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PROJECT SQUID

TECHNICAL MEMORANDUM Pr-19

PRELIMINARY INVESTIGATION OF THE RAM ROCKET AS A POWER PLANT FOR A LONG-RANGE GUIDED MISSILE

by

George S. Sutherland

PRINCETON UNIVERSITY
Princeton, New Jersey

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PROJECT SQUID

A Cooperative Program
of Fundamental Combustion Research
As Related to Jet Propulsion
for the
Office of Naval Research, Department of the Navy
and the
Research and Development Command, Department of the Air Force

A Technical Report on Phase 7 - Performance
Evaluation of the Non-Ideal Ram-Rocket Power Plant
Contract N6ori-105, Task Order III, NR220-038

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NOMENCLATURE

A = maximum missile cross sectional area, square feet

$\mathcal{R}$ = aspect ratio, $\frac{b^2}{S}$

b = wing span, feet

C_D = drag coefficient of RRM, D/qS

C_L = lift coefficient of RRM, L/qS

D = drag, pounds

g = acceleration of gravity, feet per second per second

K = $144 \frac{\gamma}{2} S$, a constant

L = lift, pounds

M = Mach number

p_o = ambient static pressure, psia

R = range, nautical miles

RRM = ram rocket missile

S = total wing area including area within fuselage, square feet

SFC = specific fuel consumption, lbs. fuel per hr. per lb. thrust

T = thrust, pounds

t = time, seconds

V = velocity, feet per second

W = instantaneous weight of missile during acceleration, lbs.

W_o = empty weight of missile, lbs.

W_1 = missile weight at start of Breguet trajectory, lbs.

W_p = propellant weight at start of Breguet trajectory, lbs.

W_t = total gross take-off weight, lbs.

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NOMENCLATURE (continued)

α = angle of attack of wing and body of missile, degrees

γ = ratio of specific heats of air

ν = ratio of total fuel weight plus booster propellant weight to total gross weight on take-off

ν_B = ratio of booster weight to total gross take-off weight

ν_L = ratio of fuel weight to total missile weight at beginning of Breguet trajectory

θ = angle of longitudinal axis of missile with horizontal, degrees

Preliminary Investigation of the Ram Rocket as a Power Plant for a Long-Range Guided Missile

I. Summary

Range calculations were made for a long-range missile using a ram rocket as the power plant. These calculations are the first of a series dealing with various applications of the ram rocket. It is felt that the long-range guided missile, while perhaps not the ideal application for the ram rocket, has certain requirements which the ram rocket can uniquely fulfill. For example, the ram rocket can supply the thrust required to accelerate a missile to design conditions while consuming only a part of the propellant weight required by a comparable booster, and can then power the missile during the subsequent mid-course or Breguet trajectory. This dual role, made possible by the wide thrust variation available, is only one example of the usefulness of the ram rocket principle.

The ram rocket power plant was adapted to the structure of the Triton missile, a 2000 mile range ramjet using JP-1 as a fuel. This adaptation procedure then allowed a direct comparison between the ramjet and ram rocket power plants and their effect upon such critical variables as range, gross take-off weight, and booster size. The main body of the report is concerned with describing the assumptions that were required in adapting the ram rocket to the Triton configuration.

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The fuel used in the ram rocket described herein is methylacetylene. It should be noted, however, that due to present high cost of methylacetylene, it is hoped that cheaper fuels with similar characteristics will be found. Meanwhile, methylacetylene is used in these calculations to illustrate the type of fuel that is theoretically desirable, and, at the same time, actually available.

A summary of the results of these calculations appears in Table III. These figures indicate that the ram rocket can obtain the same range as the rocket-boosted ramjet with a lower total take-off weight, as well as a reduction in the required booster size of the order of 80%. These savings may be directly attributed to the lower specific fuel consumption of the ram rocket during the acceleration phase as compared to the specific fuel consumption of the booster rocket of the ramjet missile.

Due to the unique dependency of the ram rocket performance upon altitude and flight speed during the acceleration period, it is suggested that further investigations be made into the problems of establishing an optimum acceleration trajectory for this type of power plant.

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II Introduction

It is the purpose of this report to investigate the application of the ram rocket as a power plant for a long-range guided missile.

The basic ram rocket, as described in Reference (5), is an air-breathing jet propulsion device consisting of a diffuser, combustion chamber, and exhaust nozzle. Fuel for the combustion chamber is provided by the exhaust gases from one or more rockets mounted at the exit of the diffuser. Figure (1) is a schematic representation of a ram rocket with a constant area combustion chamber.

The performance of the ram rocket naturally depends upon the chemical energy available in the rocket exhaust. For this reason, as well as the reduction in weight made possible by the elimination of an oxidizer and its attendant tanks, pumps, and piping, a monopropellant system is generally more desirable than a bipropellant system.

An acetylene-type monopropellant with a high negative heat of formation is ideally suited for use in the ram rocket, and hence methylacetylene, C_3H_4, was used in the calculations described in this report. With a specific impulse approaching 200 seconds at a chamber pressure of 300 psia, the decomposition products of methylacetylene have a residual chemical energy available for combustion on the order of 18,000 Btu per pound, which compares favorably with the 19,000 Btu per pound available in a ramjet using gasoline as a fuel.

At supersonic speeds, therefore, it may be expected that specific fuel consumption of the ram rocket will be roughly equal to that of a ramjet. Also, due to the nature of the fuel injection process, there is a possibility of obtaining a combustion efficiency superior to that of a ramjet. This variation in combustion efficiency and other basic differences between the ram rocket and the ramjet will be discussed further in Section IV.

Assuming for the moment that at some cruise Mach number the performance of the ram rocket and the ramjet power plants are identical, we see that, for a given range, to obtain any decrease in the take-off gross weight, it is necessary that the fuel consumed in reaching the design conditions at the start of the cruise trajectory be minimized.

To establish the importance of reducing the fuel consumption during the acceleration portion of the flight path, it is convenient to use a modified form of the simple Breguet range equation. It has been shown in Reference (1) that for an air-breathing engine such as the ramjet, the Breguet trajectory is the optimum cruising flight path. As will be shown in Section VI, this modified range equation may be written:

$$R = .592 \frac{V}{SFC}\left(\frac{L}{D}\right)_{max} \ln \frac{1}{1 - \nu_c} \qquad (1)$$

where R = range in nautical miles

V = cruising velocity, feet per second

SFC = specific fuel consumption, lb. fuel per hour per lb. thrust

$(\frac{L}{D})_{max}$ = maximum lift to drag ratio at cruise velocity

ν_c = ratio of propellant weight to total weight of missile at the beginning of the cruise trajectory.

Assuming that the missiles under consideration are aerodynamically identical, and that the power plants have equal specific fuel consumptions at the cruise velocity, we may now substitute some representative values in Equation (1). For example, let us assume that the following figures are applicable to a long-range missile:

Cruise Mach number = 3.0

Cruise velocity = 2924 feet per second

Specific fuel consumption = 1.8 lbs./lb-hr.

$(\frac{L}{D})_{max}$ = 4.5

With these values, Equation (1) becomes:

$$R = 4328 \ln \frac{1}{1 - \nu_c}$$

Suppose now that the empty weight of all missiles is 8,000 lbs., and consider the effect of a variation of $_c$ on the range.

TABLE I

W_t	W_e	ν	W_p	ν_c	$\frac{1}{1-\nu_c}$	$\ln \frac{1}{1-\nu_c}$	R
24,000	8,000	.667	12,000	.600	2.50	.9163	3970
24,000	8,000	.667	18,000	.500	2.00	.6931	3000
24,000	8,000	.667	4,000	.333	1.50	.4055	1758
24,000	8,000	.667	2,000	.200	1.25	.2225	965

As one would expect, the effect of the logarithmic term in the range equation is greatest for low values of ν_c. Assuming a total take-off weight of 24,000 pounds, an increase in ν_c from .333 to .500, which represents a 33% decrease in the fuel required to reach the design flight conditions at the start of cruise, will produce an increase of 70% in the cruise range.

From this simple comparison, it is evident that a power-plant that can start its Breguet trajectory with the highest value of ν_c, all other things being equal, is the power plant that will deliver a given payload the greatest distance, or conversely, will deliver a given payload a required distance with the minimum total take-off weight.

These considerations, then, lead us to the problem of finding means to reduce the fuel required to reach the initial cruise operating point. The ramjet, boosted to this point by means of a solid or liquid fuel rocket, is committed to a large total take-off weight if a reasonable value of ν_c is desired. The rocket-boosted ramjet may have boosters on the order of 50% or greater of the total take-off weight, depending on the desired initial cruise altitude and Mach number. It is the purpose of this report, by means of a specific example, to show that the logistical problems of a supersonic missile may be reduced by using the ram rocket power plant, which will permit a large reduction in the total external booster weight required.

The example consists of a comparison between a ramjet

missile and an equivalent ram rocket missile. Obviously, care must be taken in making such a comparison, since the results are largely dependent on the assumptions that form the basis of the calculations.

It was decided that sufficient published information concerning the performance, design, and weight calculations for a ramjet missile such as the Triton was available so that a comparison could be made with a minimum of effort. The Triton is a rocket-boosted ramjet of 30,000 pounds total take-off weight with a 2000 nautical mile range. This missile carries a 3000 pound payload and has a ν_t equal to .484. The complete analysis of the Triton missile appears in Reference (2).

In order to calculate the acceleration trajectory of the ram rocket missile, aerodynamic and powerplant data over the required Mach number range must be available. The next few sections will discuss the assumptions used in obtaining this data.

III Aerodynamic Data

Due to the nature of the calculations for the acceleration trajectory, aerodynamic data for the ram rocket missile is needed over the entire Mach number range encountered. Where the Triton and its booster accelerated to a design Mach number of 2.4 at zero lift, the ram rocket missile, which we shall hereafter designate as RRM, is assumed to vary its angle of attack so that a constant angle of climb may be maintained. Since the aerodynamic data for the Triton was unapplicable, another means had to be found to obtain reasonable data. Theoretical calculations to obtain aerodynamic data involving wing-body combinations are lengthy and of questionable accuracy. It was decided that experimental wind tunnel results were the most reliable sources of information. Therefore, all experimental data available at the time was assembled. It was found that the most information was available for a combination utilizing a delta wing and a cylindrical body with an ogival nose. All the data corresponding to this basic configuration was plotted versus flight Mach number, and a mean curve was drawn through these experimental points. This method thus provided conservative data for a hypothetical missile that represents a practical design proven by experimental tests.

Figures (2) through (6) are the results of this survey, with the shaded areas representing the spread of all the data obtained. All of the data between the Mach number range of 0 to 2.5 are from experimental wind tunnel investigations,

while the data for Mach numbers higher than 2.5 was extrapolated with the aid of theoretical calculations presented in Reference (3). It was necessary to assume a parabolic relation between the lift and drag coefficients to obtain these coefficients at angles of attack other than those for the maximum lift-to-drag ratio.

With the assumption of a supersonic inlet under all operating conditions, no correction for the open nose required by the Triton design was attempted, since in most cases, the form drag of the ducted ogive is approximately equal to the form drag of a corresponding solid ogival nose. In this case, the necessary assumption that these form drags are equal is based on the fact that, for the proper diffuser design, the two-dimensional flow at the diffuser cowl lip produces a pressure distribution that, when integrated over the smaller area of the ducted nose, produces a pressure drag roughly equal to the drag produced by integrating the three-dimensional pressure distribution of the solid ogival nose over its larger area. This assumption is put on a more conservative basis by neglecting the lower friction drag and the increased lift at angles of attack available with the ducted nose.

Two different configurations were considered: one with a wing-to-body area ratio of 20 and an aspect ratio of 4, and the other with a wing-to-body ratio of 10 and an aspect ratio of 2. Using the theoretical method of calculating lift and drag coefficients for supersonic wing-body combinations

indicated in Reference (4), it may be shown that, all other quantities remaining constant, a reduction in the wing-to-body area ratio from 20 to 10 will have a small effect on the aerodynamic coefficients. The relative independence of the aerodynamic coefficients of the wing-body combination on changes in the wing-to-body area ratio is also verified experimentally in Reference (42). Hence the data shown in Figures (2) through (6) was used for both configurations.

Three of the four cases considered have an external auxiliary power duct mounted aft of the center of gravity on the top of the fuselage. The purpose of this duct is discussed in Section V. In order to correct the aerodynamic data for this deviation from the experimental configuration, it was assumed that this auxiliary power duct was at zero angle of attack at all times. The correction then consisted of adding on the minimum drag of the auxiliary duct, obtained from Reference (9), and modifying it by the ratio of the maximum auxiliary duct area to the total wing area. Since the auxiliary duct is relatively small, the error introduced by the assumption of zero angle of attack is negligible. The auxiliary duct is assumed to be mounted on a sufficient distance from the wings and body so that the interference effects may be neglected. In Reference (43), it is shown that the assumption of negligible interference effects is conservative, since proper placement of the auxiliary duct can reduce the axial drag force by as much as 30% of the sum of the drag forces of the individual components.

It should also be mentioned that the tail area that would be required in a more detailed aerodynamic design is assumed to be part of the total wing area. This approximation is necessitated by the fact that the majority of the models tested in the wind tunnels had no tail assemblies. However, as noted previously in this section, a slight change in wing-to-body area ratio has little effect on the aerodynamic coefficients. Therefore, assuming that the mean effective angle of attack of the horizontal tail surface is approximately equal to that of the main wing, and that the form and friction drag of the vertical tail surface is small with respect to that of the rest of the wing-body combination, we see that the error due to this compromise is within the limits of accuracy of the aerodynamic data itself.

It is essential that conservative aerodynamic data be used in the step-by-step calculation of the acceleration trajectory. Even though the procedure described above utilizes data that does not strictly correspond to the Triton configuration, it is an easy method of obtaining accurate data applicable to a supersonic missile that is representative of present-day design. The radical departure of the RMI from the rocket-boost type of trajectory necessitates this approximation.

IV Power Plant Data

In Reference (5), a complete discussion of the theoretical development followed to produce the performance figures for the ram rocket is presented. This performance is discussed in terms of two parameters: specific fuel consumption and thrust per unit maximum cross sectional area. The data covers altitudes up to 100,000 feet for duct mixture ratios between 15 and 30. For the acceleration phase of the flight, however, it was necessary to extrapolate this data down to values of the duct mixture ratio on the order of 2 and 3 to obtain the thrust necessary for acceleration.

A brief renumeration of the assumptions upon which the theoretical performance data is based will perhaps be helpful. First, it was assumed that the diffuser operation was on-design at all flight speeds. In addition, a constant burner inlet Mach number was maintained by the use of a variable geometry diffuser and exhaust nozzle. Secondly, the effects of friction and heat transfer are negligible. Finally, it was assumed that the influence of the variation in altitude and angle of attack on the combustion efficiency is also negligible. This assumption is justified by the fact that the fuel is introduced into the combustion chamber as a hot, high velocity gas stream composed mostly of hydrogen and carbon. A combustion chamber length of about 6 diameters was used in these calculations, though experimental investigations of this problem indicate that in practice artificial means must be employed to obtain equilibrium over the mixing length assumed. Obviously, the

foregoing assumptions are rather optimistic, but off-design trajectory calculations are prohibitively laborious. Thus we must accept the results as those stemming from the use of an optimum power plant that can be approached in practice with the use of variable geometry equipment.

The primary advantage of the ram rocket is its ability to deliver a varying thrust, with a similarly varying specific fuel consumption. To accomplish this, a variable rocket mass flow must be available. To do this in practice will necessitate the use of a battery of rockets in the main duct, with perhaps the additional requirement of a smaller, auxiliary duct for cruising. The maximum rocket flow required is governed by the air flow through the duct at low altitudes and low duct mixture ratios. The minimum rocket mass flow is fixed by the mean cruising condition at high altitudes, where the criterion of constant velocity will demand a thrust which is a small part of the thrust required during acceleration. The auxiliary duct and multiple rockets in the main duct provide the means of obtaining this thrust variation without exceeding reasonable combustion mixture ratios.

In summary then, it is expected that the theoretical ram rocket will evidence certain advantages over the ramjet as a power plant for a supersonic long-range guided missile. First, the elimination of much of the sensitivity of the combustion process to changes in altitude and operating conditions will be the result of the unique method of fuel injection. The fuel consumption of the ram rocket during accelera-

tion is substantially less than a solid or liquid fuel rocket boost system for a ramjet, and hence, for the same total take-off weight, the ram rocket will have a higher value of γ_c. The size of the booster and its attendant logistic problems may also be reduced to a minimum by the use of the self-accelerating ram rocket power plant. This reduction in the required booster size is of primary importance for certain applications such as shipboard launchings.

V Adaptation

The next problem that should be mentioned is the rather arbitrary task of adapting the ram rocket power plant to the Triton missile. Since complete stress, weight, and layout calculations have been made for the Triton configuration, a minimum number of changes is desirable. The most drastic alteration was necessitated by the addition of six diameters of combustion chamber length to allow for an adequate mixing length. From the design analysis in Reference (2), an estimate of the weight per foot of the entire cross section of the missile can be made at a station that includes the combustion chamber. This figure may then be multiplied by the necessary length increase, and the result will be approximately equal to the weight that must be added to the structural weight to compensate for the missile length increase. The flame holders and fuel injectors of the ramjet engine will be replaced by a cluster of rockets and their supports. It is assumed that this change will have no effect on the overall body weight.

Using Reference (6), it was possible to reevaluate the wing weights, basing the calculation on a wing of triangular planform and 3% thickness. As would be expected, the smaller aspect ratio and area of one of the two wings considered allowed a decrease in the structural weight, whereas the larger wing required an increase in the structural weight over the figure used for the original Triton configuration. Other minor changes included a weight addition to allow for the variable diffuser

and exhaust nozzle geometry. Since the rockets in the main power duct are inoperative during the cruise trajectory, the three ram rocket missiles with auxiliary power ducts would be under a handicap due to the increase in drag caused by the cold flow through the main power duct. One method of circumventing this situation would be to jettison the diffuser cowl at the start of the Breguet flight path, and, with the aid of the variable diffuser equipment, position the spike of the diffuser so that it closes off the entrance to the main power duct. The resulting configuration is shown at the bottom of Figure (8).

The auxiliary duct weight was estimated from Reference (7) on the basis of an average burner diameter of 2.5 feet. In all cases, the sum of main burner duct area and auxiliary duct burner area was kept constant at 16.5 square feet. This figure corresponds to the maximum cross sectional area of the missile body, and, for the case of zero auxiliary duct area, would represent a combustion chamber occupying the entire cross section of the missile body. However, as mentioned previously, it was convenient to have an auxiliary duct available to satisfy the cruising condition of constant velocity. The area required for this auxiliary duct depends on the altitude and speed of the cruise trajectory, and will vary for the four cases considered. Since the variation is small, it was assumed that the effect on the body weight was negligible. This approximation was justified by the fact that the sum of the areas of the two

burners was kept constant, and hence the sum of the weights of the two ducts will also be roughly constant. This approximation also allowed a great simplification in the trajectory calculations.

Another alteration stemmed from the fact that the RRM is essentially a single-stage missile. All the fuel required could not be carried internally without a major change in the Triton dimensions, hence an external, disposable fuel tank was added to the RRM. This auxiliary was assumed to fit close to the belly of the missile so that its influence on the aerodynamic coefficients was negligible. About 46% of the fuel is carried in these tanks, and is burned within the first four minutes of flight.

Finally, it should be mentioned that due to the requirements of launching from a stationary position it is necessary to have a small solid-propellant booster for take-off assistance. The booster rocket was selected from Reference (7), and has an unrestricted burning time of four seconds while delivering almost 50,000 pounds of thrust. Three of these units were used, and, at the launching angle assumed, the total impulse provided by the three units was close to the required minimum. The total booster weight for the RRM was approximately one fifth of the Triton booster weight, and the ν_b was only .140 . Further reduction in the booster size would require the use of a take-off ramp.

A brief summary of the calculations outlined above appears in Appendix A.

VI Trajectory Analysis

This section contains a brief review of the simple equations used to calculate the acceleration and cruise trajectories.

First, during the acceleration part of the flight path, it was necessary to use a step-by-step integration of the equations of motion. Considering the missile as a point mass, and restricting its movement to a vertical plane, the equations of motion in the tangential and normal directions to the flight path may be written:

$$\frac{W}{g}\frac{dV}{dt} = T\cos\alpha - D - W\sin\theta \qquad (1)$$

and

$$\frac{W}{g}V\frac{d\theta}{dt} = T\sin\alpha + L - W\cos\theta \qquad (2)$$

where V= velocity of missile in feet per second

W= weight of missile, pounds

g= acceleration of gravity, feet per second per second

T= thrust, pounds

D= drag, pounds

L= lift, pounds

α= angle of attack of missile and wing

θ= angle of missile with respect to the horizontal which is positive when missile is climbing.

Writing the lift and drag forces in coefficient form, and assuming the angle of attack α to be small, Equations (1) and (2) become:

$$\frac{dV}{dt} = \frac{g}{W}\left[T - 144 C_D \frac{\gamma}{2} p_o M^2 S - W\sin\theta\right] \qquad (3)$$

$$\frac{d\theta}{dt} = \frac{g}{VW}\left[T\sin\alpha + 144 C_L \frac{\gamma}{2} p_o M^2 S - W\cos\theta\right] \qquad (4)$$

where M = Mach number

S = total wing area, square feet

p_o = ambient static pressure, psia

γ = ratio of specific heats of air

C_L = lift coefficient of wing-body combination

C_D = drag coefficient of wing-body combination

For use in the stepwise integration, the equations are written in an incremental form, where Δv and $\Delta\theta$ are calculated for a given increment of time, Δt. With this modification, Equations (3) and (4) may be written:

$$\Delta v = \frac{g\Delta t}{W}\left[T - K C_D p_o M^2 - W\sin\theta\right] \qquad (5)$$

$$\Delta\theta = \frac{g\Delta t}{VW}\left[T\sin\alpha + K C_L p_o M^2 - W\cos\theta\right] \qquad (6)$$

where $K = 144 \frac{\gamma}{2} S$, a constant.

Finally, the familiar Breguet range equation is used to calculate the cruising range. In Reference (1), it was shown that a constant velocity cruise path was optimum for a ramjet. Obviously this criterion may be extended to cover the case of the ram rocket. A brief derivation of the Breguet equation in the form used in these calculations may be of interest.

Since the change in weight of the missile during cruise

is determined by the fuel consumption rate, we may write:

$$\frac{dW}{dt} = - \frac{SFC \cdot T}{3600} \qquad (7)$$

and, applying the condition that $\Delta v = 0$, Equation (7) becomes:

$$dt = - \frac{3600}{SFC \cdot D} dW \qquad (8)$$

The range is the integral of the velocity over the flight time, so:

$$\text{Range} = \int_0^t V \, dt \qquad (9)$$

or changing limits,

$$\text{Range} = - \int_{W_1}^{W_e} \frac{3600 \, V}{SFC \cdot D} dW \qquad (10)$$

where W_1 = missile weight at time t= 0, or at beginning of cruise, and W_e = empty weight of missile at time t.

With the condition that $\theta = \Delta\theta = 0$, Equation (6) supplies the relation:

$$L = W - T \sin \alpha \qquad (11)$$

The value of T and α required to satisfy the constant velocity restriction are relatively small. Hence Equation (11) may be approximated by the relation:

$$L = W \qquad (12)$$

Substituting this identity in Equation (10), we see that:

$$\text{Range} = -\int_{W_1}^{W_e} \frac{3600\ V}{SFC}\ \frac{L}{D}\ \frac{dW}{W} \qquad (13)$$

V and $\frac{L}{D}$ may be held constant by varying the thrust and the angle of attack. The specific fuel consumption will vary with the thrust and altitude, but for purposes of integration, the SFC may be replaced by a mean value that is described in the next section. Integrating Equation (13), we obtain:

$$\text{Range} = V\ \frac{3600}{SFC}\ \frac{L}{D}\ \ln \frac{W_1}{W_e} \qquad (14)$$

However:

$$\frac{W_1}{W_e} = \frac{1}{1 - \nu_c}$$

and therefore the range, in nautical miles, may be written as:

$$R = .592\ \frac{V}{SFC}\ \frac{L}{D}\ \ln \frac{1}{1 - \nu_c} \qquad (15)$$

Equation (5) will provide the value of T to satisfy the condition that $\Delta v = 0$ at any Mach number and altitude. Equation (6) establishes the angle to maintain level flight during cruise, i.e., $\Delta\theta = 0$. However, since for any particular cruising velocity there is a corresponding maximum lift-to-drag ratio, it is required that this value of $(\frac{L}{D})_{max}$ be maintained to ensure maximum range. Therefore, the angle of attack α must be held constant, and Equation (6) then provides the variation in alti-

tude required to maintain $\left(\frac{L}{D}\right)_{max}$ as the weight of the missile decreases. This information enables us to calculate the mean thrust and hence the mean specific fuel consumption for the cruise trajectory to use in the range equation.

Again we see that for maximum range, the quantities V, $\frac{L}{D}$, and ν_c should be maximized, while the SFC should be minimized. A glance at the aerodynamic and power plant performance data shows that for flight Mach numbers above 2, the changes with Mach number in the SFC and maximum lift-to-drag ratio are small, and consequently have much less influence on the range than the variation of the quantity V ln $\frac{1}{1-\nu_c}$. Since ν_c decreases as as the cruise velocity is increased, the quantity V ln $\frac{1}{1-\nu_c}$ should be maximized. For usual values of the overall loading factor ν, the quantity V ln $\frac{1}{1-\nu_c}$ increases with increasing velocity for flight speeds up to and beyond the critical speed determined by aerodynamic heating and the missile materials. It may thus be concluded that, up to the point of prohibitive vehicle weight, the maximum range will usually be obtained by flying at the highest possible velocity commensurate with temperature-stress limitations of the missile construction.

VII Results of Range Calculations

A summary of the results is presented in Table II. As previously mentioned, four cases were calculated for comparison with the Triton ramjet missile. Two values of aspect ratio and wing area were considered, as well as two flight Mach numbers. In addition to having a lower wing weight, the RRM with the smaller wing benefited by its requirement of a lower initial cruise altitude, and consequently, an acceleration through a denser atmosphere. For a given configuration, the effect of increasing the cruise Mach number from 2.5 to 4.0 is clearly shown. It is interesting to note that, contrary to rocket missile and rocket boosted ramjet missile technique, it is beneficial to accelerate in as low an altitude range as is compatible with the Breguet trajectory altitude requirements. This characteristic is a result of the increased margin of thrust over drag, at higher ambient pressures due to the dependency of the thrust per unit cross sectional area of the ram rocket on the pressure level during its combustion cycle.

Further comparison between the Triton missile and the two RRM missiles with a design Mach number of 2.5 reveals that the missiles with the ram rocket power plant achieved approximately the same range as the Triton at essentially the same flight speed, but with almost 20% less total take-off weight and with a reduction in the booster weight of about 80%. In Table II, the cruise ranges of the RRM No. 1 and No. 3 are modified by

employing the value of $(\frac{L}{D})_{max}$ and SFC listed for the Triton. With identical aerodynamic and power plant performance, this comparison serves only to show the importance of minimizing the fuel consumption during the acceleration trajectory and indicates that comparable values of ν_c may be obtained with sizable reductions in take-off and booster weight.

TABLE II

	Triton	RRM #1	RRM #3
Cruise SFC, hr.$^{-1}$	1.92	1.92	1.92
$(\frac{L}{D})_{max}$	4.32	4.32	4.32
ν_o	.496	.415	.413
Range, nautical miles	2044	1769	1791

No attempt was made to optimize the acceleration trajectory of the RRM. It is almost certainly true that some other trajectory other than the constant angle of climb trajectory employed will prove more adaptable to the ram rocket characteristics and will permit even greater reductions in the total take-off weight of the Triton-type missile. Even though the conditions assumed for ram rocket power plant operation represent optimum performance, the results clearly indicate the ability of the ram rocket principle to reduce the specific fuel consumption during the acceleration period and to eliminate the problems of large, bulky boosters.

There are certain practical difficulties to be overcome

before the ram rocket missile becomes a reality. Perhaps the greatest disadvantage is the large variation in duct mixture ratio necessary to satisfy the thrust requirements during both the acceleration and the cruise trajectory. This variation in the duct mixture ratio imposes the condition that the rocket massflow must be controlled by some such method as was suggested earlier in the report, i.e., a battery of rockets with the possibility of using the mass flow from all or part of these rockets. Obviously, the auxiliary power duct, though convenient for the purposes of this report, should be avoided if possible in an actual application.

However, the assumptions and calculations made in this report are generally conservative, and the results outlined in this preliminary investigation certainly establish the desirability of further detailed investigations of the application of the ram rocket as a power plant for a long-range guided missile.

TABLE III

ITEM	TRITON	RRM#1	RRM#2	RRM#3	RRM#4
Total wing area/body area - $\frac{S}{A}$	7.22	20	20	10	10
Aspect Ratio	3.6	4.0	4.0	2.0	2.0
Cruise Mach number	2.4	2.5	4.0	2.5	4.0
Gross take-off weight, lbs.	30,000	24,700	24,700	24,700	24,700
Booster weight, lbs.	14,500	3450	3450	3450	3450
ν_s	.484	.140	.140	.140	.140
Fuel weight, lbs.	7700	12,060	11,500	11,900	11,900
Empty weight(with payload), lbs.	7800	8890	9490	9050	9050
External fuel tanks, lbs.	-	300	300	300	300
Fuel in body, lbs.	7700	6500	6500	6500	6500
Fuel in external tanks, lbs.	-	5560	5000	5400	5400
Gross weight at end of acceleration	15,500	15,186	13,943	15,428	13,939
ν_o	.496	.415	.320	.413	.350
Angle of launching, degrees	46	60	60	60	60
Angle of climb, degrees	zero lift	30	30	30	30
Initial Breguet altitude, feet	57,000	84,000	103,400	61,000	80,500
Final Breguet altitude, feet	70,500	95,000	111,000	72,500	90,000
Maximum $\frac{L}{D}$ during cruise	4.32	4.66	5.20	4.83	6.00
Cruising SFC, $hr.^{-1}$	1.92	1.85	1.75	1.71	1.88
Cruising duct mixture ratio	-	30	26	30	30
Acceleration time, seconds	-	190	290	95	158
Range, nautical miles	2044	2000	2757	2200	3230
Range, acceleration	154	50	107	80	50
Range, Breguet	1890	1950	2650	2180	3180

TABLE III

ITEM	TRITON	RRM#1	RRM#2	RRM#3	RRM#4
Main Power Duct:					
Number of rockets	-	12	10	10	10
Thrust per rocket, lbs.	-	1,000	1,000	1,000	1,000
Maximum mass flow, lbs./sec.	-	60	50	50	50
Area, square feet	11.2	16.5	10.0	10.5	14.00
Auxiliary Power Duct:					
Number of rockets	-	-	10	10	10
Thrust per rocket, lbs.	-	-	200	200	200
Maximum mass flow, lbs./sec.	-	-	10	10	10
Area, square feet	-	0	6.5	6.0	2.5
Booster:					
Burning time, seconds	6	4	4	4	4
Thrust of each unit, lbs.	94,000	49,580	49,580	49,580	49,580
Number of units	4	3	3	3	3
Overall impulse-weight ratio	160	173	173	173	173

REFERENCES AND BIBLIOGRAPHY

(1) Rand Corporation:

Long-Range Surface-to-Surface Rocket and Ramjet Missiles. Flight Mechanics. Report Number R-179; May, 1950, (Secret).

(2) Cornell Aeronautical Laboratory, Inc.:

Design of the Triton Long-Range Missile. CAL/CM-617; June, 1950, (Confidential).

(3) Rand Corporation:

Long-Range Surface-to-Surface Rocket and Ramjet Missiles. Aerodynamics. Report Number R-181; May, 1950, (Secret).

(4) Katzen, Elliot D., and Kaattari, George E.:

Drag Interference Between a Pointed Cylindrical Body and Triangular Wings of Various Aspect Ratios at Mach Numbers of 1.50 and 2.02. NACA RM A51C27; May, 1951, (Confidential).

(5) Charyk, J.V., and Sutherland, George S.:

A Performance Study of the Ram Rocket Power Plant and Related Problems. Princeton University Aeronautical Engineering Laboratory Report No. 193; March, 1952, (Restricted).

(6) Cornell Aeronautical Laboratory, Inc.:

Investigation of Weights for Wings and Tails of Supersonic Vehicles. CAL/CM - 384-A; July, 1947, (Confidential).

(7) Cornell Aeronautical Laboratory, Inc.:

Body Weight Estimates for a Family of Long-Range Ramjet Missiles. CAL/CM - 402; August, 1947, (Confidential).

(8) Denison, Frank G.:

Comparative Performance of Solid and Liquid Propellant Systems for Boosters and High-Altitude Rockets. Jet Propulsion Laboratory; October, 1949, (Confidential).

REFERENCES AND BIBLIOGRAPHY (continued)

(9) Weinstein, Maynard I., and Davids, Joseph:

Force and Pressure Characteristics for a Series of Nose Inlets at Mach Numbers from 1.59 to 1.99. III-Conical-Spike All-External-Compression Inlet with Supersonic Cowl Lip. NACA RM E50J30; February, 1951, (Confidential).

(10) Heitmeyer, John C., and Smith, Willard G.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Plane Triangular Wing of Aspect Ratio 2 with NACA 0003-63 Section. NACA RM A50K24a; February, 1951, (Confidential).

(11) Phelps, E.R., and Smith, W.G.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Triangular Wing of Aspect Ratio 4 with NACA 0005-63 Thickness Distribution, Cambered and Twisted for Trapezoidal Span Load Distribution. NACA RM A50K24b; February, 1951, (Confidential).

(12) Heitmeyer, John C., and Stephenson, J.D.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Plane Triangular Wing of Aspect Ratio 4 with NACA 0005-63 Section. NACA RM A50K24, February, 1951, (Confidential).

(13) Heitmeyer, John C.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Plane Triangular Wing of Aspect Ratio 4 with 3-percent-Thick, Biconvex Section. NACA RM A51D30, June 1951, (Confidential).

(14) Smith, D.W., and Heitmeyer, John C.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Plane Triangular Wing of Aspect Ratio 2 with NACA 0005-63 Section. February, 1951, (Confidential).

(15) Hall, C.F., and Keitmeyer, John C.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Twisted and Cambered Triangular Wing of Aspect Ratio 2 with NACA 0003-63 Thickness Distribution. NACA RM A51B01; June, 1951, (Confidential).

(16) Reese, D.E., and Phelps, E.R.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Plane Tapered Wing of Aspect Ratio 3.1 with 3-percent-Thick, Biconvex Section. NACA RM A50K28; January, 1951, (Confidential).

(17) Smith, W.G. and Phelps, E.R.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Triangular Wing of Aspect Ratio 2 with NACA 0005-63 Thickness Distribution, Cambered and Twisted for a Trapezoidal Span Load Distribution. NACA RM A50K27a; February, 1951, (Confidential).

(18) Cornell Aeronautical Laboratory, Inc.:

Design Evaluation of Long-Range Bombardment Missiles. CAL/CM - 491; January, 1948, (Confidential).

(19) Osborne, Robert S.:

A Transonic-Wing Investigation in the Langely 8-foot High-Speed Tunnel at High Subsonic Mach Numbers and at a Mach Number of 1.2 . NACA RM L50H08; October, 1950, (Confidential).

(20) Wood, R.B., and Fleming, F.P.:

A Transonic-Wing Investigation in the Langely 8-foot High-Speed Tunnel at High Subsonic Mach Numbers and at a Mach Number of 1.2 . NACA RM L50J25; January, 1951, (Confidential).

(21) Nelson, Robert L.:

Large Scale Flight Measurements of Zero-Lift Drag at Mach Numbers from 0.76 to 1.5 of a Wing-Body Combination Having a 60° Triangular Wing with NACA 65A003 Sections. NACA RM L50D26; June, 1950, (Confidential).

(22) Cleveland Laboratory Staff:

Performance and Ranges of Application of Various Types of Aircraft Propulsion Systems. NACA TN 1349; August, 1947.

(23) Spreiter, John R.:

Aerodynamic Properties of Slender Wing-Body Combinations at Subsonic, Transonic, and Supersonic Speeds. NACA TN 1662; July, 1948.

(24) Nielson, Jack N., Katzen, Elliott D., and Tang, Kenneth K.:

Lift and Pitching-Moment Interference Between a Pointed Cylindrical Body and Triangular Wings of Various Aspect Ratios at Mach Numbers of 1.50 and 2.02 . NACA RM A50F06; September, 1950, (Confidential).

(25) Donlan, Charles J., Myers, Boyd C., and Mattson, Axel T.:

A Comparison of the Aerodynamic Characteristics at Transonic Speeds of Four Wing-Fuselage Configurations as Determined From Different Test Techniques. NACA RM L50H02; October, 1950, (Confidential).

(26) Allen, Julian H.:

Estimation of the Forces and Moments Acting on Inclined Bodies of Revolution of High Fineness Ratio. NACA RM A9126; November, 1949, (Confidential).

(27) Sutton, Fred B., and Martin, Andrew:

Aerodynamic Characteristics Including Pressure Distributions of a Fuselage and Three Combinations of the Fuselage with Swept-Back Wings at High Subsonic Speeds. NACA RM A50J26a; February, 1951, (Confidential).

(28) Hall, James R., and Sandahl, Carl A.:

Effect of Nose Shape and Wing Thickness Ratio on the Drag at Zero Lift of a Missile Having Triangular Wings and Tails. NACA RM L50C16a; May, 1950, (Confidential).

(29) Schult, Eugene D.:

Comparison of Large-Scale Flight Measurements of Zero-Lift Drag at Mach Numbers from 0.9 to 1.7 of Two Wing-Body Combinations Having Similar 60° Triangular Wings with NACA 65A003 Sections. NACA RM L50I22; October, 1950, (Confidential).

(30) NACA Conference on Supersonic Aerodynamics, August 30-31, 1948, (Confidential).

(31) Johns Hopkins University:

Symposium on Aerodynamics. Bumblebee Report No. 29; December, 1945, (Confidential).

(32) Ellis, Mason C., and Brown, Clinton E.:

Analysis of Supersonic Ramjet Performance and Wind-Tunnel Tests of a Possible Supersonic Ramjet Airplane Model. NACA Acr. No. L5L12; December, 1945, (Confidential).

(33) D'Aiutolo, Charles T., and Mason, Homer P.:

Preliminary Results of the Flight Investigation Between Mach Numbers of 0.80 and 1.36 of a Rocket-Powered Model of a Supersonic Airplane Configuration Having a Tapered Wing with Circular-Arc Sections and 40° Sweepback. NACA RM L50H29a; October, 1950, (Confidential).

(34) Princeton University:

Performance of Aircraft at High Supersonic Speeds. Report Number 153; September, 1949.

(35) Ellis, Mason C., Mazel, Lowell E., and Grigsby, Carl E.:

Supersonic-Tunnel Tests of Two Supersonic Airplane Model Configurations. NACA RM L7J15; December, 1947, (Confidential).

(36) Grigsby, Carl E.:

Investigation at a Mach Number of 1.93 to Determine Lift, Drag, Pitching-Moment, and Average Downwash Characteristics for Several Missile Configurations Having Rectangular Wings and Tails of Various Spans. NACA RM L50I08; October, 1950, (Confidential).

(37) Spahr, Richard J., and Robinson, Robert A.:

Wind Tunnel Investigation at Mach Numbers of 2.0 and 2.9 of Several Configurations of a Supersonic Ramjet Test Vehicle. NACA RM A50C20; August, 1950, (Confidential).

(38) Elliott, Macon C., and Grigsby, Carl E.:

Aerodynamic Investigation at Mach Number 1.92 of a Rectangular Wing and Tail and Body Configuration and Its Components. NACA RM L9L28a; March, 1950, (Confidential).

(39) Heitmeyer, John C., and Peterson, Robert B.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Twisted and Cambered Triangular Wing of Aspect Ratio 2 with NACA 0005-63 Thickness Distribution. NACA RM A52B06; April, 1952, (Confidential).

(40) Heitmeyer, John C.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Plane 45° Swept-Back Wing of Aspect Ratio 3, Taper Ratio 0.4 with 3-percent-Thick, Biconvex Section. NACA RM A51H10; September, 1951, (Confidential).

(41) Heitmeyer, John C.:

Lift, Drag, and Pitching Moment of Low-Aspect-Ratio Wings at Subsonic and Supersonic Speeds. Plane Triangular Wing of Aspect Ratio 3 with NACA 0003-63 Section. NACA RM A51H02; September, 1951, (Confidential).

(42) Hopkins, Edward J., and Carol, Hubert C.:

Experimental and Theoretical Study of the Effects of Body Size on the Aerodynamic Characteristics of of an Aspect-Ratio 3.0 Wing-Body Combination. NACA RM A51G24; October, 1951, (Confidential).

(43) Kremzier, Phil J., and Dryer, Murray:

Aerodynamic Interference Effects on Normal and Axial Force Coefficients of Several Engine-Strut-Body Configurations at Mach Numbers of 1.8 and 2.0. NACA RM A52B21; April, 1952, (Confidential).

Appendix A Summary of Design Changes

(1) Change in Body Design

For an average combustion chamber area of 11.5 square feet, the combustion chamber diameter is 3.83 feet. To obtain a mixing length of 6 diameters, the combustion chamber must be 23 feet long.

It is necessary to add an extra internal fuel tank to provide fuel volume for 6500 lbs. of fuel. This extra fuel tank also requires an additional increase in the body length. Methylacetylene has a specific gravity of .678, hence the capacity of the forward fuel tank is

$$3040 \times \frac{.678}{.800} = 2580 \text{ lbs.}$$

Therefore, the extra internal tank must hold

$$6500 - 2580 = 3920 \text{ lbs.}$$

This weight of fuel requires a tank volume of

$$\frac{3930}{42.3} = 92.6 \text{ cubic feet.}$$

A tank 46.6 inches in diameter and 7.80 feet long will provide

$$11.88 \times 7.80 = 92.6 \text{ cu. ft.}$$

Therefore, the total length required aft of station 300 on the Triton fuselage is

$$7.80 + 23.00 = 30.80 \text{ feet.}$$

Of this distance, 18.80 is already available in the Triton fuselage, and hence the total increase in body length required is 12 feet. The aft fuel tanks which are 9.57 feet long and weigh 500 pounds, are replaced by an outer skin and combustion chamber structure.

Assuming that the additional internal fuel tank is constructed in the same manner as the forward fuel tank in the Triton design, the following weight estimates may be made:

Triton fuel tank = 200 lbs.

effective length of the Triton fuel tank = 6.67 ft.

weight per foot = 30 lbs./ft.

Therefore, the additional internal fuel tank weighs

$30 \times 7.7 = 231$ lbs.

For the outer shell, we have the following estimates:

outer shell at combustion chamber = 160 lbs.

length of outer shell = 7.59 ft.

weight per foot = 21.1 lbs./ft.

Therefore, the additional weight of the outer shell is

$21.57 \times 21.1 = 455$ lbs.

Since 7.59 feet of combustion chamber space are available in the original Triton configuration, the additional combustion chamber structure required is

$23.00 - 7.59 = 15.41$ feet.

From Reference (7), the combustion chamber structure of a 6 foot diameter burner weighs 85.6 pounds per foot. Assuming that the cross-sectional weight of the burner construction varies as the square of the diameter, then, for a 3.83 foot diameter burner, we find that the weight is

$$\frac{(3.83)^2}{(6)^2} \times 85.6 = 35 \text{ lbs. per ft.}$$

For 15.4 feet of additional burner length, the weight increase is

$$35 \times 15.4 = 540 \text{ lbs.}$$

The original 60 lbs. of booster fittings are reduced to 20 lbs. since the booster weight is only one fifth of the original booster weight. Finally, an arbitrary addition of 100 lbs. representing the variable diffuser geometry was assumed, as well as 50 lbs. for a variable exhaust nozzle.

(2) Auxiliary Duct

The sum of the main burner cross-sectional area and the auxiliary duct cross-sectional area must remain constant at 16.5 square feet, and therefore, for a basic burner area of 11.5 square feet, the basic auxiliary duct area is 5.0 square feet. With an overall power duct length-to-diameter ratio of 8, the burner length must be 15 feet to satisfy the criterion of a burner L/D of approximately 6. The linear weight estimates can be written as

linear weight of combustion chamber $= \frac{(2.5)^2}{(6)^2} \times 85.6$

$= 25$ lbs./ft.

linear weight of outer shell $= \frac{(2.6)^2}{(3.83)^2} \times 21.1$

$= 10$ lbs./ft.

diffuser structure $= \frac{(2.6)^2}{(3.83)^2} \times \frac{(160)}{(4.3)}$

$= 17$ lbs./ft.

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Using the above linear weight estimates, the weight of the auxiliary duct is distributed as follows:

outer shell = 150 lbs.

combustion chamber = 225 lbs.

diffuser and spike = 85 lbs.

rockets and supports = 100 lbs.

strut = 40 lbs.

Total weight of auxiliary duct = 600 lbs.

(3) Wing Weights

Using the method found in Reference (6), the parameter of interest is the quantity

$$P = \frac{f_b \, (t/c)}{\sigma \; C_r}$$

where f_b = stress of outer fiber in bending, psi

t/c = thickness ratio of wing

σ = planform parameter

$\sigma = \frac{A^2}{16}$ for a delta wing

C_r = root chord, inches

Case (a):

$S/A = 10$

$t/c = .03$

$A = 2.0$

total wing area = 165 sq. ft.

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external wing area = 92 sq. ft.

root chord = 13.55 ft.

f_b = 25,000 psi (Dural)

σ = .250

wing loading = 268 lbs./sq. ft.

The value of P for case (a) is 18.5, and the resulting value of the wing weight per square foot of external wing area is 2.4 lbs. per square foot. The wing weight can then be written as

$$W_w = 2.4 \times 92$$
$$= 221 \text{ lbs.}$$

Case (b):

S/A = 20

t/c = .05

AR = 4.0

total wing area = 330 square ft.

external wing area = 251 square ft.

root chord = 15.85 ft.

f_b = 25,000 psi (Dural)

= 1.0

wing loading = 100 lbs. per square ft.

Therefore, the value of P for case (b) is 6.58, and the wing weight per square foot is 2.63 lbs. per square foot. The wing weight then becomes

$$W_w = 2.63 \times 251$$
$$= 660 \text{ lbs.}$$

A summary of the weight changes entailed in adapting the ram rocket power plant to the Triton configuration is outlined below.

Item - pounds	RRM#1	RRM#2	RRM#3	RRM#4
Basic empty weight of Triton (with payload)	7800	7800	7800	7800
Additional internal fuel tank	230	230	230	230
Outer shell	455	455	455	455
Combustion chamber	540	540	540	540
Variable geometry equipment	150	150	150	150
Auxiliary duct	-	600	600	600
Wing (external)	660	660	220	220
Total additions	9836	10436	9997	9997
Triton external wing	405	405	405	405
Aft fuel tank	500	500	500	500
Booster fittings	40	40	40	40
Total removals	945	945	945	945
Empty weight of RRM missile	8890	9490	9050	9050

It should also be mentioned that since the weight of the external, disposable fuel tank is dependent on materials, shape, and construction, etc., an arbitrary weight of 300 lbs. was assumed for this tank. This figure appears to be consistent with the purpose and size of this external tank.

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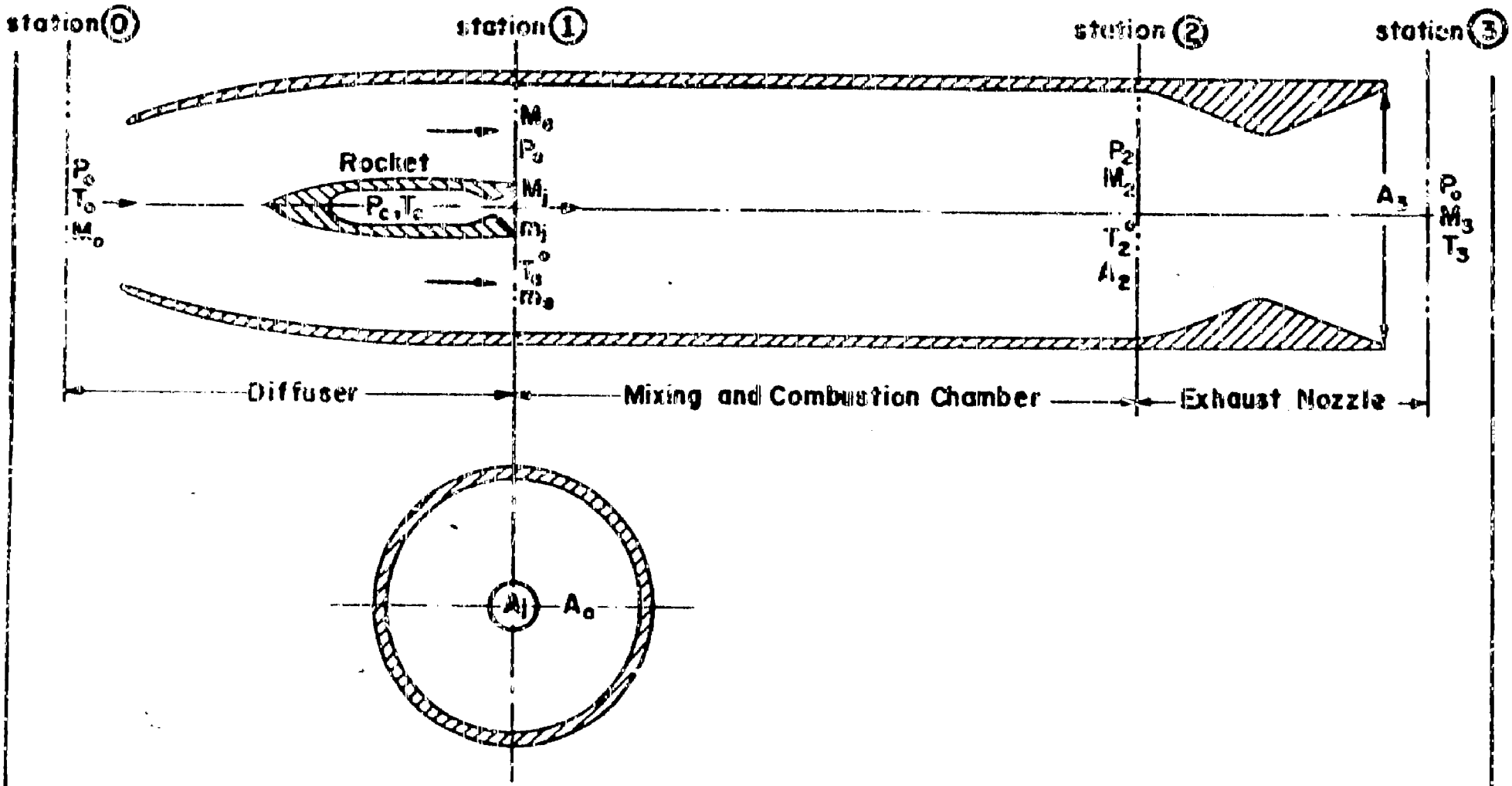

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Fig. 1.

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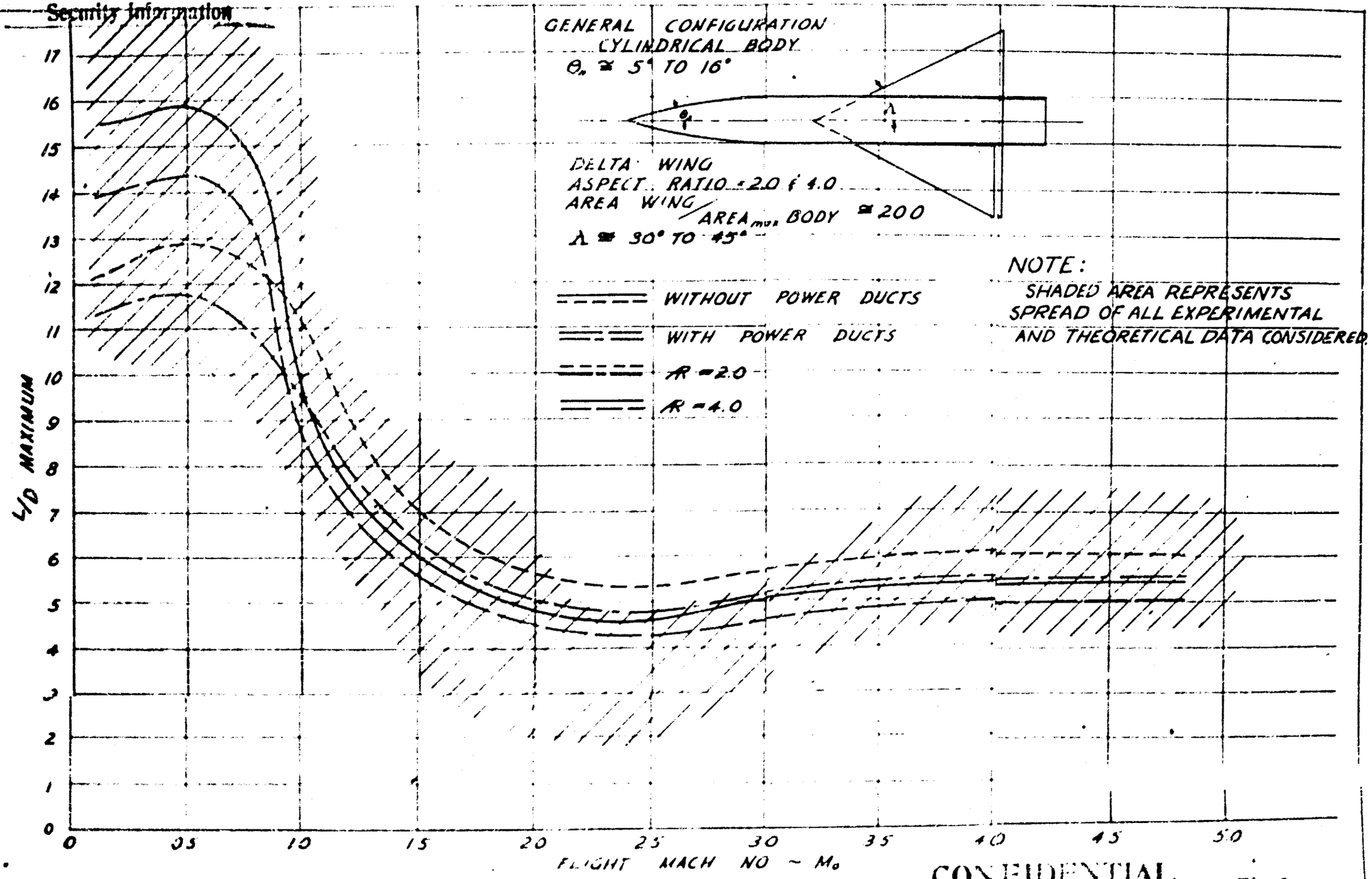

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Fig. 2.

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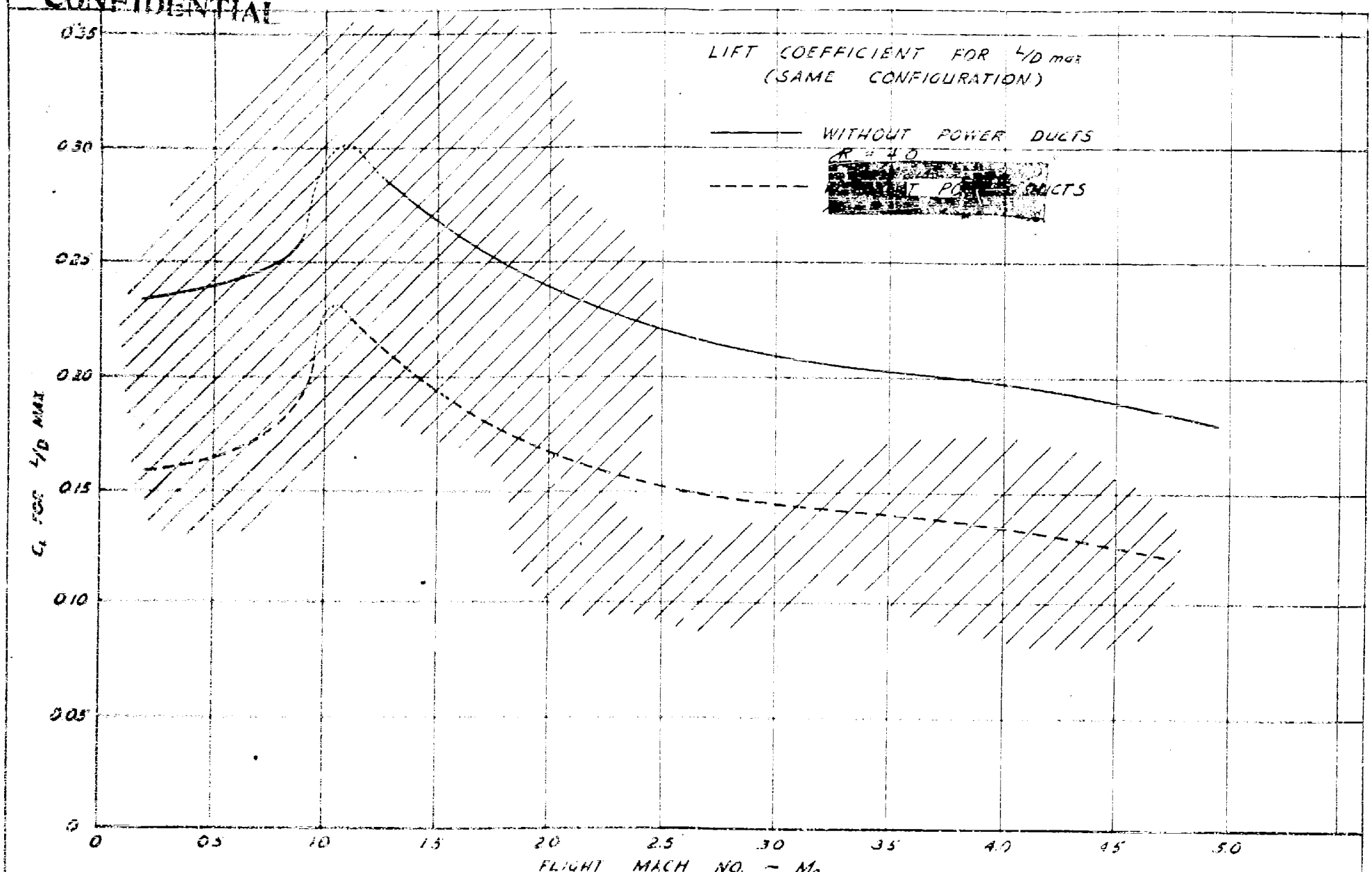

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Fig.3.

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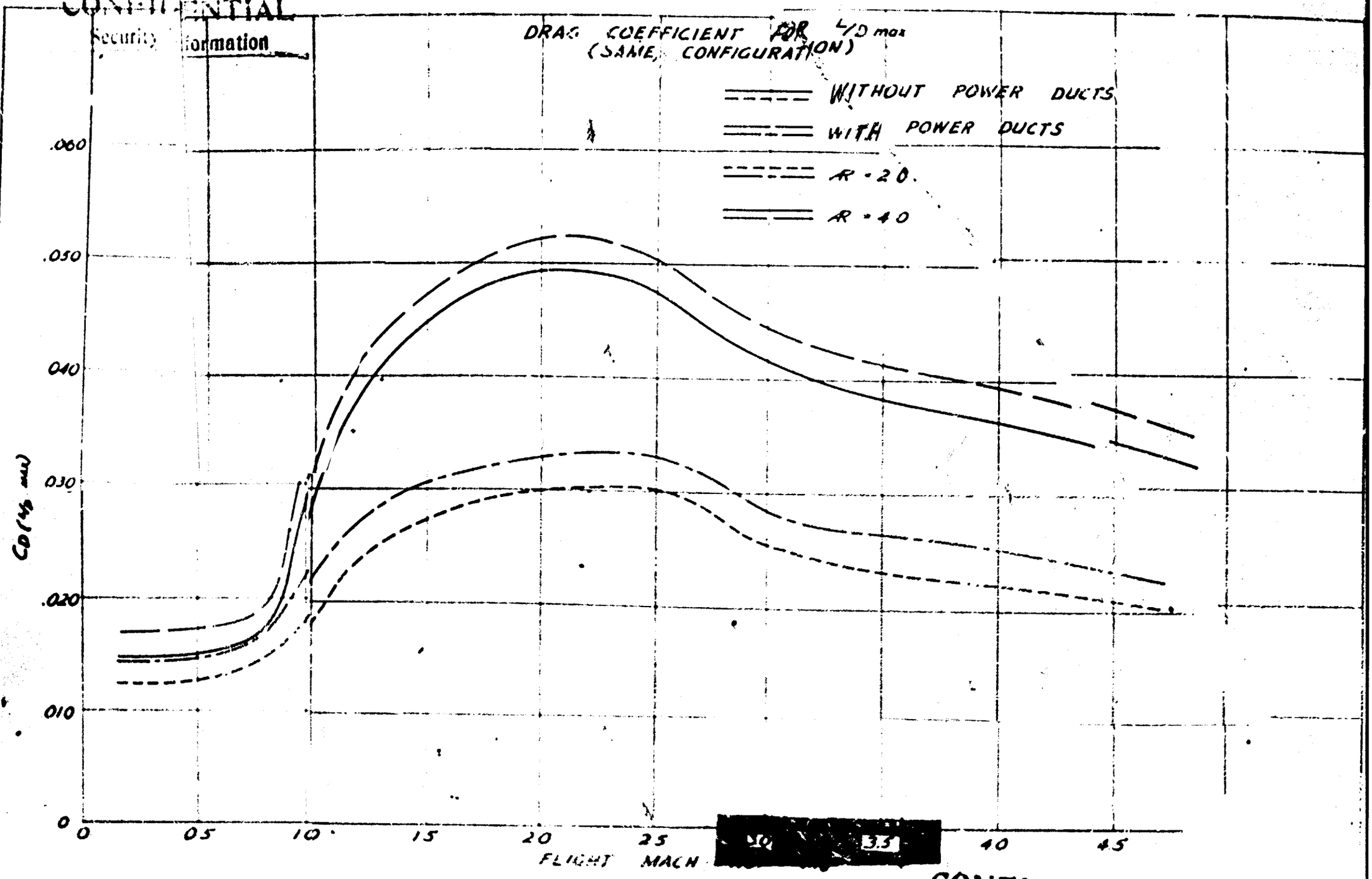

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Fig 4

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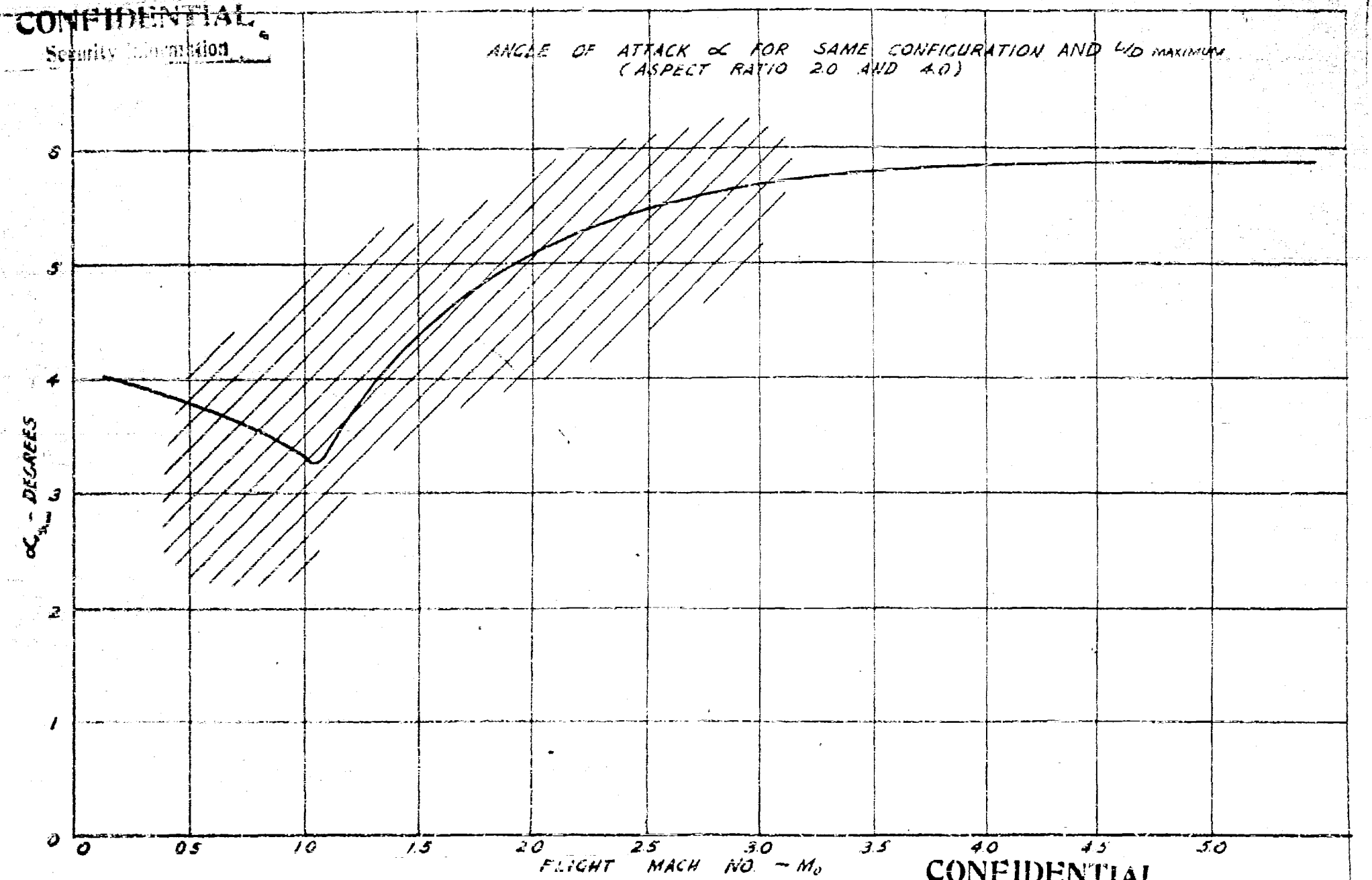

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Fig.5.

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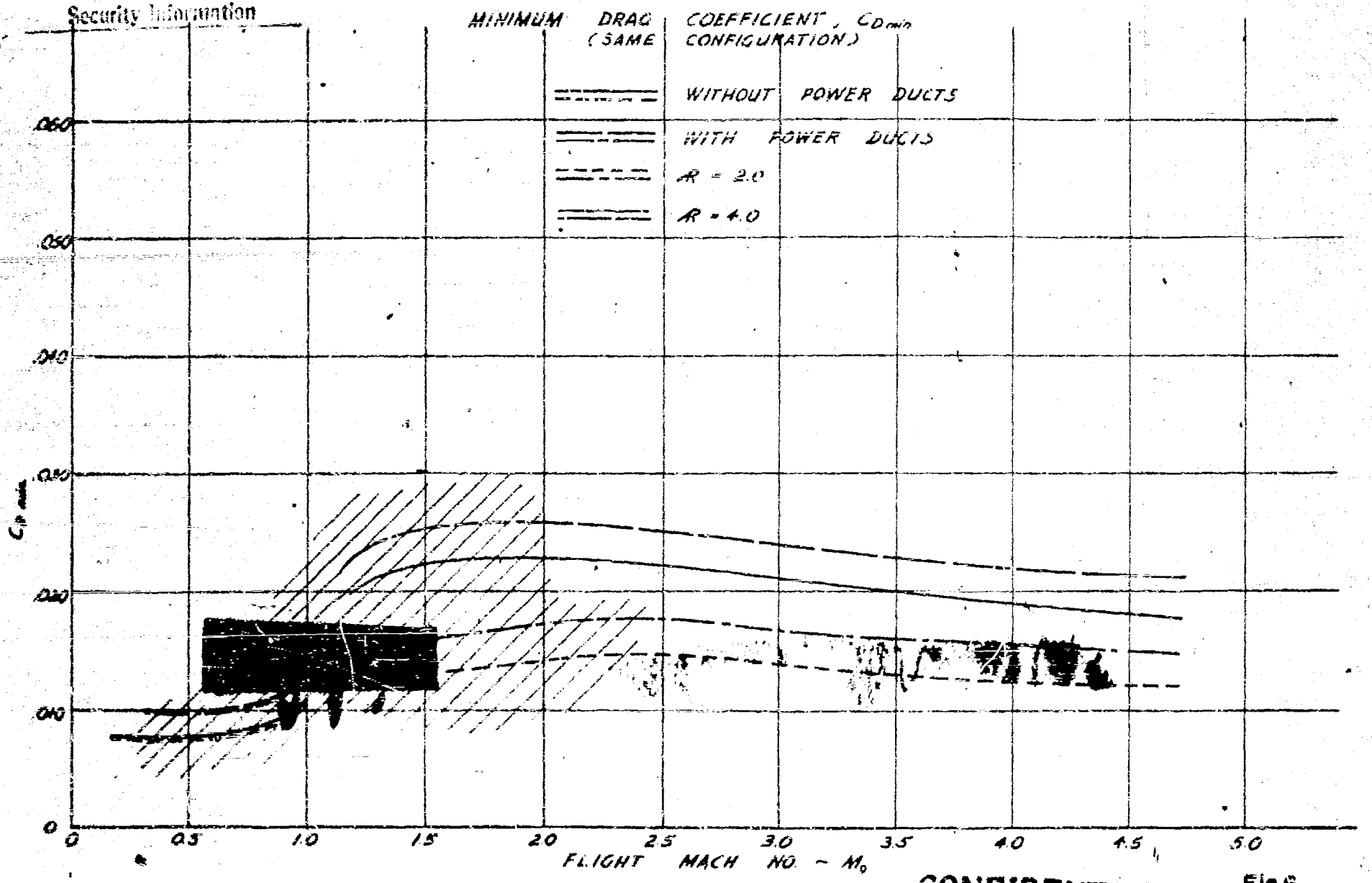

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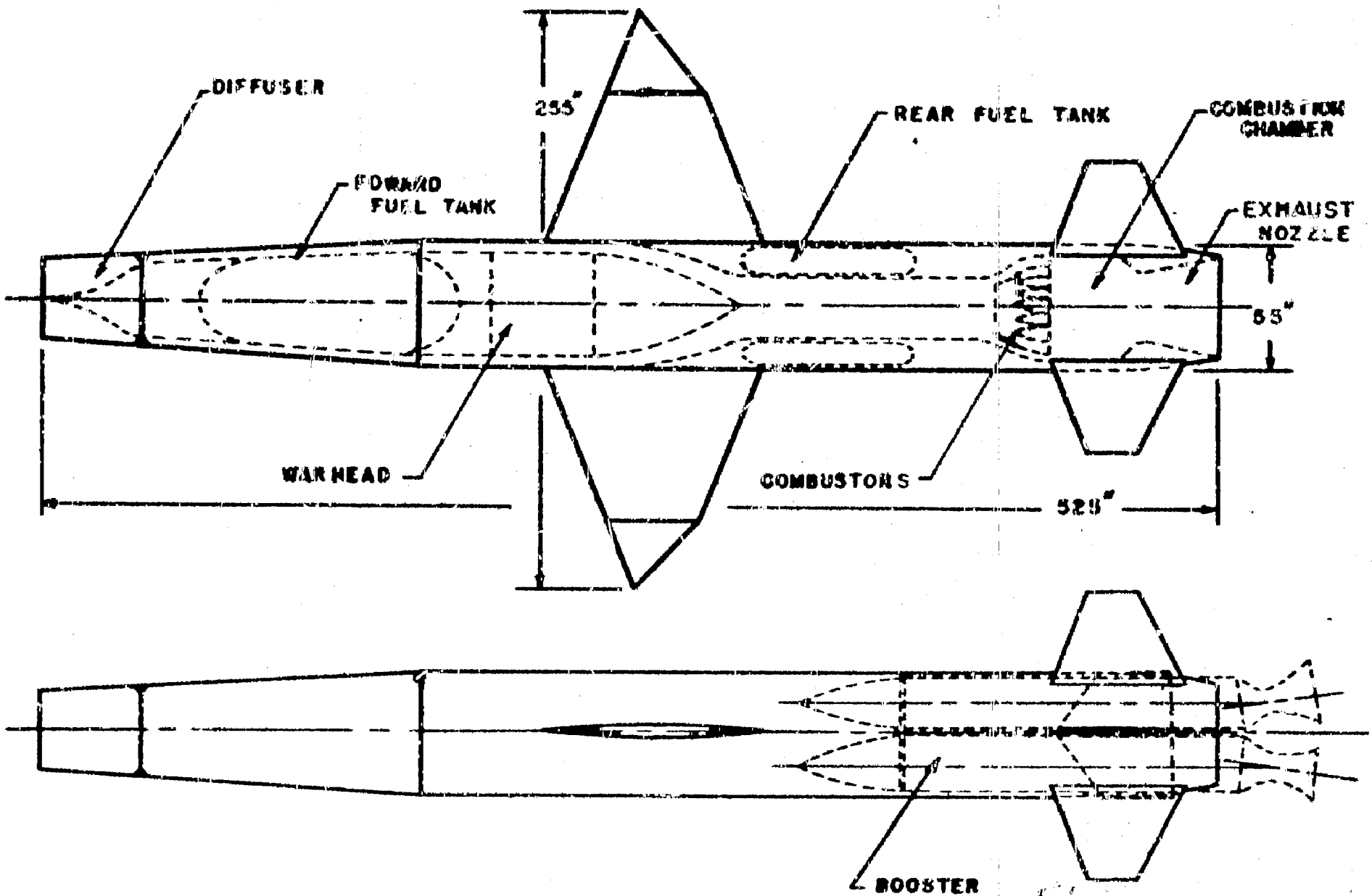
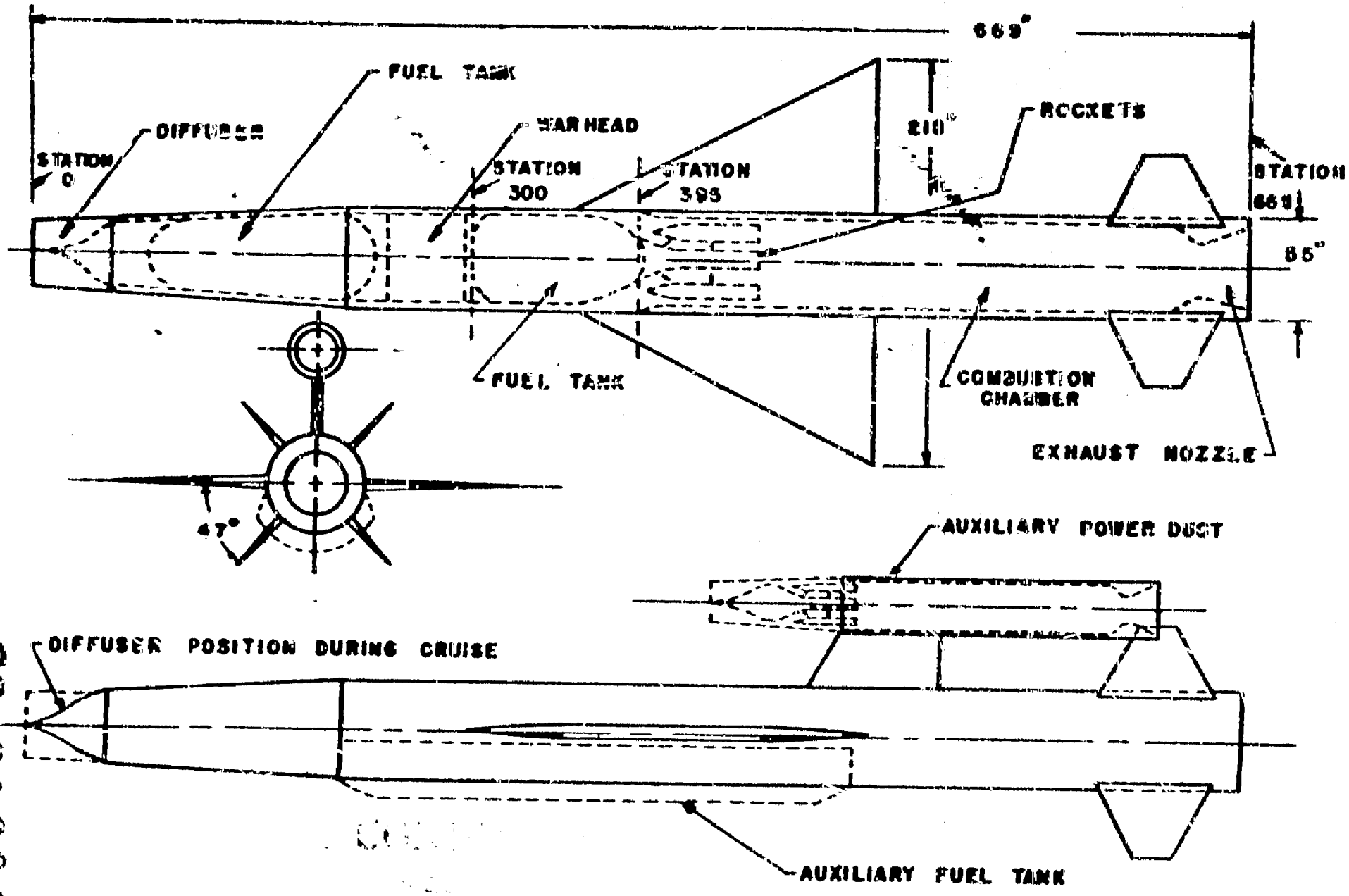
Fig.6.



SKETCH of TRITON CONFIGURATION WITH RAMJET ENGINE

Scale ≈ 1/60

Figure: 7.



SKETCH of MODIFIED TRITON CONFIGURATION WITH RAM ROCKET ENGINE.

Scale = 1/70

Figure: 8.

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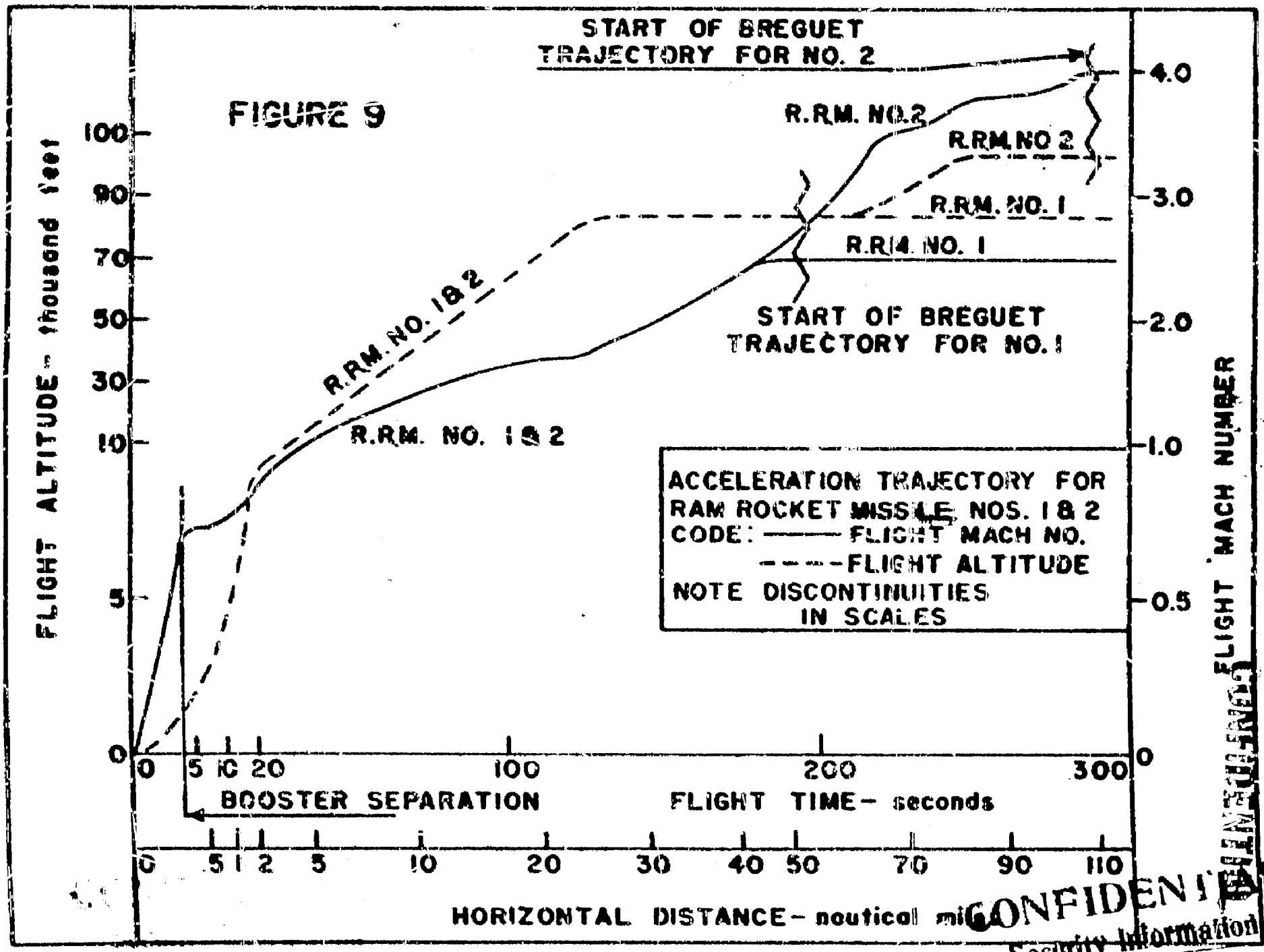

FIGURE 9

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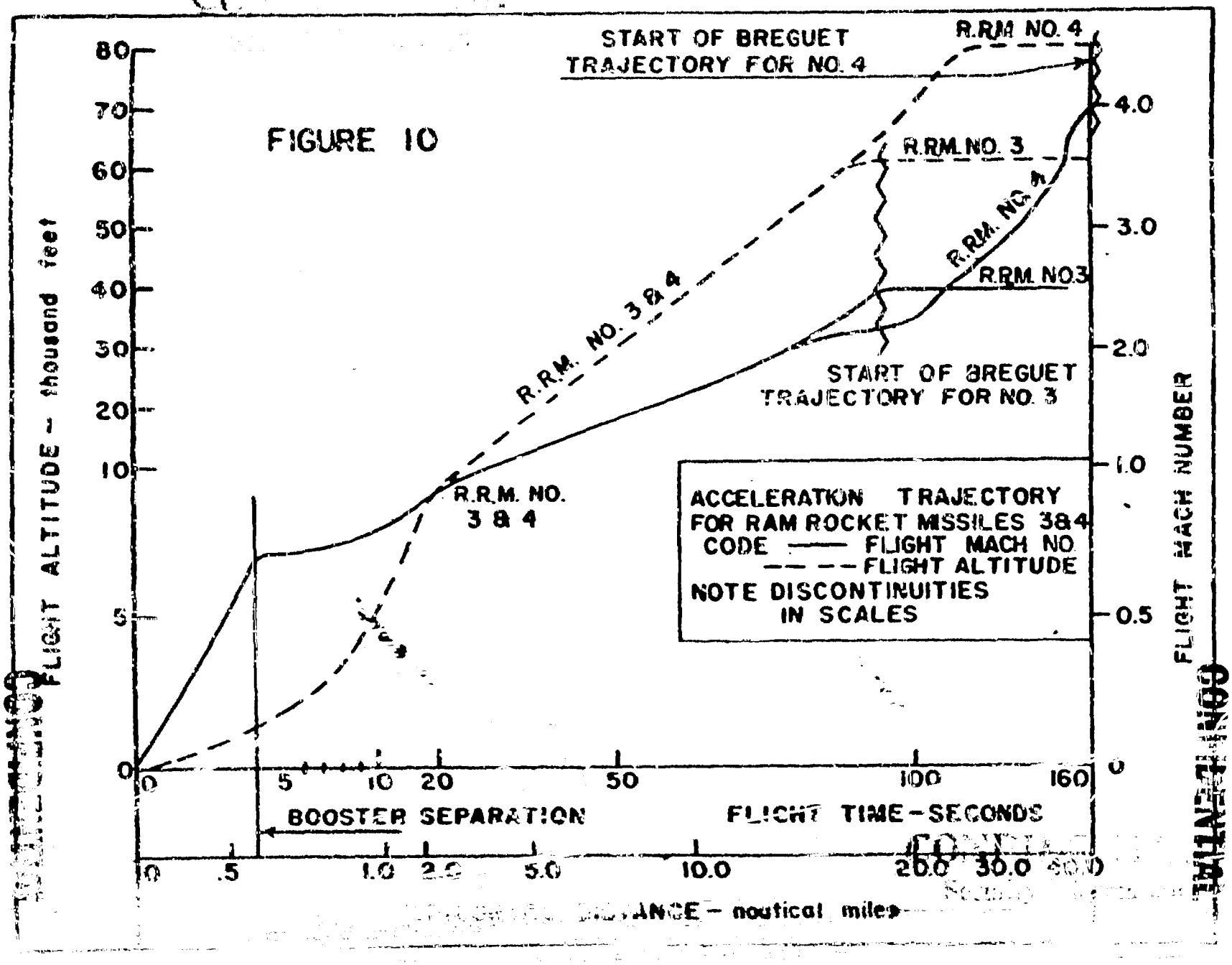
FIGURE 10
START OF BREGUET TRAJECTORY FOR NO. 4
R.R.M. NO. 4
R.R.M. NO. 3
R.R.M. NO. 4
R.R.M. NO. 3 & 4
R.R.M. NO. 3
START OF BREGUET TRAJECTORY FOR NO. 3
R.R.M. NO. 3 & 4
ACCELERATION TRAJECTORY FOR RAM ROCKET MISSILES 3&4
CODE —— FLIGHT MACH NO
- - - - FLIGHT ALTITUDE
NOTE DISCONTINUITIES IN SCALES
FLIGHT ALTITUDE - thousand feet
FLIGHT MACH NUMBER
BOOSTER SEPARATION
FLIGHT TIME - SECONDS
RANGE - nautical miles
80
70
60
50
40
30
20
10
5
0
4.0
3.0
2.0
1.0
0.5
0
5
10
20
50
100
160
.5
1.0
2.0
5.0
10.0
20.0
30.0