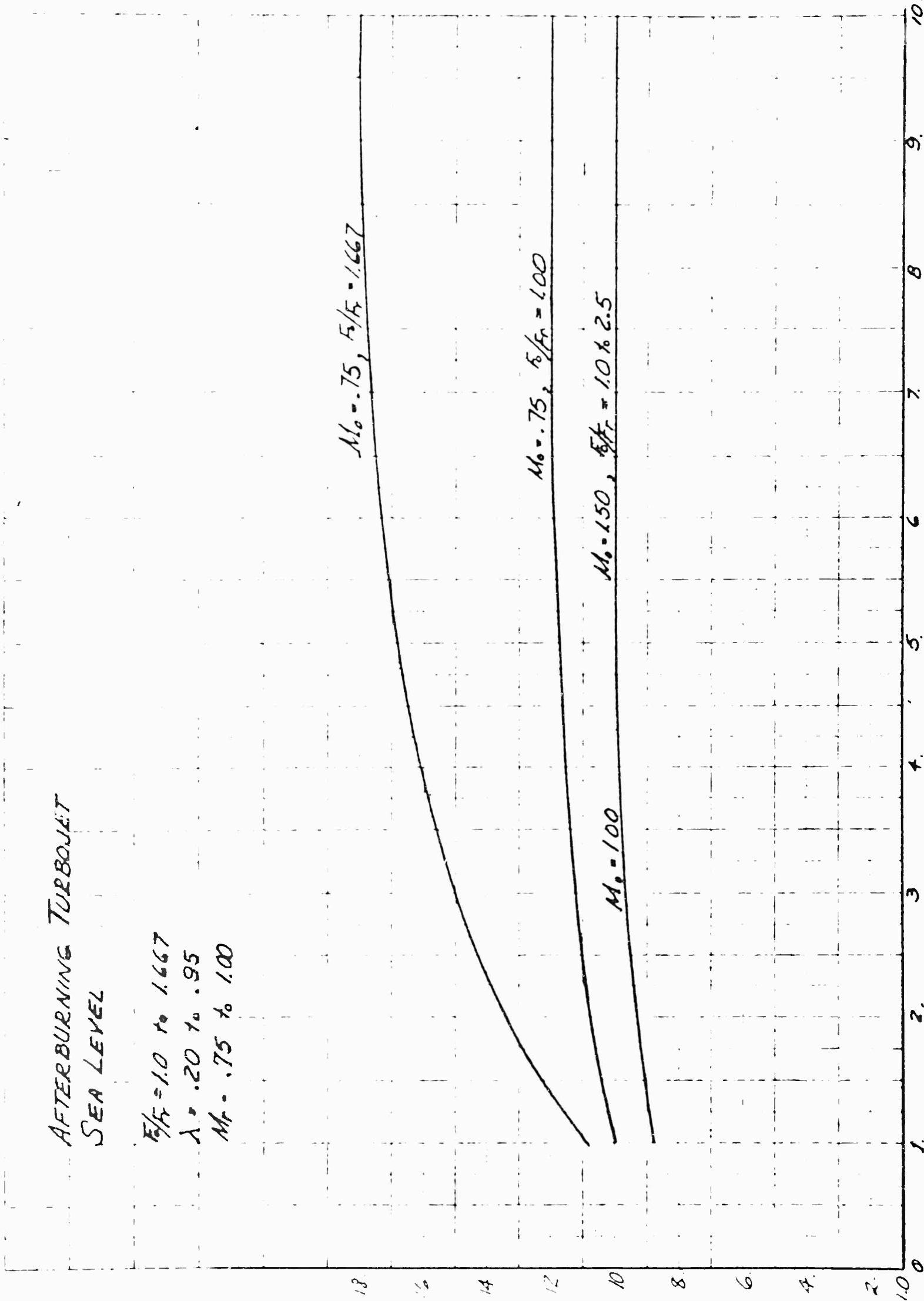


FIGURE 17
B1

AFTERBURNING TURBOJET SEA LEVEL

$M_0 = 1.00$
 $\lambda = .20 \text{ to } .45$
 $P/P_0 = 2.5$

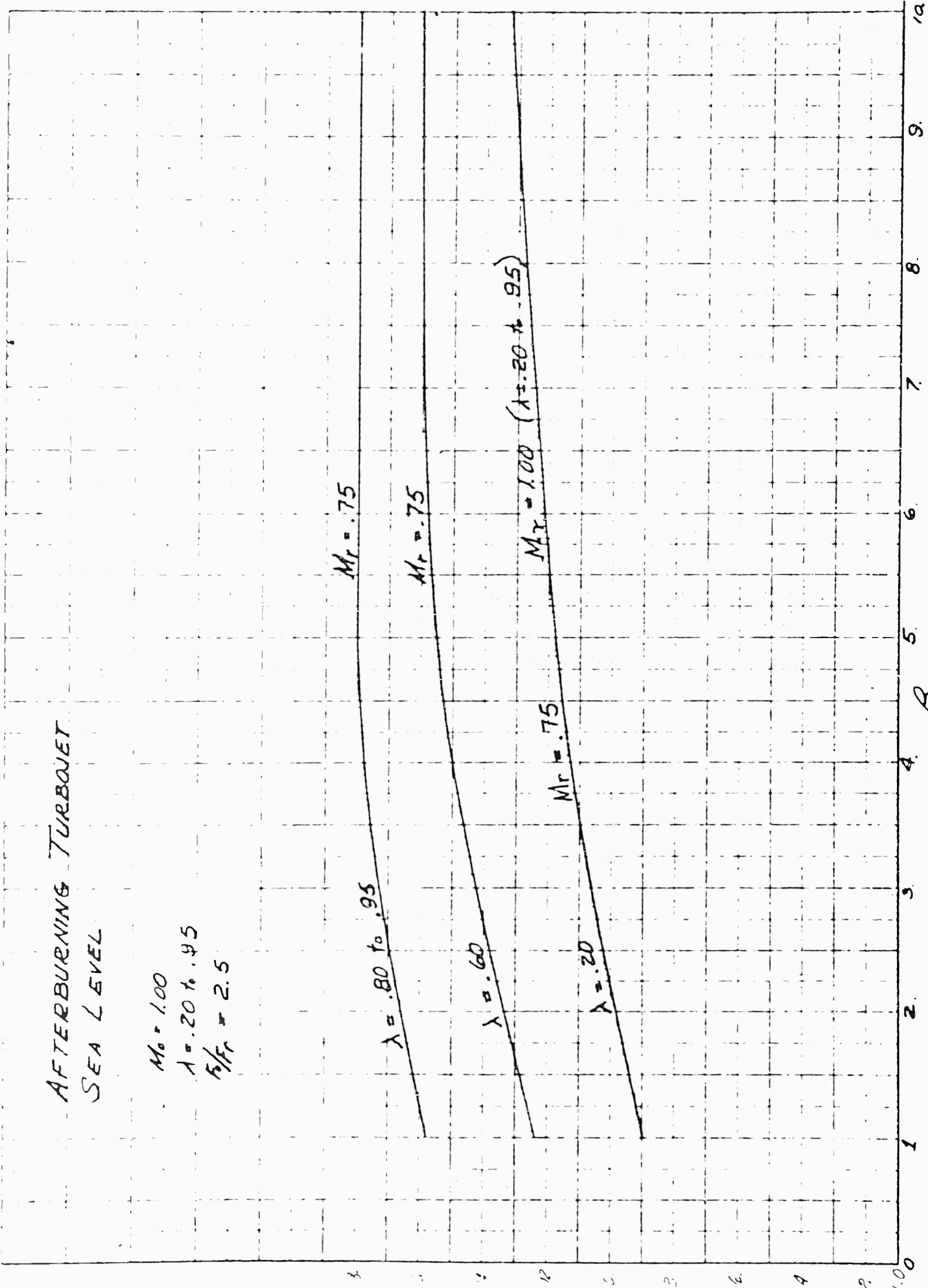
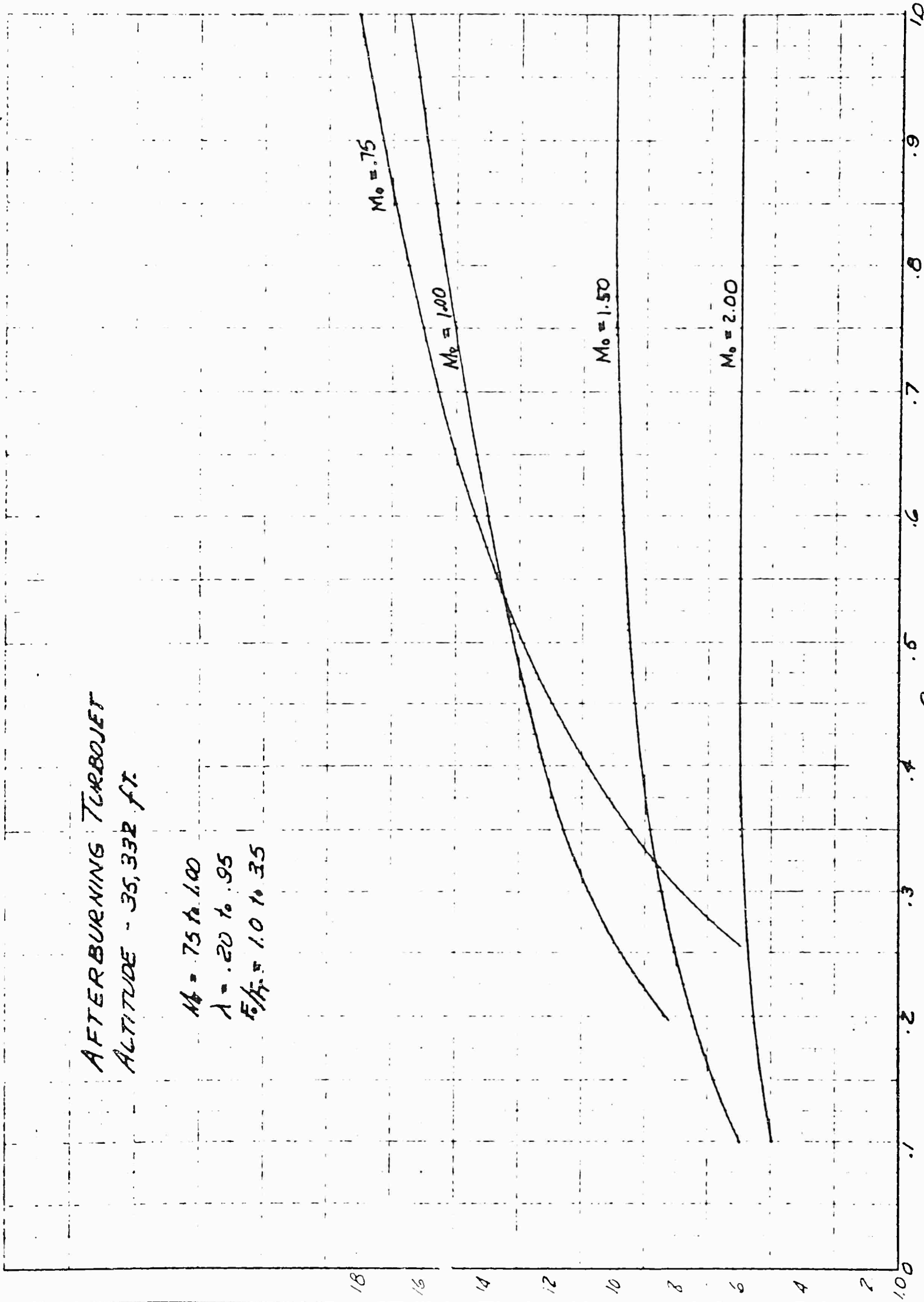


FIGURE 18

B₁

AFTERBURNING TURBOJET
ALTITUDE - 35,332 ft.

$M_0 = 75$ to 1.00
 $\lambda = .20$ to $.95$
 $F/\dot{W}_0 = 1.0$ to 3.5

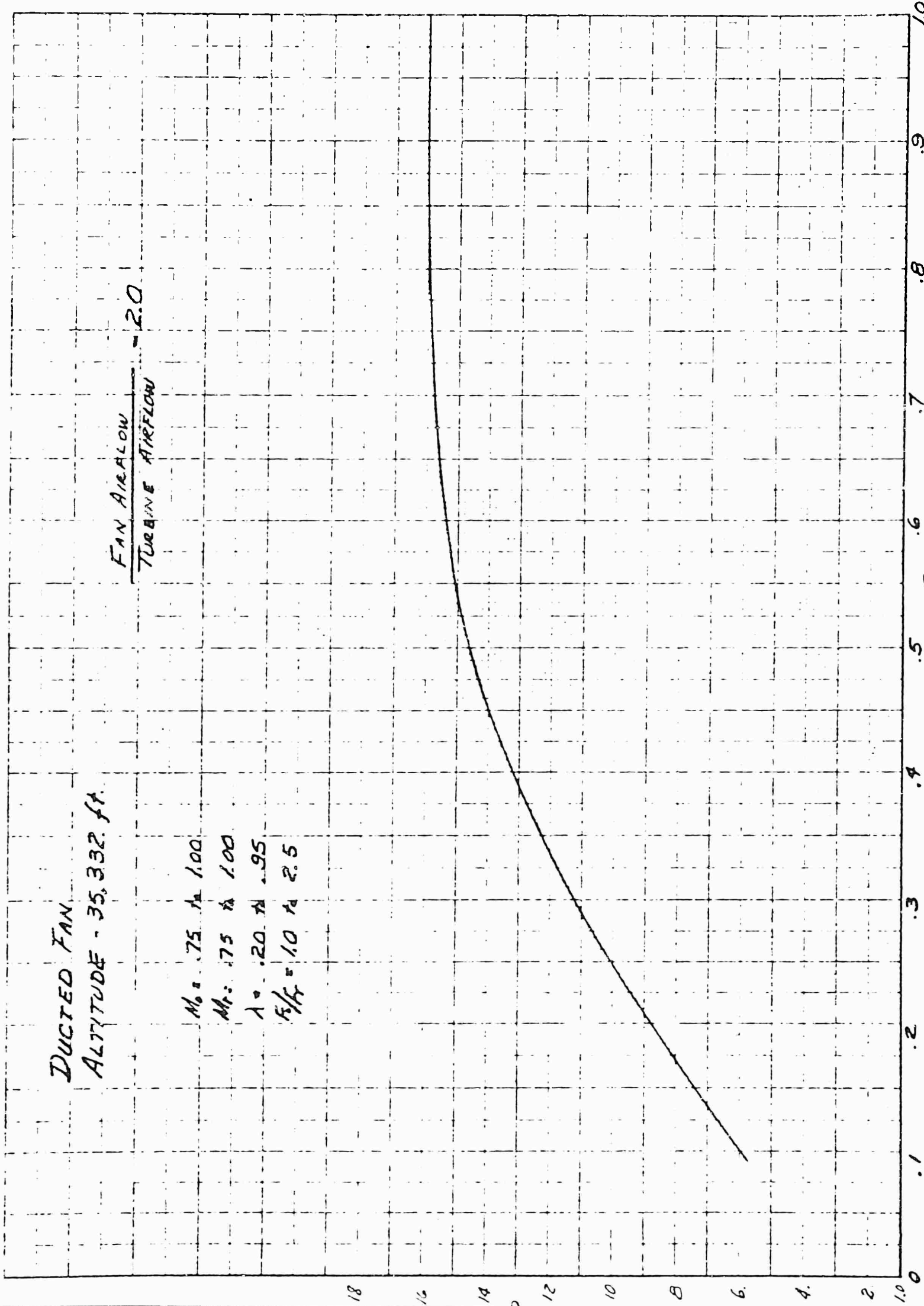


β_1 FIGURE 19

DUCTED FAN
ALTITUDE - 35,332 ft.

$M_0 = .75$ to 1.00
 $M_1 = .75$ to 1.00
 $\lambda = .20$ to $.95$
 $F/K = 1.0$ to 2.5

$\frac{\text{FAN AIRFLOW}}{\text{TURBINE AIRFLOW}} = 2.0$



B₁ FIGURE 20

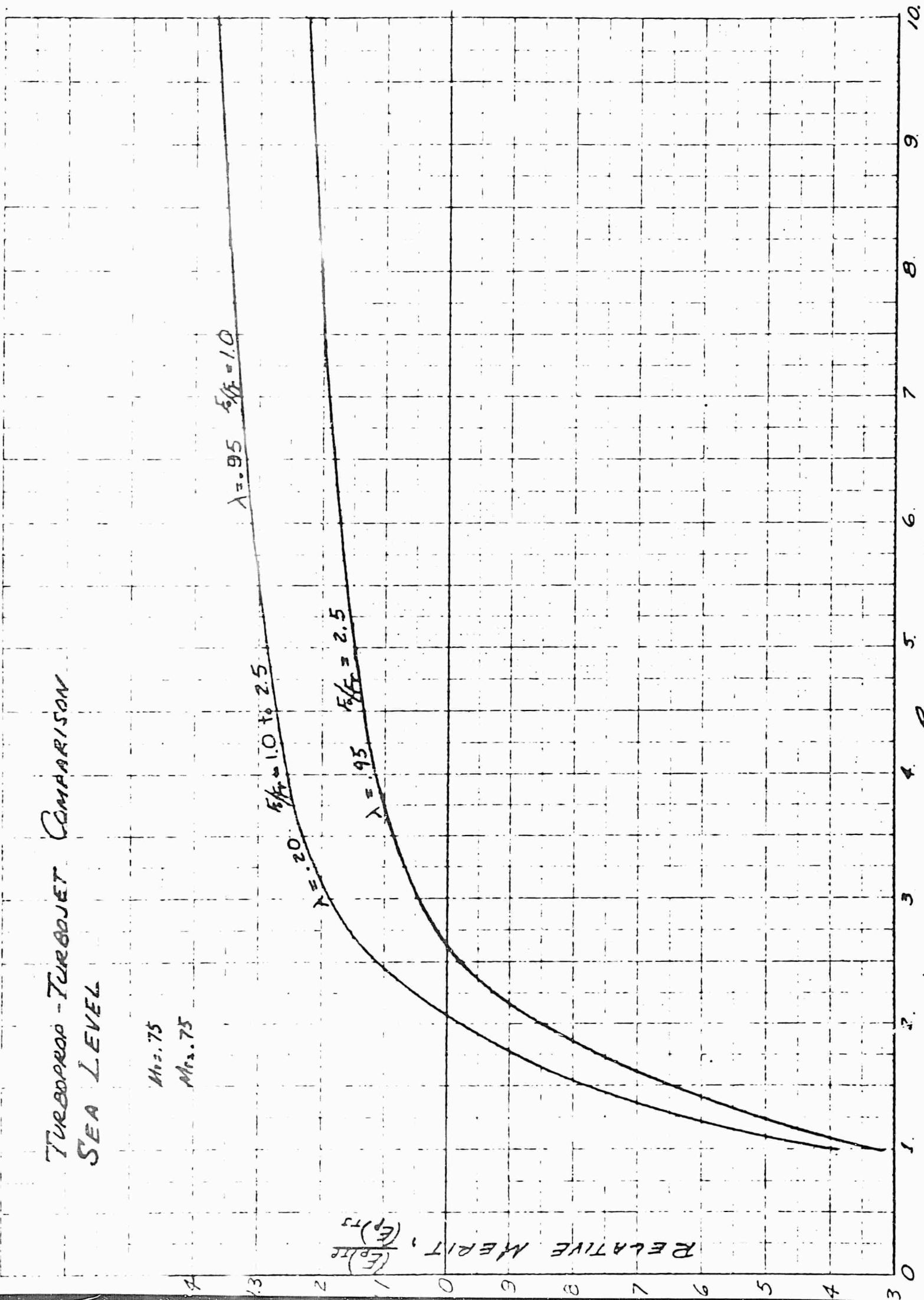


FIGURE 21

TURBOJET - TURBOPROP COMPARISON SEA LEVEL

$M_0 = 1.00$
 $M_p = .75$ to 1.00
 $\lambda = .20$ to $.95$
 $F/F_0 = 1.0$ to 1.25

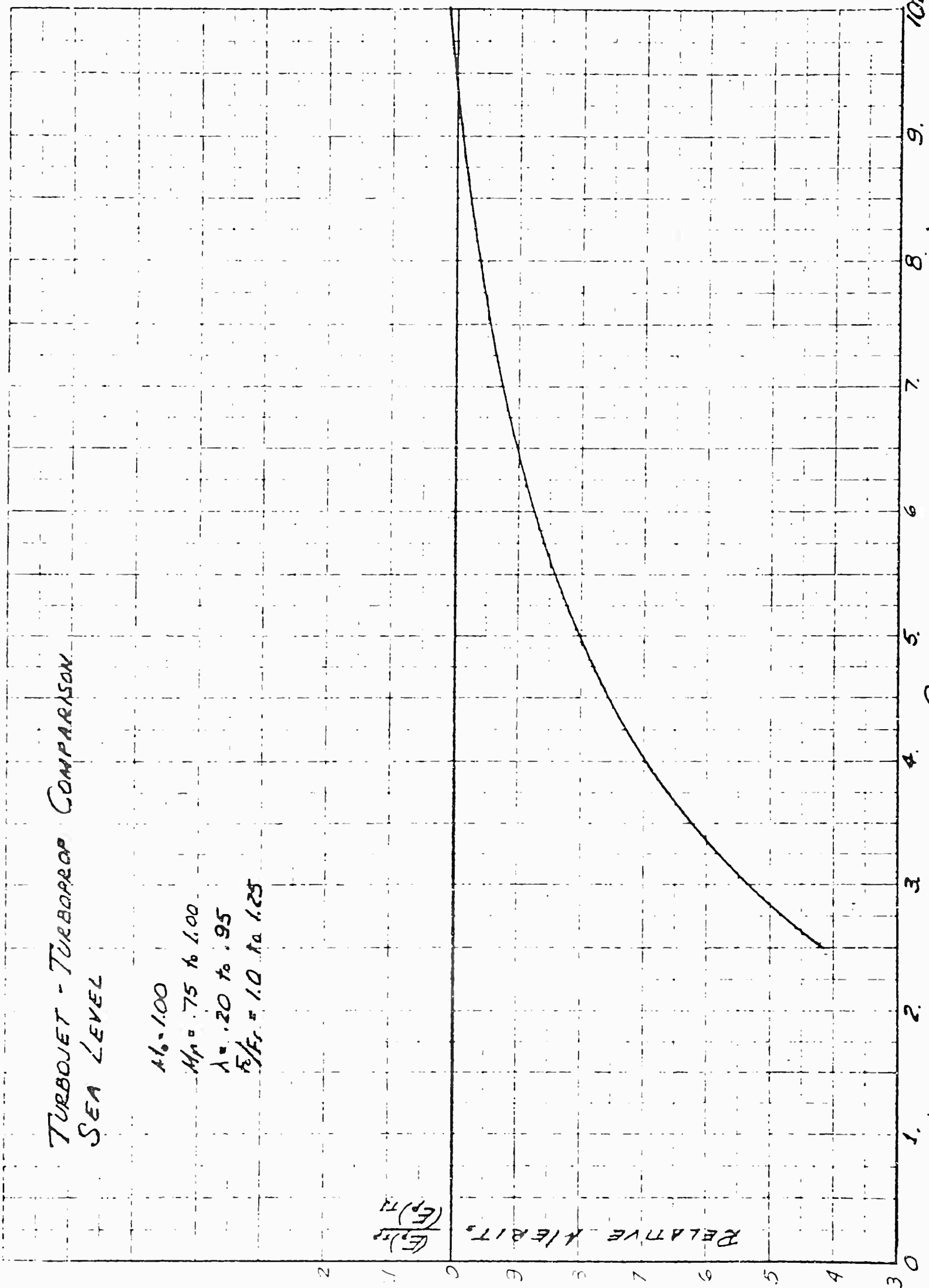


FIGURE 22

TURBOPROP - TURBOJET COMPARISON
ALTITUDE - 35,332 ft.

$M = .75$
 $M_r = .75$

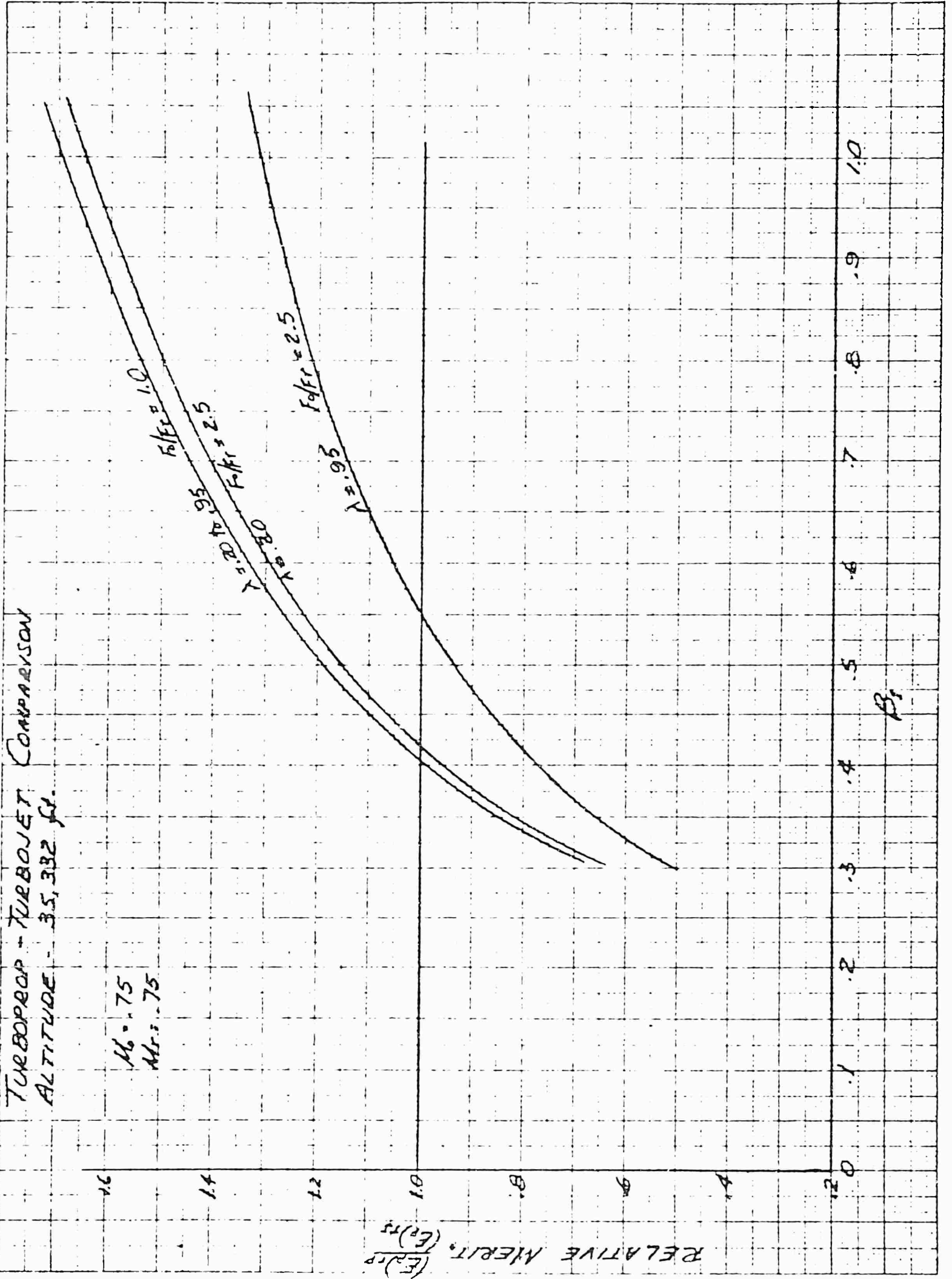


FIGURE 23

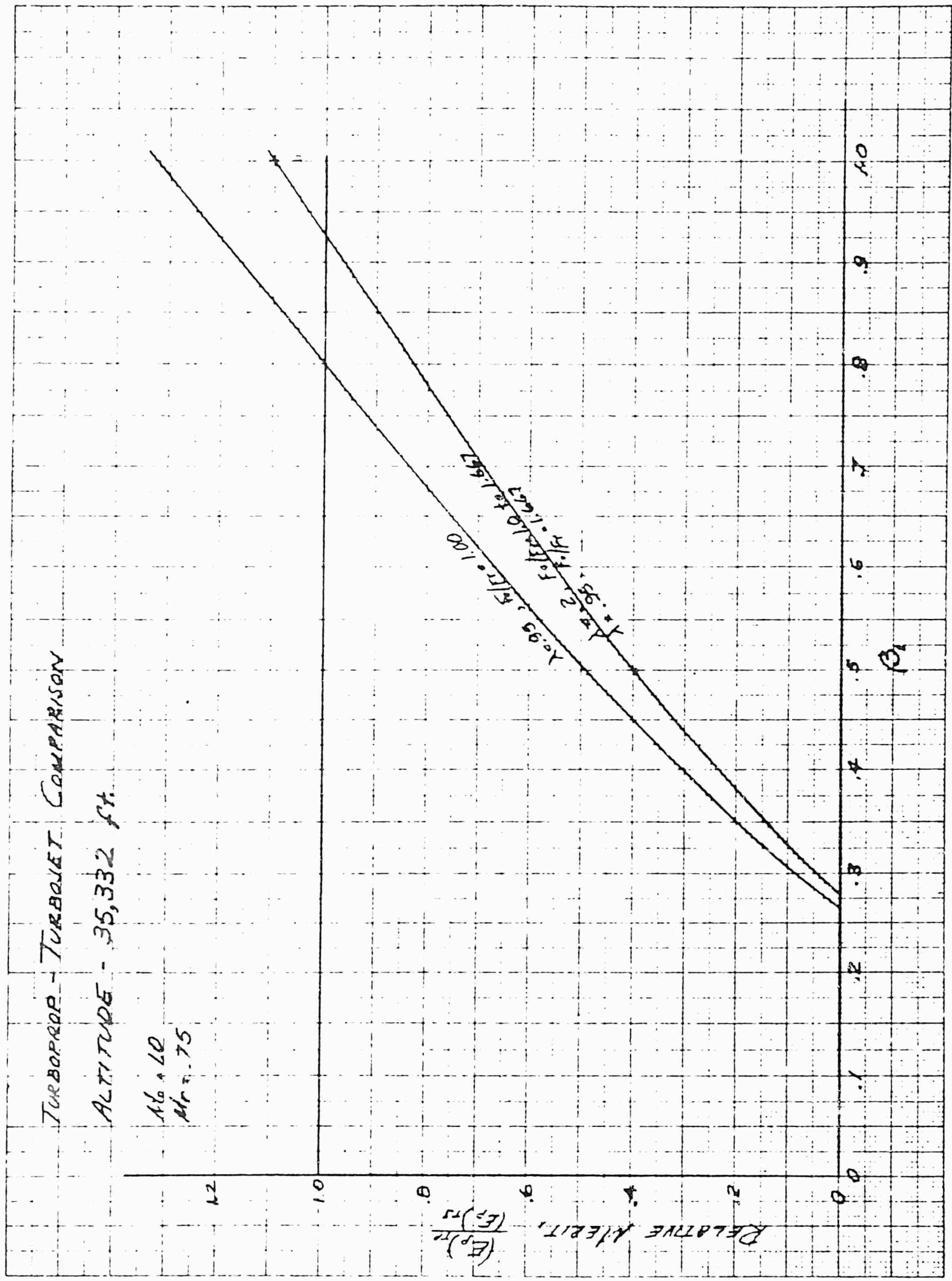


FIGURE 24

TURBOPROP - TURBOJET COMPARISON

ALTITUDE - 35,332 ft

$M_0 = 1.2$

$M_1 = 1.0$

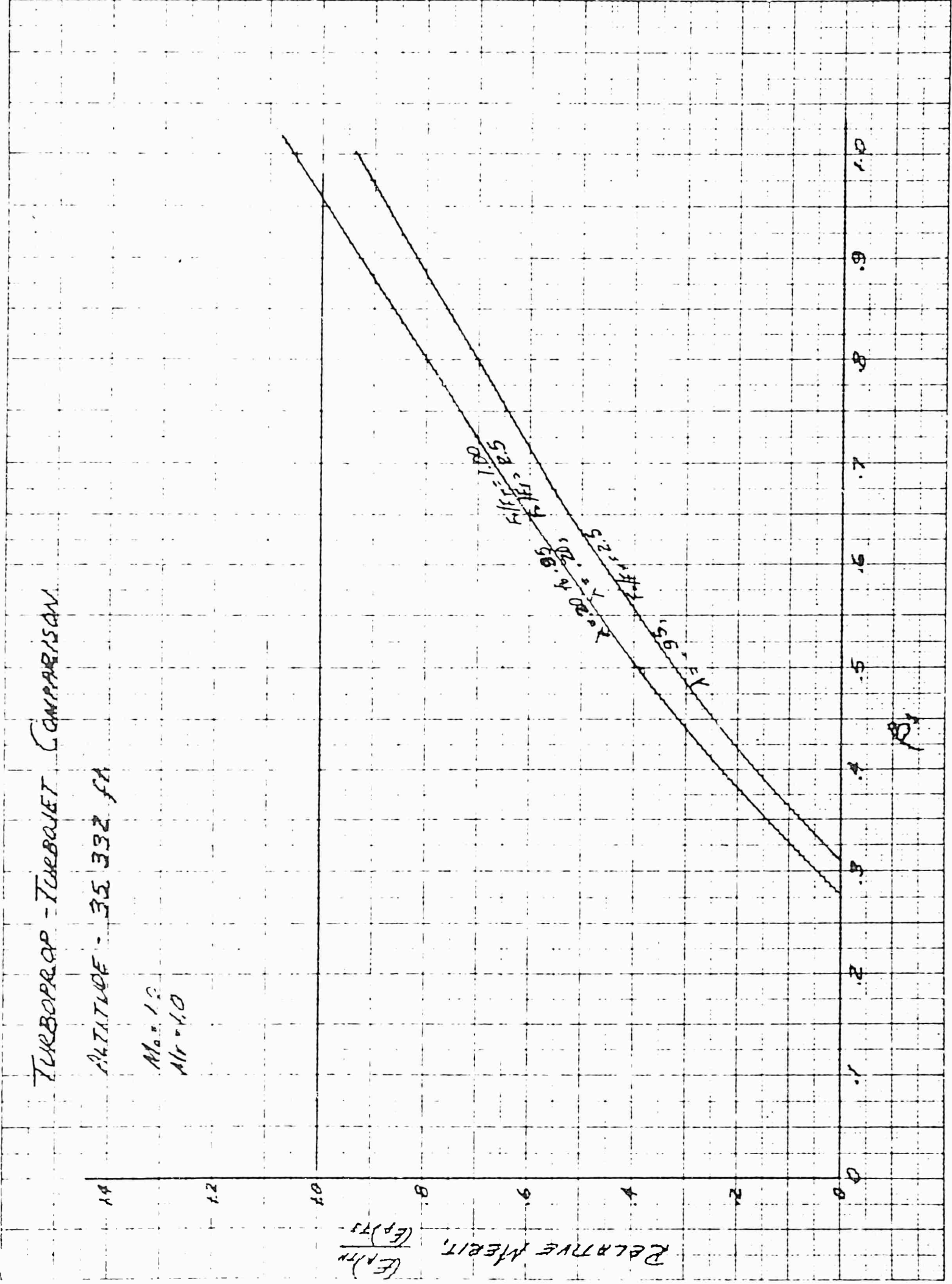


FIGURE 25

AFTERSURNING TURBOJET TO TURBOJET WITHOUT AFTERSURNER COMPARISON
SEA LEVEL

$M_0 = 1.00$
 $M_1 = .75$

$\lambda = .95$
 $R/F_T = 2.5$

$\lambda = .20$
 $R/F_T = 1.0 \text{ to } 2.5$

RELATIVE ALTITUDE (ft)

10 9 8 7 6 5 4 3 2 1 0

1 2 3 4 5 6 7 8 9 10

B_2

FIGURE 26

AFTERBURNING TURBOJET TO TURBOJET WITHOUT AFTERBURNER COMPARISON

SEA LEVEL

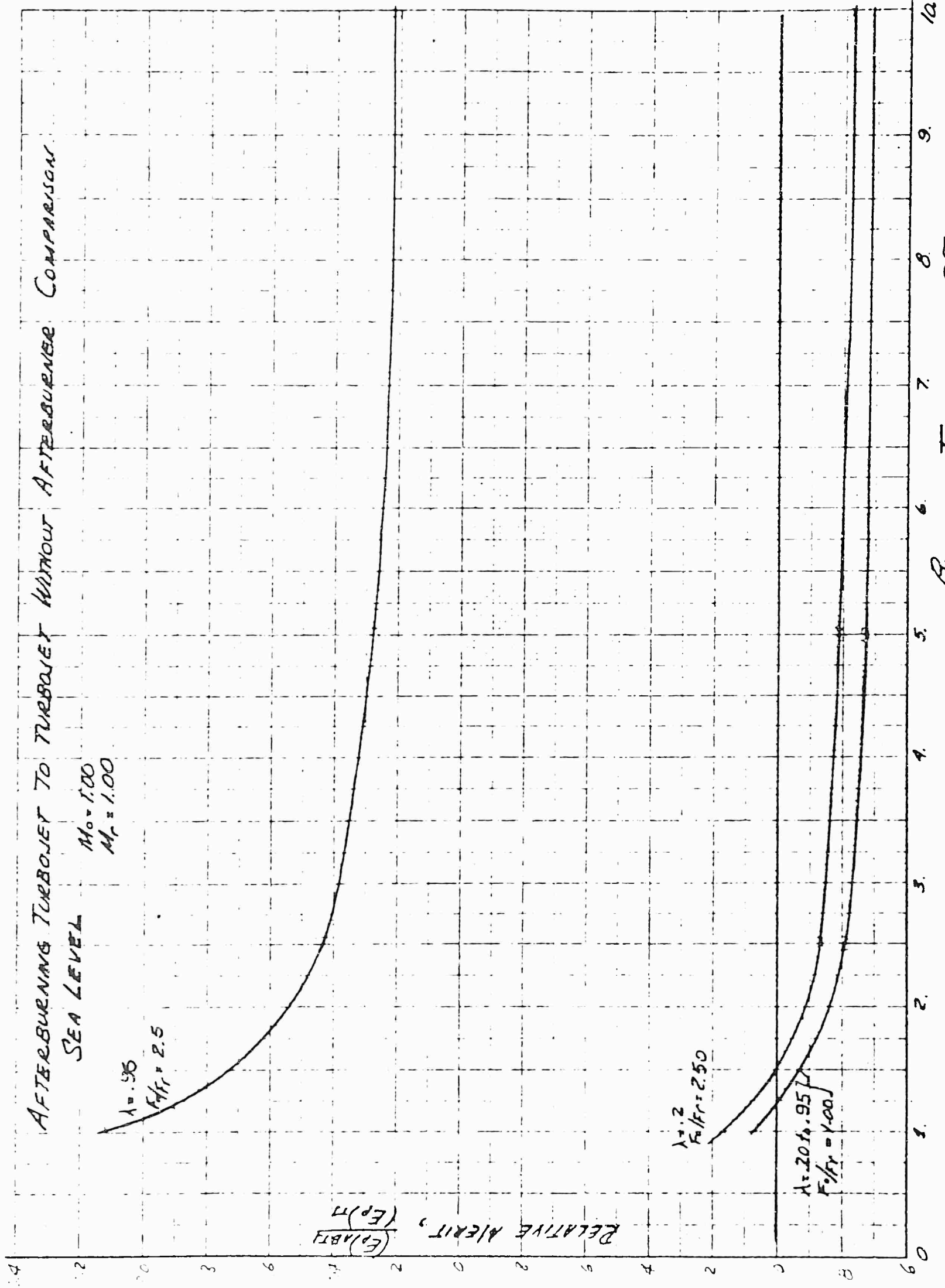
$M_0 = 1.00$
 $M_T = 1.00$

$\lambda = .96$
 $F_0/F_T = 2.5$

$\lambda = .2$
 $F_0/F_T = 2.50$

$\lambda = .20$, $.95$
 $F_0/F_T = 1.00$

RELATIVE MERIT, $(E_p)/M$



B₁ FIGURE 27

AFTERBURNING TURBOJET TO TURBOJET WITHOUT AFTERBURNER COMPARISON
ALTITUDE - 35,332 ft.

$M_0 = 1.00$
 $M_1 = .75$

$\lambda = .95$
 $F_0/F_1 = .85$

$\lambda = .70$
 $F_0/F_1 = 1.0$ to 2.5

RELATIVE HEAT, $\frac{(E)_{alt}}{(E)_{no alt}}$

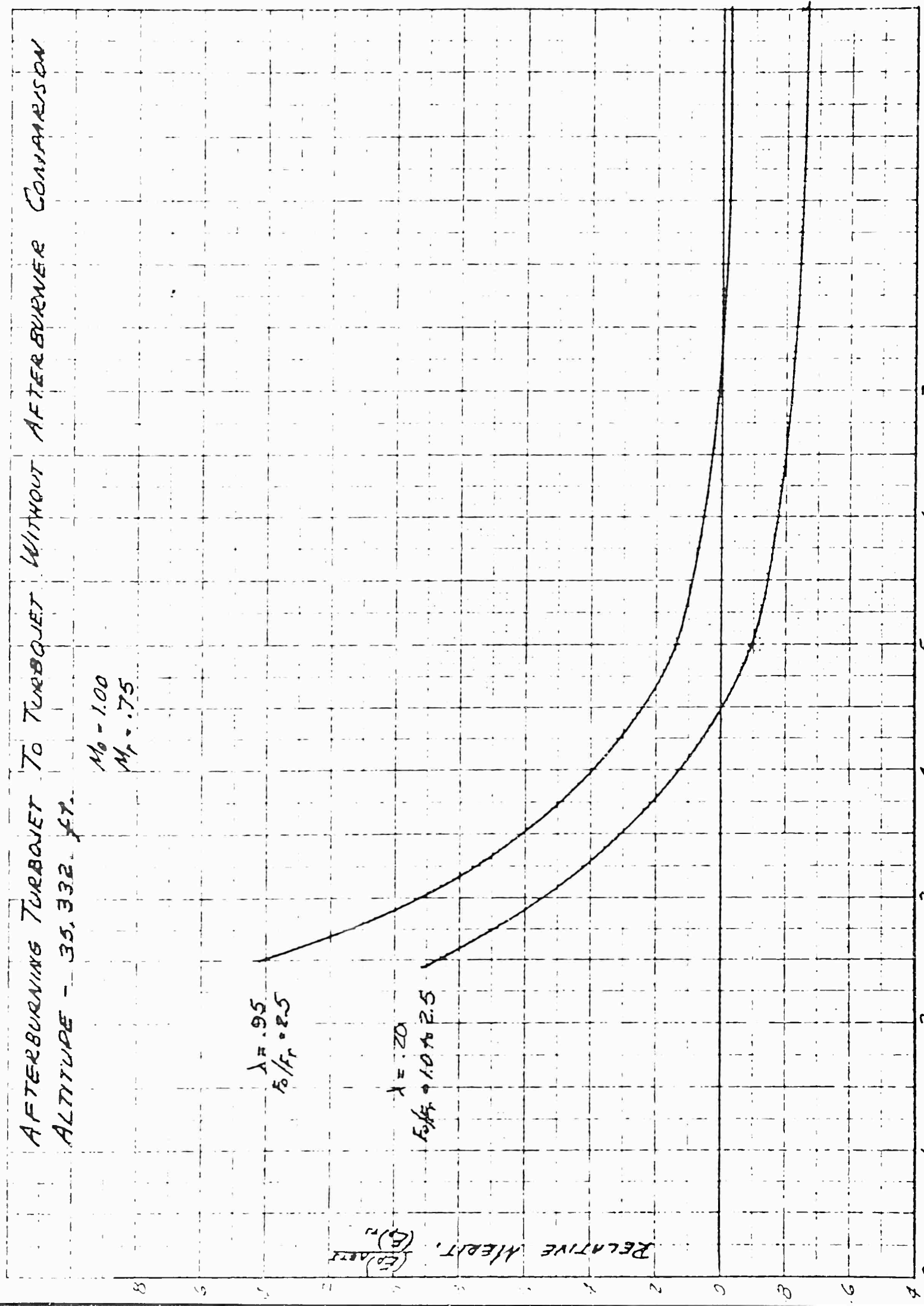


FIGURE 28

AFTERBURNING TURBOJET TO TURBOJET WITHOUT AFTERBURNER COMPARISON ALTITUDE - 35,332. FT.

$M_0 = 1.00$
 $M_T = 1.00$

$\lambda = .95$
 $F_{p/A} = .25$

$\lambda = .20$
 $F_{p/A} = 10.1625$
 $\lambda = .95$
 $F_{p/A} = 10$

RELATIVE MERIT, $\frac{(E_p)_{ABT}}{(E_p)_{TJ}}$

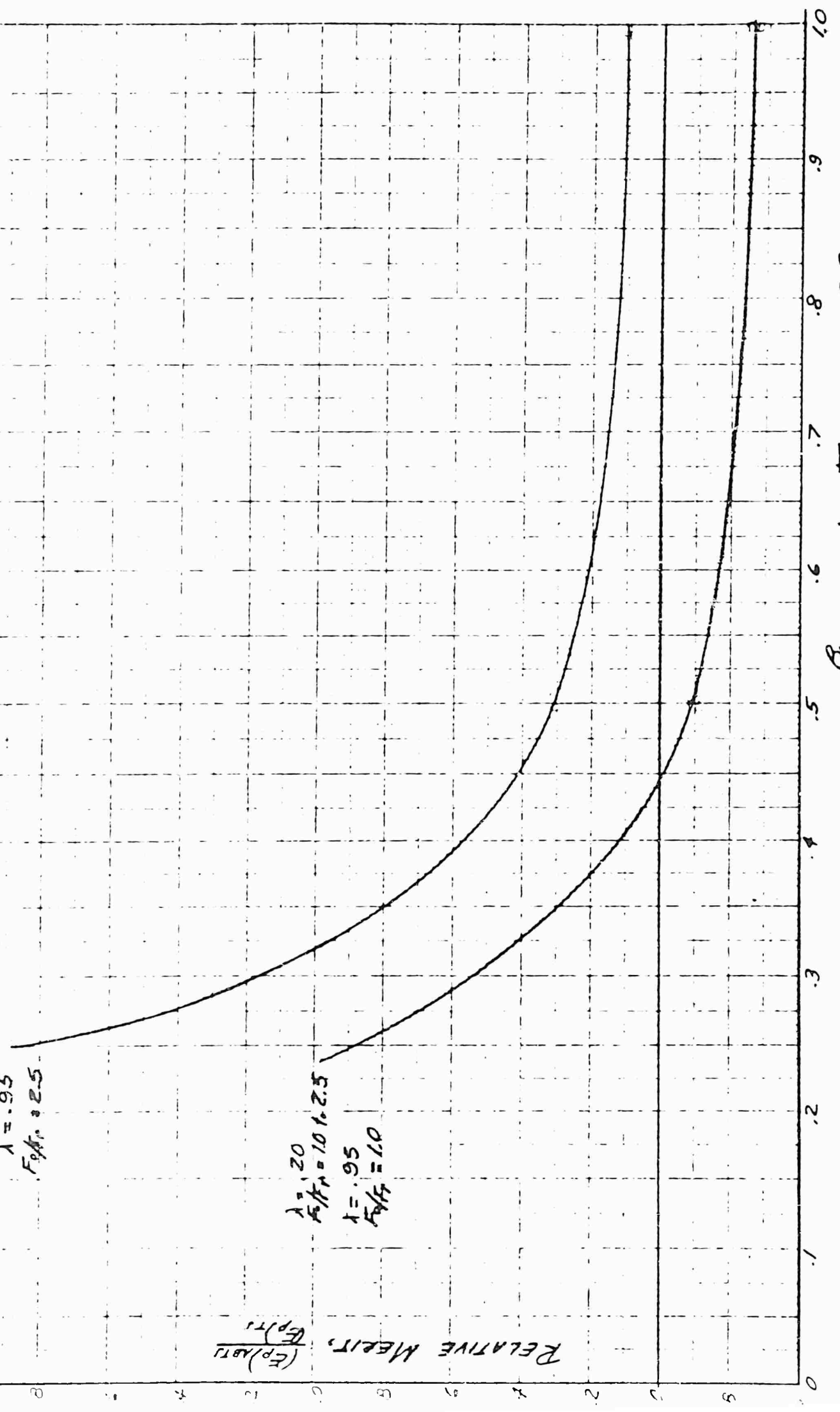


FIGURE 29

AFTERBURNING TURBOJET TO TURBOJET WITHOUT AFTERBURNER COMPARISON
 ALTITUDE - 35,392. FT.

$M_0 = 1.00$
 $M_T = 1.00$

$\lambda = .95$
 $F/A_T = .25$

$\lambda = .20$
 $F/A_T = 101.25$
 $\lambda = .95$
 $F/A_T = 1.0$

RELATIVE MERIT, $(E_p)_{M/T}$
 $(E_p)_{M/T}$

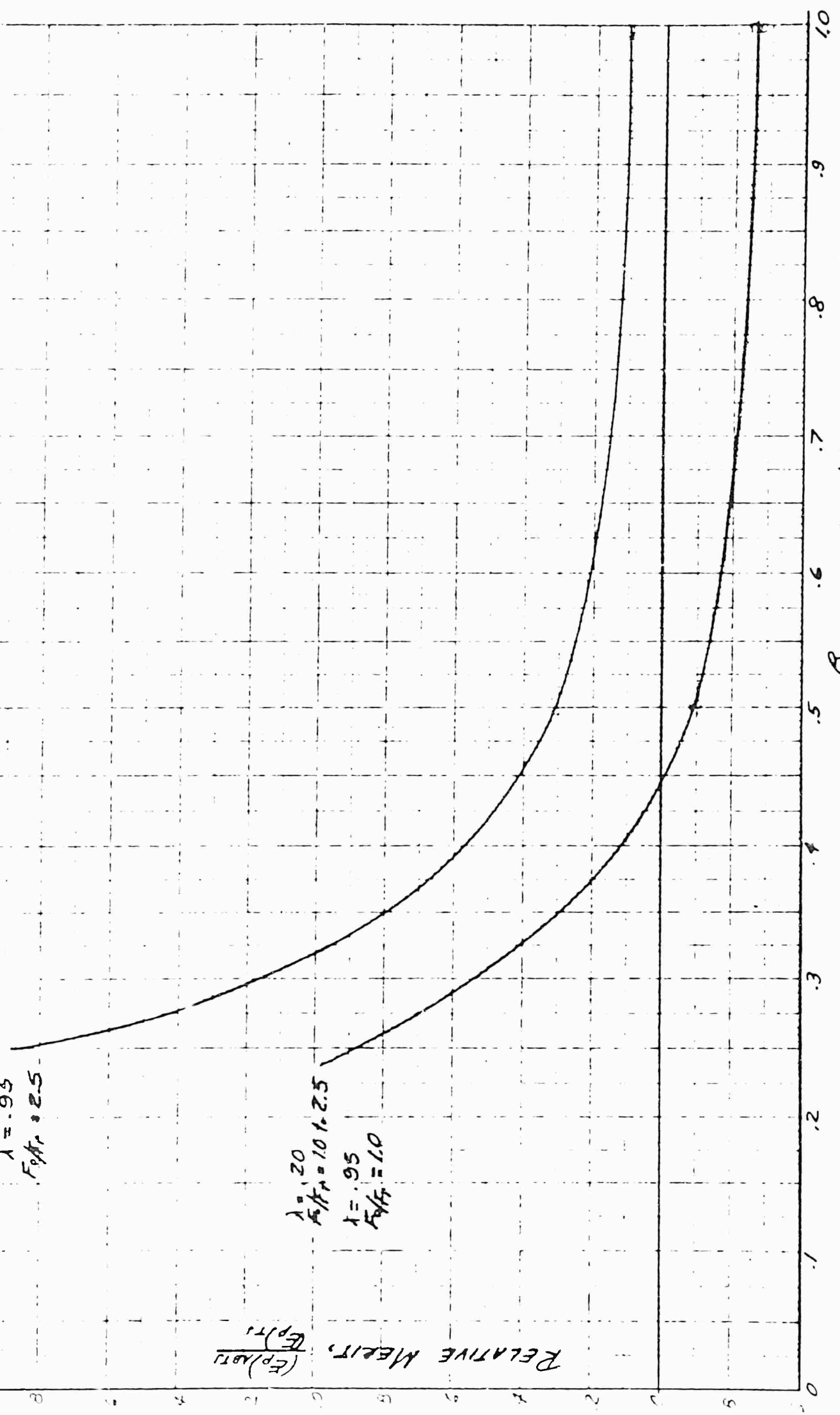


FIGURE 29

AFTERBURNING DUCTED FAN TO AFTERBURNING TURBOJET COMPARISON
 ALTITUDE - 35,332. ft

$M_0 = 1.00$
 $M_T = .75$

RELATIVE MERIT,
 $\frac{(E_p)_{ADP}}{(E_p)_{TJ}}$

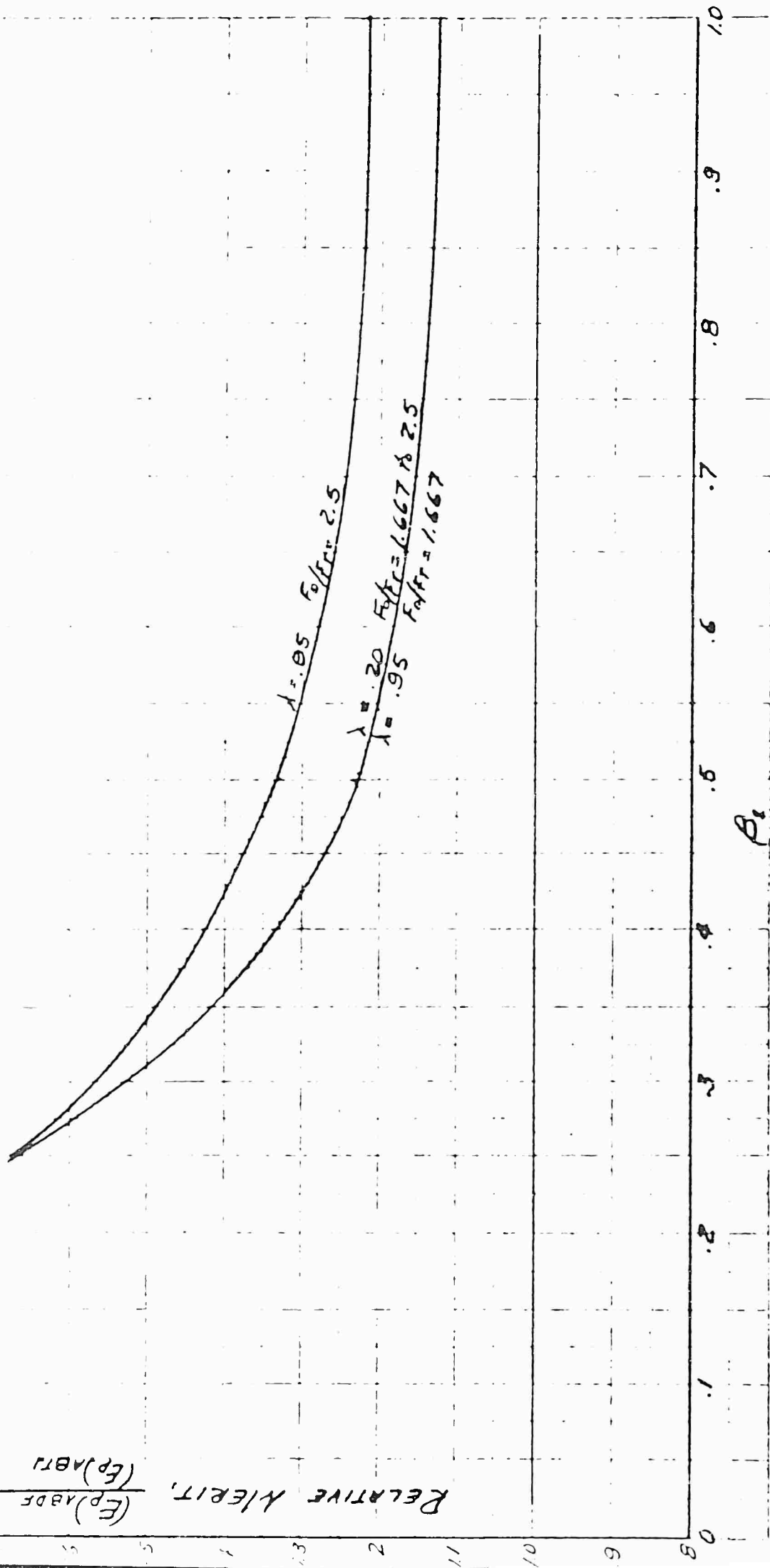


FIGURE 30

DUCTED FAN TO AFTERBURNING TURBOJET COMPARISON

ALTITUDE - 35,332 ft.

$M_0 = 1.00$

$M_T = 1.00$

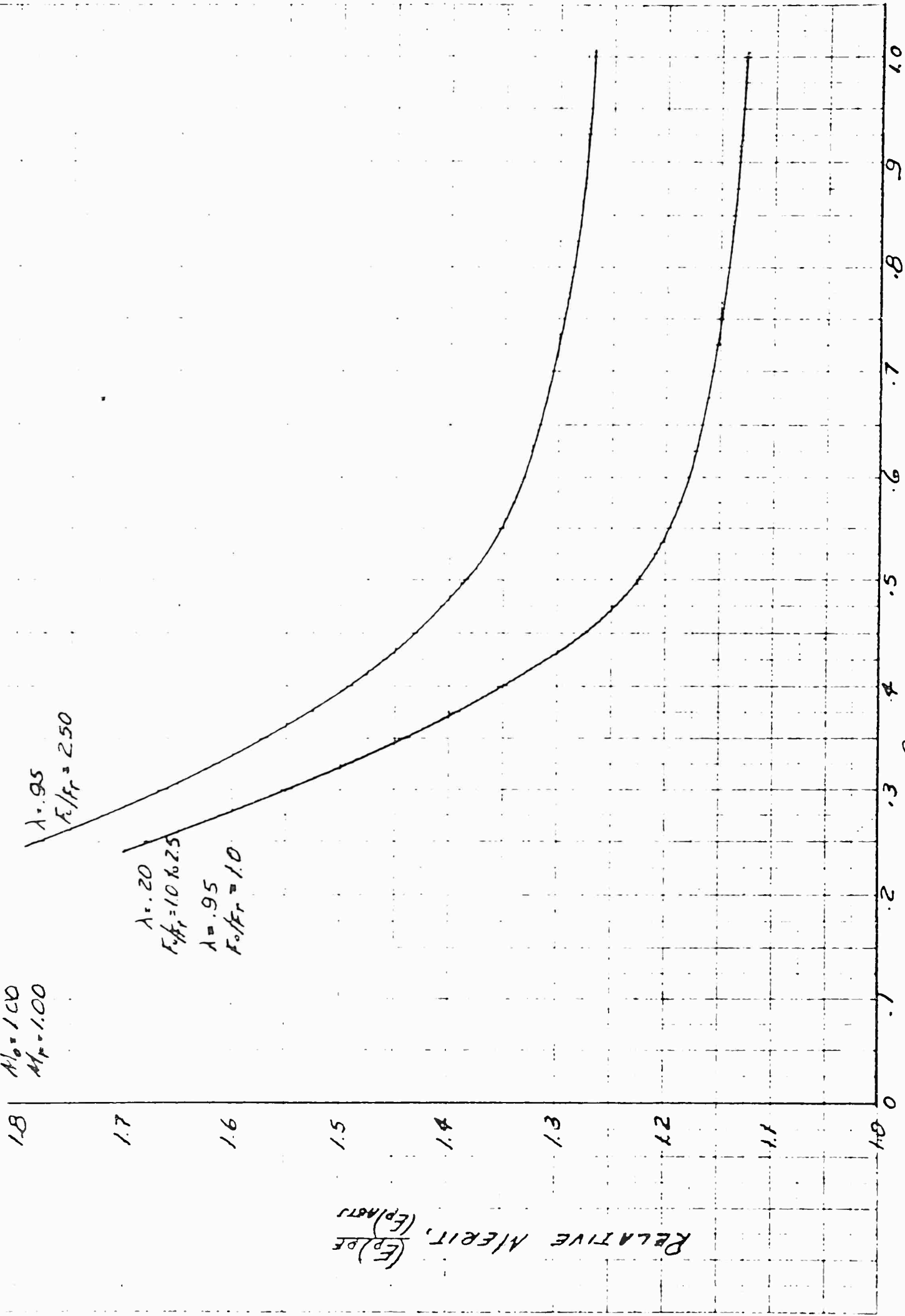


FIGURE 31

β

AFTERBURNING TURBOJET AND ROCKET
TO AFTERBURNING TURBOJET COMPARISON

ALTITUDE - 55,332 ft

$\lambda = .95$
 $M_T = .75$

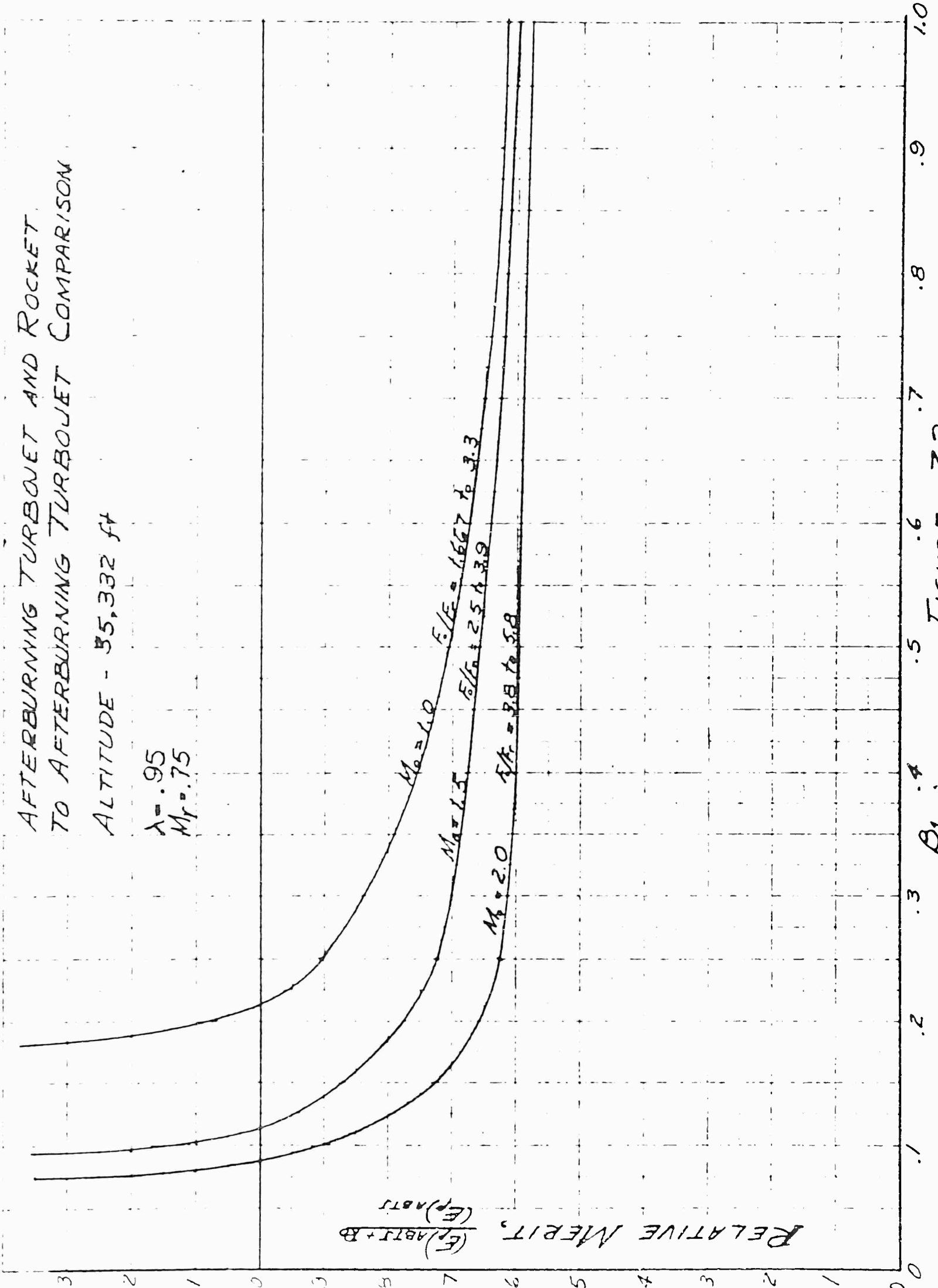


FIGURE 32

B_1

AFTERBURNING TURBOJET AND RAMJET TO AFTERBURNING TURBOJET COMPARISON

ALTITUDE - 35,332 ft.
 $\lambda = 0.20$ TO 0.95
 $M_T = 0.75$

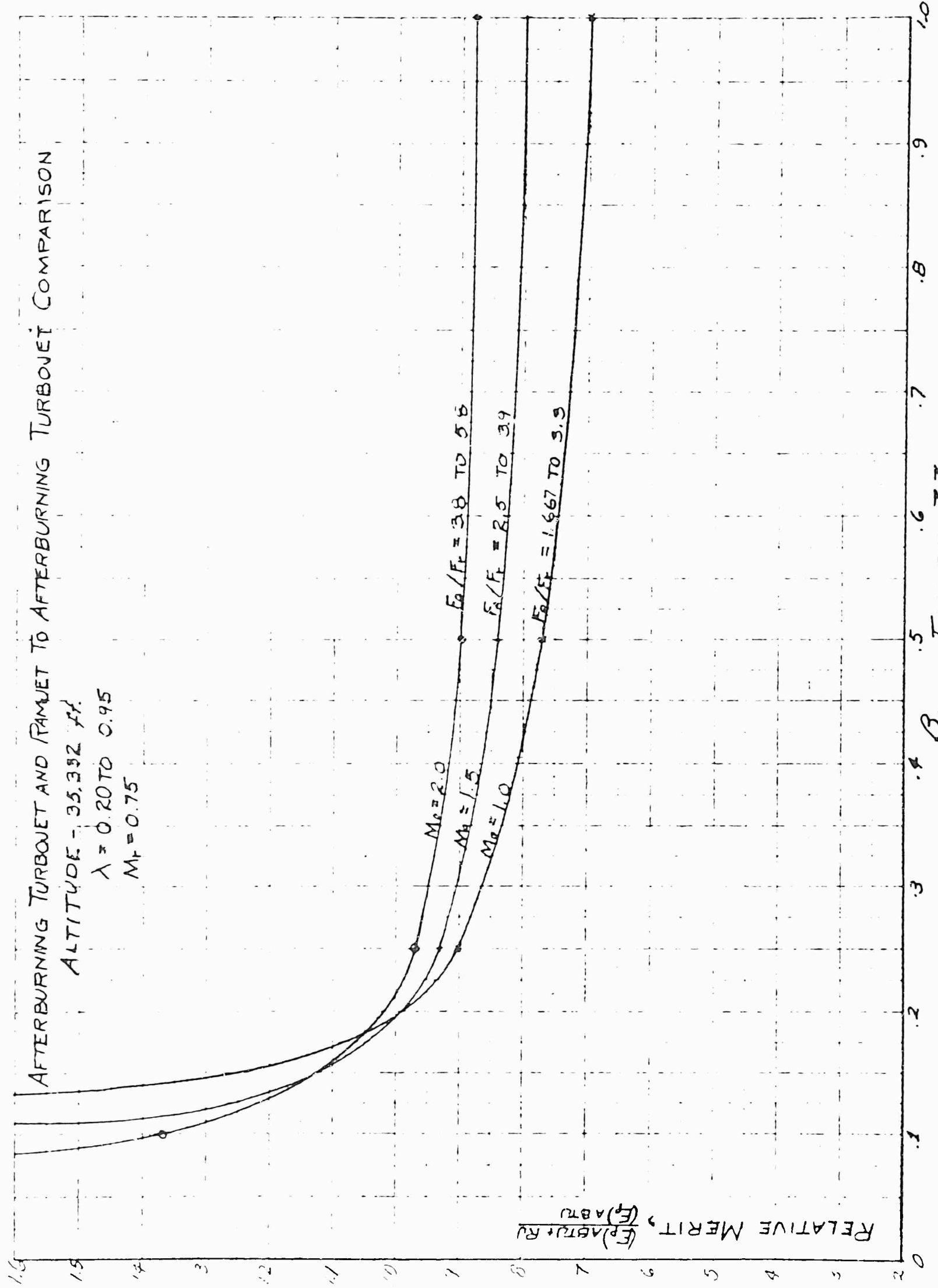


FIGURE 33

COMPARISON OF ENGINES AT SEA LEVEL

$\lambda = 1.0$
 $F_0/F_1 = 1.0$
 MACH 0.25 and 0.5

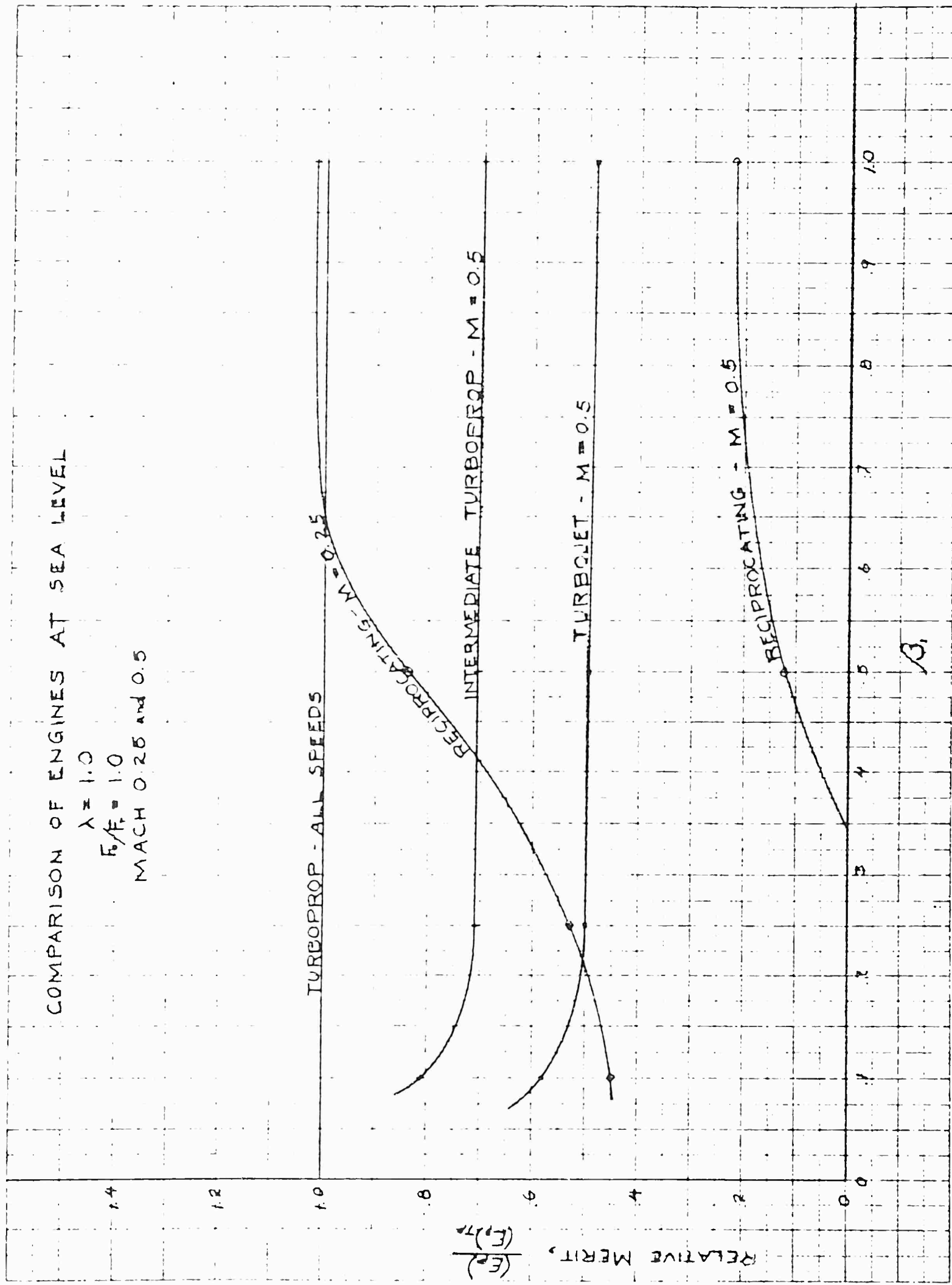


FIGURE 34

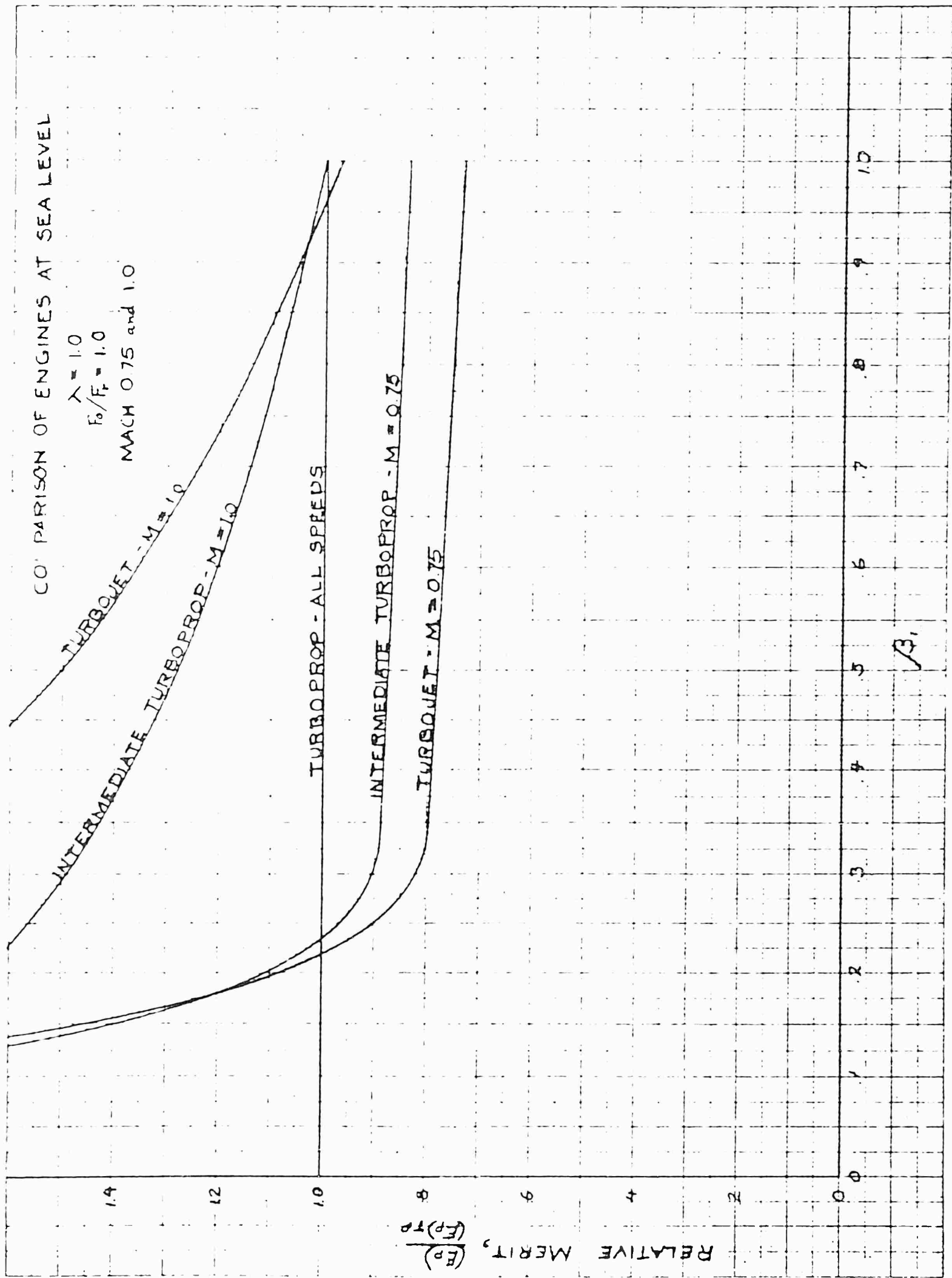


FIGURE 35

ENGINES COMPARISON

ALTITUDE - 35,332 FT.

$$\lambda = 1.0$$

$$\frac{P}{P_0} = 1.0$$

$$M = .25$$

RECIPROCATING

TURBOPROP

RELATIVE MERIT

12
10
8
6
4
2
0

0 .2 .4 .6 .8 1.0 1.2 1.4 1.6 1.8 2.0

FIGURE 36

ENGINES COMPARISON

ALTITUDE - 35,332 ft.

$\lambda = 1.0$

$F/H = 1.0$

MACH .50

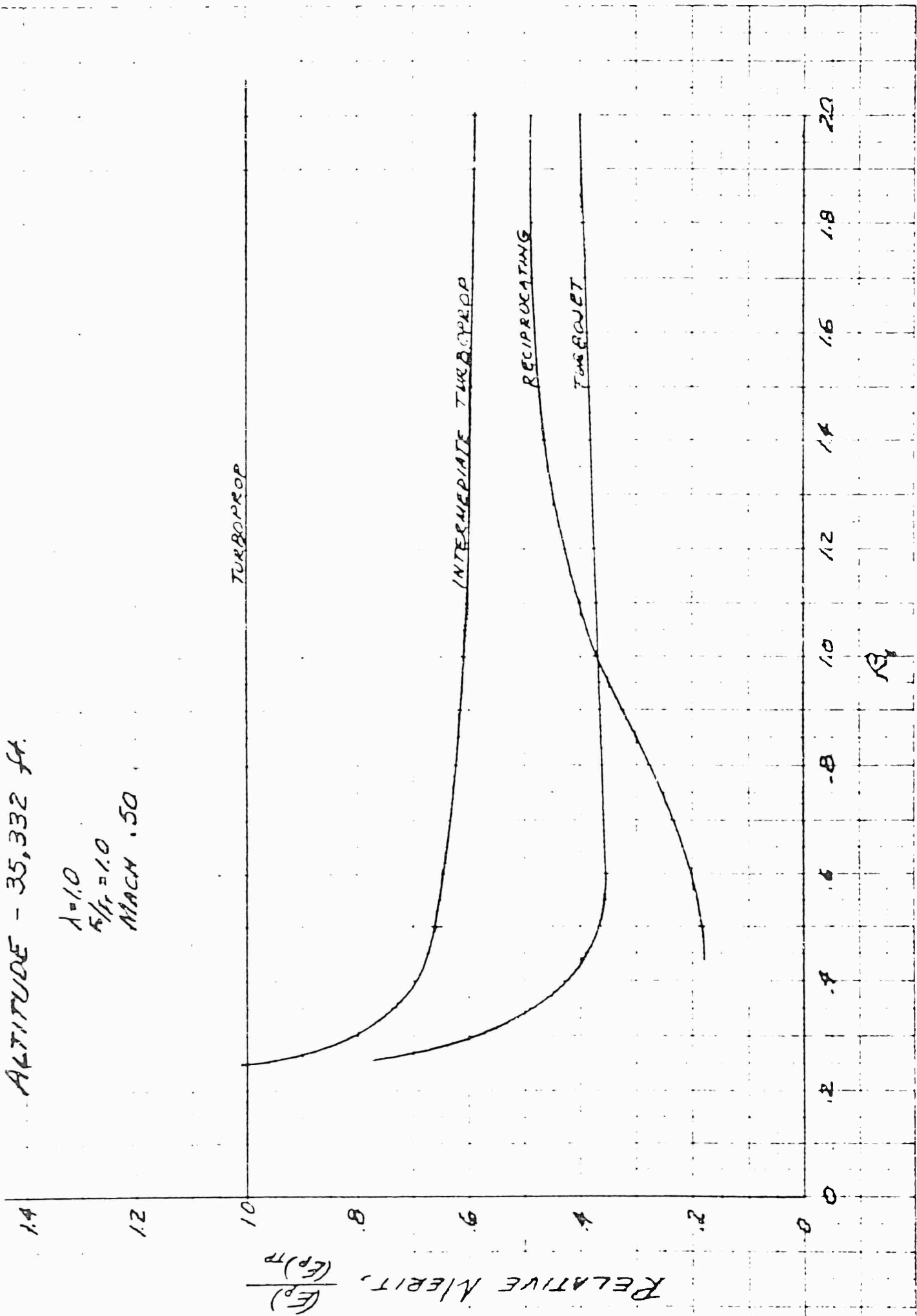


FIGURE 37

ENGINES COMPARISON
 ALTITUDE - 35,332

$A = 1.0$
 $\frac{5}{4} \leq 1.0$
 MACH .75 and 1.00

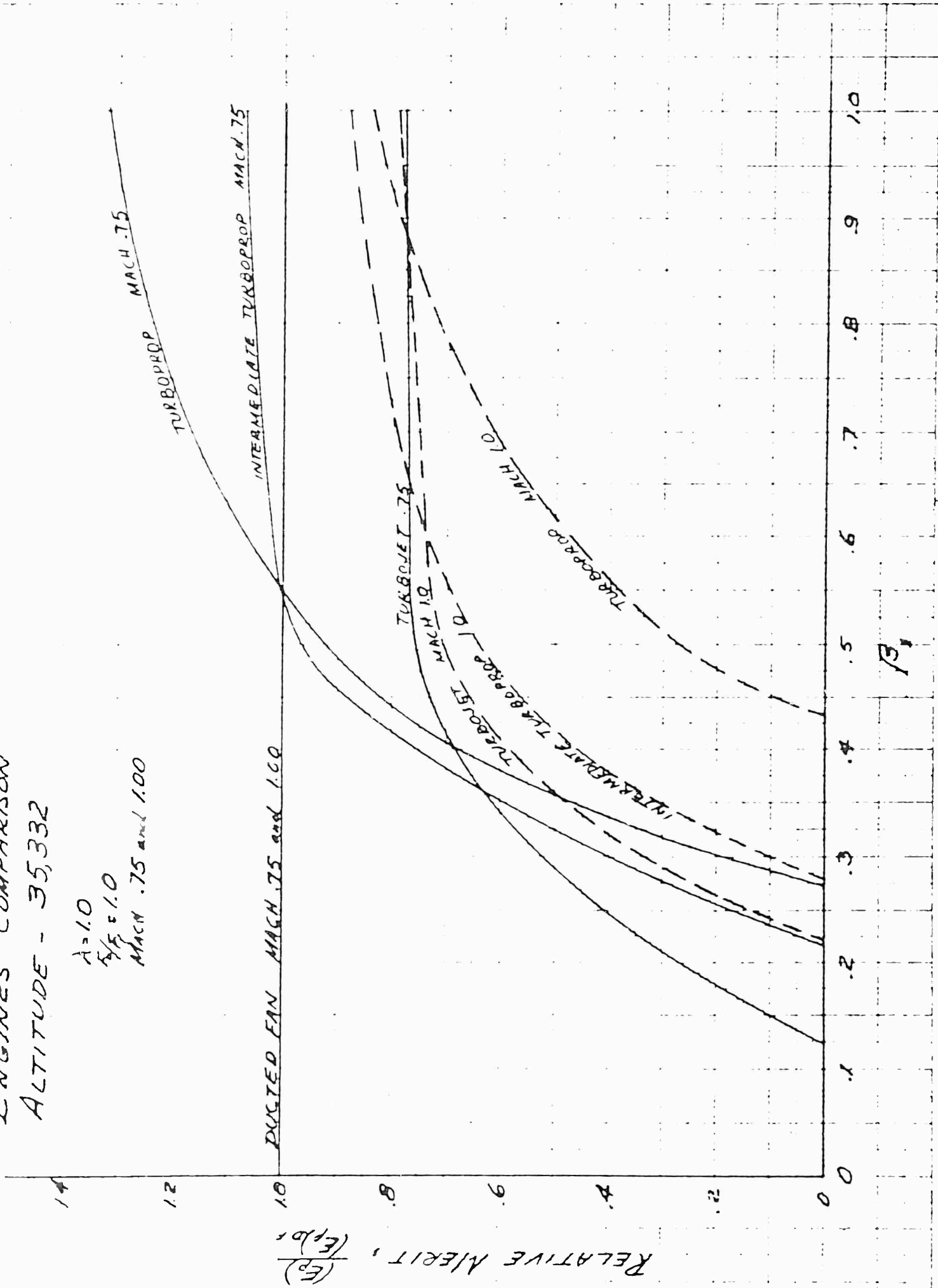


FIGURE 38

ADOPTED VARIATION OF WACELLE DRAG COEFFICIENT WITH MACH NUMBER

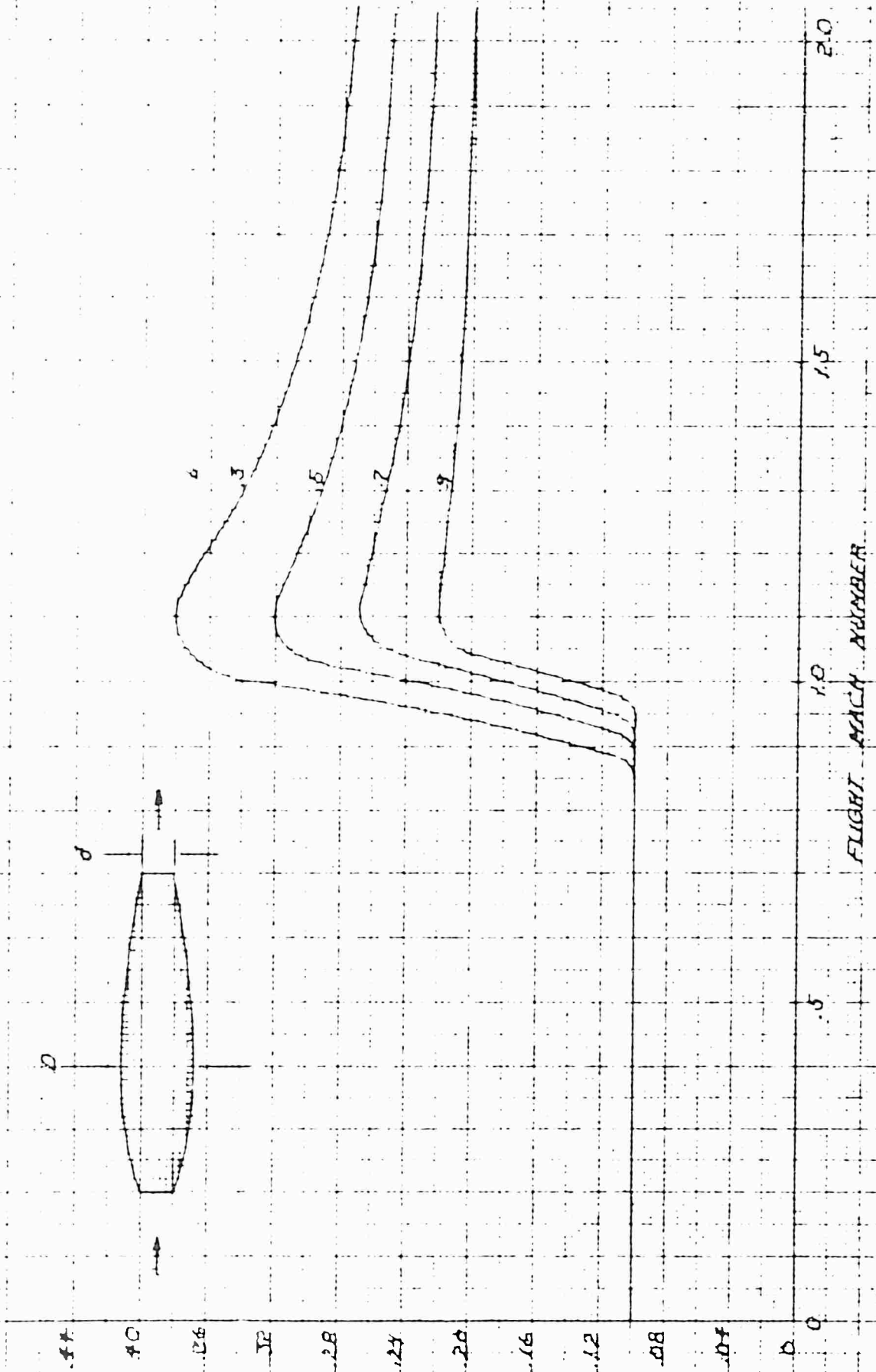


FIGURE 39

EFFECT OF TOWER DIVISION
 BETWEEN
 PROPELLER AND JET
 $P_r = 18.1$
 TURBINE INLET TEMPERATURE 1800 °F
 ALT = 35,332 FT

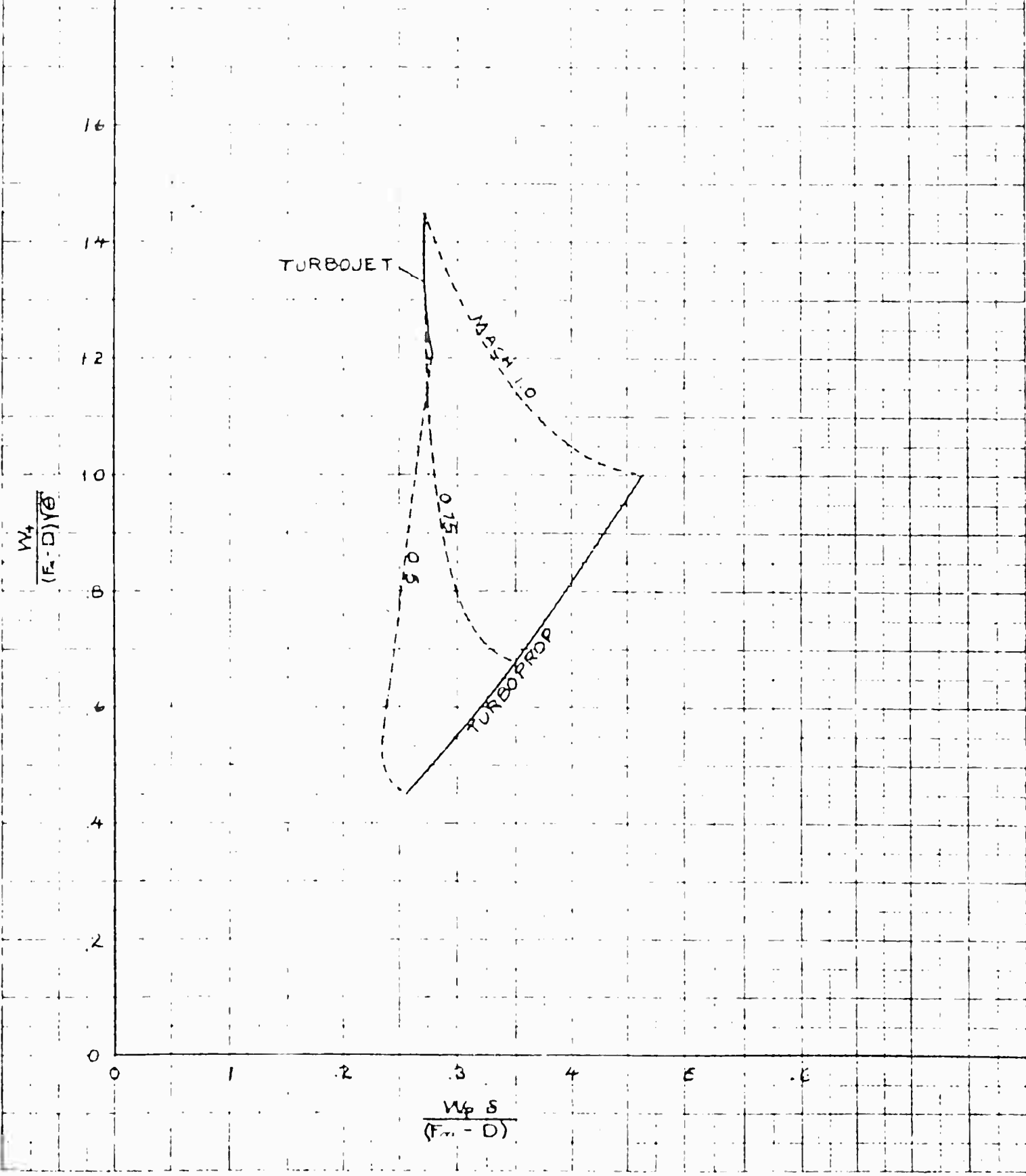


FIGURE 10