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SUBJECT **NOTE ON SUPERSONIC BOMBERS POWERED BY TURBO-JETS**

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ISSUED TO Internal

SUMMARY

This note examines the possibility of achieving long range with turbo-jet bombers designed to cruise at supersonic speeds. It is concluded that still air ranges up to 5000 miles from the top of the climb are possible at low supersonic speeds in view of recent aerodynamic advances. At Mach numbers between 1.5 and 2.0, however, maximum range appears to decrease to about 3000 miles. In all cases little increase in range is achieved by increasing aircraft gross weight above 300,000 to 400,000 pounds. Altitudes over the target are over 50,000 ft., and in some cases 70,000 ft.

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LIST OF SYMBOLS

a	speed of sound (ft./sec.)
A	wing aspect ratio
b	wing span (ft.)
c	engine specific fuel consumption (lb./lb.-hr.)
C_{D_b}	fuselage drag coefficient (based on fuselage frontal area)
C_{D_f}	wing skin friction drag coefficient
C_{D_t}	wing thickness drag coefficient
$C_{D_{o_w}} = C_{D_f} + C_{D_t}$	= wing zero lift drag coefficient
D_c	cruising drag (lb.)
e	wing span efficiency
K_e	$\frac{\text{engine weight x o}}{\text{thrust}}$ (assumed to be a function of Mach number only in the stratosphere)
M	design cruising Mach number
n	ultimate load factor (assumed to be 6.0)
R	still air range from top of climb (miles)
S	gross wing area (sq.ft.)
S_b	fuselage fontal area (sq.ft.)
W	aircraft weight (lb.)
W_o	aircraft gross weight at take-off (lb.)
W_f	weight of fuel remaining at top of climb (lb.)
W_{f_o}	weight of fuel at take-off (lb.)
W_1	aircraft weight at top of climb (lb.)
W_w	wing structure weight (lb.)
$\frac{dC_D}{dC_L^2}$	drag due to lift factor
β	$\sqrt{M^2-1}$
Λ_o	wing leading edge sweepback

Λ wing structural sweep

ρ_o standard sea level relative density = 0.00238 slug/cu.ft.

σ relative density at cruising altitude (varies during flight)

σ_o relative density at effective initial cruising altitude

τ wing thickness-chord ratio.

LIST OF ILLUSTRATIONS

1.0 INTRODUCTION

Some months ago it was decided to initiate within the Division a study of supersonic bomber capabilities, in order to provide a guide toward future work relating to interception devices. Mr. A.D. Wood has completed one part of this study, dealing with long range ballistic rockets (Reference 1).

The present memorandum describes the results of calculations of range for conventional bomber configurations powered by turbo-jet engines. It is understood that a third memorandum will be published by the Gas Dynamics Laboratory, which discusses the possibility of increasing range by the use of "lifting" engines of a type similar to those which have been proposed by Messrs. Rolls-Royce.

The cruising speed range dealt with in the present analysis extends from a Mach number of 0.9 to 2.0.

In addition to the primary reason for this study, several other aims were borne in mind in carrying out this part of the work. In the first place, it provided an opportunity to assess the potential benefits from the area rule and from the use of wing camber at low supersonic speeds. Secondly, the collection and correlation of supersonic wing data, which was carried out by the Aerodynamics Laboratory during the past two years, had never been put to use in a systematic analysis of aircraft configurations. For the supersonic bombers considered here, the wing configurations have been chosen from the results of these empirical correlations.

2.0 OUTLINE OF METHOD OF ANALYSIS

This section (paragraphs 2.01 to 2.16 inclusive) may be omitted by the reader who is interested only in the results of the analysis. The following paragraphs describe the method of calculating range and the assumptions made regarding items of weight and the estimation of drag.

2.01 Payload

Since in general the weight of some parts of the structure of an aircraft can not be assumed to be a constant fraction of design gross weight, independently of gross weight itself, it appeared necessary to assume at the outset an absolute value for payload. This was chosen to be 10,000 lb.

2.02 Fuselage

For a manned bomber carrying this order of payload, the fuselage diameter tends to be fixed at about 10 feet. Since, also, we are discussing supersonic bombers, it is possible to specify a desirable fuselage fineness ratio of about 10 in order to keep total fuselage drag at a minimum. Hence a fuselage 100 ft. long and 10 ft. in diameter was chosen. The fuselage is thus a body of fixed size (and structure weight), whose drag can be calculated immediately for any given Mach number and altitude. At supersonic speeds its drag coefficient was taken to be 0.20 (based on frontal area) and at subsonic speeds, 0.008. The supersonic value is conservative when compared with measured drag coefficients of good smooth bodies. A slightly conservative value was chosen because it was decided in the interests of simplicity not to assume any drag associated with the propulsion system installation.

According to the statistical data of Reference 2, this fuselage weighs 10,000 lb.

2.03 Engine Weight

In order to be consistent with the analysis carried out by the Gas Dynamics Section, the engine specific weight in the present case was taken to be the same as that assumed by Mr. Tyler. This is outlined below:

Mach number:	1.0	1.25	1.50	1.75	2.0
Engine weight per lb. of thrust at 50,000 ft:	1.385	1.16	1.01	0.93	0.97

These data were extrapolated to a value of 1.5 at a Mach number of 0.9. It is understood from Mr. Tyler that these figures are slightly optimistic as compared with practical values at the present time.

In order to calculate engine weight as a fraction of aircraft weight it was assumed that the engines are just large enough to produce a thrust equal to cruising drag at maximum continuous rating.

2.04 Fixed Equipment

The weight of fixed equipment, including electronic bombing aids, instruments, etc., was taken to be 5000 lb.

2.05 Undercarriage

The weight of undercarriage was assumed to be 0.06 W_o where W_o is the aircraft gross weight.

2.06 Climb Fuel

It was assumed that the weight of fuel required for take-off and climb to cruising altitude is 0.05 W_o. This value was taken from an analysis of turbo-jet transports carried out in the laboratory a number of years ago. It appears likely that the supersonic bombers considered here would climb initially at subsonic speeds.

2.07 Tail Weight

The weight of the tail is usually a small fraction of gross weight and need not be estimated with great accuracy. According to Driggs (Fig. 37) the tail may be expected to weigh about 5 lb. per square foot of tail area. If the tail area is about one-third of the wing area and if the aircraft wing loading is, say, 100 lb./sq.ft., the tail weight is of the order of 0.02 W_o. This value has been assumed.

In cases where the aircraft might conceivably be of tailless design this value is still retained. Usually there is very little to choose between a tailed and a tailless configuration (where both are possible) because the structure weight saving in a tailless design is likely to be largely offset by increased trimming drag.

2.08 Wing Weight

The wing structure weight has been estimated using Drigg's wing weight equation (Reference 2). The actual formula given by Driggs was simplified for the present case by assuming that all of the likely wings would be highly tapered. The revised wing weight equation is:

$$\frac{W_w}{W_o} = \frac{0.809\ n\ b}{45000\ \cos\Lambda}\left[4.95 + \frac{0.065\ A}{\tau \cos\Lambda}\right]$$

where n = ultimate load factor (assumed to be 6.0)
b = wing span (feet)
Λ = wing structure sweep (assumed to be the sweep of the mid-chord line)
A = wing aspect ratio
τ = wing thickness-chord ratio

2.09 Fuel Weight

When all of the above items are added and subtracted from gross weight, the remainder is the available weight of fuel for cruising, W_f.

2.10 Range Equation

The aircraft is assumed to cruise at constant Mach number M, in such a way that W/σ remains constant, where W is aircraft weight and σ is the relative density at cruising altitude. Thus the altitude increases as fuel is consumed. This cruise procedure results in constant lift coefficient and hence constant ratio of drag to weight. Thus cruising drag decreases as altitude increases and at the same rate as the reduction of engine thrust with altitude. Hence maximum continuous power is required throughout the cruise. Under these circumstances it can be shown that:

$$R = \frac{Ma}{c\left(\frac{D_c}{W}\right)} \log_e\left(1 + \frac{W_f}{W_1}\right)$$

where R = range (miles)
a = speed of sound
c = specific fuel consumption (lb./lb.-hr.)
D_c/W = cruising drag-weight ratio
W_f = fuel weight available for cruise
W_1 = aircraft weight at top of climb

Now it is a good approximation (for W_f/W_1 up to 0.6), to approximate this by the relation

$$R = \frac{Ma}{c\left(\frac{D_c}{W}\right)} \times 0.83\frac{W_f}{W_1} \quad \text{if a is in miles per hour}$$

$$\text{or } R = 0.570 \frac{Ma}{c\left(\frac{D_c}{W}\right)} \times \frac{W_f}{W_1} \quad \text{if a is in ft./sec.}$$

$$= 0.600 \frac{Ma}{c\left(\frac{D_c}{W}\right)} \times \frac{W_{f_o}}{W_o}$$

where W_{f_o} is the total fuel load at take-off, such that

$$\frac{W_{f_o} - W_f}{W_o} = 0.05$$

The fuel weight fraction $\frac{W_{f_o}}{W_o}$ can be computed by adding up all other items of fractional weight and subtracting the sum from unity. However, before some of these items can be calculated (wing weight for example) it is necessary to know the wing aspect ratio, sweep and thickness ratio, as well as wing area, and cruising altitude. Similarly, these quantities must be chosen before a calculation of the cruising drag-weight ratio is possible. The choice of these quantities is discussed below.

2.11 <u>Choice of Wing Aspect Ratio</u>

The above considerations may be summarized by writing the range equation in the following functional form:

$$R = f(W_o, M, A, \Lambda_o, \tau, S, \sigma)$$

where Λ_o = wing leading edge sweep
σ = relative density associated with the cruising altitude at a particular point on the cruise.

All the other quantities are as previously defined.

The method of analysis used here was to fix the cruising Mach number M at a particular value and compute range R for several values of W_o usually ranging from 100,000 lb. to 500,000 lb. Thus at any one value of W_o and M, the above relation reduces to

$$R = f(A, \Lambda_o, \tau, S, \sigma)$$

Ordinarily it would be desirable to choose values for all of these variables such that R is a maximum for the given values of W_o and M. However, the computations involved would be prohibitive and not worth the effort in the present case at least. If the functional relation could be written down analytically, the optimum solution could be found in theory at least by solving the 5 equations:

$$\frac{\partial f}{\partial A} = 0, \quad \frac{\partial f}{\partial \Lambda_o} = 0, \quad \frac{\partial f}{\partial \tau} = 0, \quad \frac{\partial f}{\partial S} = 0, \quad \frac{\partial f}{\partial \sigma} = 0$$

The drag correlation data of References 3 and 4 do provide all the information necessary, together with the assumptions already made regarding weight items, to permit the function f to be so expanded, but it is so complex that partial differentiation is nearly hopeless, much less a solution of the resulting equations.

On the other hand, some degree of optimization is desirable since it is clearly impossible to try and guess simultaneous values of all of the above five variables which will guarantee something like the best possible range. Furthermore it is of interest to know what the optimum values of some of these variables are, at least approximately. For example if the optimum value of cruising altitude (represented by the variable σ) should turn out to be a low one, say less than 35,000 ft., then this in itself tends to rule out such a bomber from serious consideration as a threat.

Intuitively one expects that each one of these five variables has an optimum value, for fixed values of the other four. This can be seen by considering what happens at extreme low and extreme high values of each. Consider, for example, wing thickness-chord ratio τ. If all of the other four quantities are held fixed temporarily while τ is allowed to vary from very low to very high values it is obvious that at very low values no fuel can be carried because the wing structure weight becomes too high. At very high values of τ, wing weight is low and fuel can be carried, but the wing drag eventually becomes so large that the range decreases rapidly. Hence there is an optimum value of τ. Similarly for A, Λ_0, S, and σ.

Fortunately, for two of these variables, aspect ratio A, and leading edge sweep Λ_0, the correlation data in References 3 and 4 permit a choice to be made which is clearly not far from an optimum.

The correlation of drag due to lift of swept wings given in Reference 3 showed that to a reasonable degree of approximation, this parameter can be calculated (for uncambered wings) by the Busemann relation

$$\frac{dC_D}{dC_L^2} = \frac{\beta}{4} \times \frac{\beta A}{(\beta A - \frac{1}{2})}$$

where $\beta = \sqrt{M^2-1}$

It is noteworthy that this expression as developed by Busemann was meant to apply only to finite rectangular wings. The expression shows that dC_D/dC_L^2 has a minimum value equal to $\beta/4$ and that the effect of aspect ratio is negligible if A is large. Hence there is no point in choosing an aspect ratio larger than some certain value, since the only result will be an increase in wing weight. It was therefore decided to choose aspect ratio so that dC_D/dC_L^2 is just 20 percent above the theoretical minimum, $\beta/4$. The above equation can be used to show that this results in the condition that $\beta A = 3$, and hence the aspect ratio is specified at each design cruising Mach number. When this condition is evaluated, the following values of wing aspect ratio are obtained for the design cruising Mach numbers assumed:

M =	1.2	1.4	1.6	1.8	2.0
A =	4.60	3.08	2.40	2.00	1.73

It should be emphasized that the correlation of data presented in Reference 3 applies only to uncambered wings and analysis showed that such wings develop little or no leading edge suction at supersonic speeds. In the present study it was decided to apply modifications to the M = 1.2 bomber in the form of area-rule drag savings, and wing camber. Consequently one case was worked out for the unmodified and uncambered bomber, and in this case drag due to lift was calculated from the Busemann formula. Later, computations were made assuming that a reasonably large fraction of the full theoretical leading edge suction was realized at the design lift coefficient. In this case the wing plan form was left unchanged, but a value of the span efficiency e was chosen equal to 0.6, instead of the value 0.35 which the above method of calculation predicts for the uncambered wing at M = 1.2.

For the subsonic bomber an aspect ratio of 6 was chosen arbitrarily.

2.12 Choice of Wing Leading Edge Sweep

The drag correlation for swept wings at supersonic speeds, which is contained in Reference 4, permits a crude but rational choice of wing leading edge sweep to be made. The correlation showed that provided the Mach lines are swept behind the trailing edge, but ahead of the leading edge, the wing thickness drag (wave drag) is given approximately by the relation:

$$\frac{C_{D_t}}{\tau^2} \tan \Lambda_o = 5$$

On the other hand, when the tangent of the sweep of the Mach lines is greater than about 1.5 times the tangent of leading edge sweep, the following relation holds

$$\frac{C_{D_t}}{\tau^2} = \frac{7}{\beta}$$

Both of these expressions hold only for wings with "conventional" aerofoil sections.

The second expression shows that a straight, or nearly straight wing has a comparatively high drag at low supersonic speeds, and the first of the two expressions shows that it can be greatly reduced by wing sweep, at least up to a sweepback of 50°. On the other hand, sweepback much in excess of this causes a rapid increase in wing structure weight. As a guess, therefore, a sweep of 50° was chosen for low supersonic speeds. At the other end of the scale, at M = 2, the effects of sweep on thickness drag are small until the sweep exceeds 50°. However, the low aspect ratio already chosen means that even for a leading edge sweep of 50°, the penalty on wing structure weight is negligible because the structural sweep is small. For convenience, therefore, a leading edge sweep of 50° has been chosen for all of the supersonic bombers. It should be pointed out here that the drag correlation of Reference 4 failed to confirm that swept wings have higher drag than straight or nearly straight wings when the Mach lines are swept behind the leading edge, at least for wings with conventional aerofoil sections.

2.13 Choice of Thickness-Chord Ratio

For the longest range bombers of this series the wing weight is a considerably smaller fraction of gross weight than is the weight of fuel. Hence it might be expected that it is more important to save drag and fuel consumption than to save wing weight. In other words the optimum wing thickness-chord ratio may be such that the second term in the wing weight equation (see paragraph 2.08) is somewhat greater than the first term, which does not contain thickness ratio. In order to check this the range was calculated for a representative case ($M = 1.6$, $W_o = 200{,}000$ lb.) leaving only thickness-chord ratio a variable. It was found that for maximum range the thickness ratio was such that the thickness term in the weight equation was about 1.5 times the other term. This ratio was held constant for all other cases, and results in the following values of thickness ratio for the supersonic bombers:

Design Cruising Mach No:	1.2	1.4	1.6	1.8	2.0
Thickness-Chord Ratio:	0.0505	0.0306	0.0223	0.0178	0.0151

The very small wing thickness at high Mach numbers is interesting.

For the area rule bomber which cruises at M = 1.2, the wing thickness ratio was arbitrarily increased to 10 percent on the assumption that the wing wave drag could be cancelled by indenting the fuselage. For the subsonic bomber the thickness ratio was again taken to be 10 percent.

2.14 Calculation of Wing and Tail Zero Lift Drag Coefficient

In order to compute the cruising drag-weight ratio, it is necessary to estimate the zero lift drag of the aircraft. The assumptions regarding fuselage drag have been discussed previously (paragraph 2.02). For the wing a value of skin friction coefficient of 0.005 was assumed at supersonic speeds and 0.006 at subsonic speeds. The wing thickness drag was calculated from the relations given in paragraph 2.12, which are based on the empirical correlation of Reference 4. In estimating tail weight (paragraph 2.07) it was assumed that the tail area is about 30 percent of wing area. Thus the zero lift drag of the wing has been increased by 30% to include the tail drag.

In the case of the area rule bomber, a different procedure was followed. The wing skin friction drag coefficient was again taken to be 0.005, and this was increased by 30% as a tail allowance. However, the effective wing and tail wave drag coefficient was assumed to be zero, since the experimental evidence to date indicates that, apart from skin friction, the drag of a wing-body combination can be reduced approximately to that of the body alone, by suitable changes of body shape.

Before the total zero lift drag coefficient of the aircraft can be calculated, the wing area must be chosen because the fuselage size is fixed.

2.15 Choice of Wing Area and Design Cruising Altitude

The method used to choose near-optimum values of these two variables requires that they be dealt with together.

Strictly speaking, design cruising altitude is not a constant for any one aircraft, but varies throughout the flight in accordance with the assumption that W/σ remains constant (see paragraph 2.10). If W/σ remains constant it is meaningful to specify an effective initial cruising altitude at which the relative density is σ_0, where

$$\sigma_0 = \frac{W_0}{W/\sigma}$$

As pointed out in paragraph 2.10, an aircraft flying at constant Mach number and constant W/σ, cruises with a constant ratio of drag to weight.

As a first approximation it might be supposed that the optimum design cruising altitude would be such that total drag is a minimum (for constant wing area, wing configuration, Mach number and weight) since this would require minimum fuel consumption and since this variable has no effect on structure weight. In this case the drag due to lift (induced drag) would be equal to total profile drag. However, further consideration will show that if the design cruising altitude is lowered somewhat from this value, the cruising drag will increase slightly, but the required engine weight will decrease appreciably. Hence more fuel can be carried and range will be greater. Sample calculations of range versus design cruising altitude were carried out for the same case as previously used in obtaining a criterion for thickness ratio. It was found that the optimum design cruising altitude was such that total aircraft profile drag was about 1.5 times the induced drag. This ratio was retained.

Hence,

$$\frac{\sigma_o \rho_o M^2 a^2}{2W_o}\left[C_{D_b}S_b - 1.3C_{D_{o_w}}S\right] = 1.5 \times \frac{2W_o}{\sigma_o \rho_o M^2 a^2 S} \times \frac{dC_D}{dC_L{}^2}$$

where ρ_o = sea level standard density

C_{D_b} = body drag coefficient

S_b = body frontal area

$C_{D_{o_w}}$ = wing zero lift drag coefficient

S = wing area

The other quantities in the above are as defined previously.

The choice of wing area would appear at first to be more complex because it has a direct effect on both drag and structure weight. However, once again a crude guess can be made, which can be checked by sample calculations. In the present group of aircraft the body drag is fixed if cruising altitude and Mach number are determined. Hence changes in wing area have no effect on this part of the drag. It might therefore be expected that the optimum wing area would be close to the value which gives minimum total wing and tail drag, i.e., where wing plus tail profile drag is equal to wing induced drag. In this condition the fuel consumption will be a minimum and also the engine weight will be a minimum (at constant altitude, aircraft weight, Mach number, etc.). As wing area decreases, wing weight also decreases although not rapidly, and because of this it is to be expected that the optimum wing area is actually a little lower than the first guess. However, the same sort of sample calculations as were carried out for thickness-chord ratio and cruising altitude showed that maximum range was achieved for a wing area such that wing plus tail profile drag was very nearly equal to (but slightly less than) wing induced drag. Hence, in the analysis for all bombers these two drag items were kept equal. In other words,

$$\frac{\sigma_o \rho_o M^2 a^2}{2W_o}\left[1.3C_{D_{o_w}} S\right] = \frac{2W_o}{\sigma_o \rho_o M^2 a^2 S} \times \frac{dC_D}{dC_L{}^2}$$

This equation, together with the one given above can be solved for σ_o and S, and the result is

$$S = \frac{2C_{D_b}S_b}{1.3C_{D_{o_w}}}$$

and

$$\frac{\sigma}{W} = \frac{\sigma_o}{W_o} = \frac{\sqrt{1.3\frac{dC_D}{dC_L{}^2} \times C_{D_{o_w}}}}{C_{D_b}S_b \rho_o M^2 a^2}$$

It will be noted that wing area is independent of design gross weight and cruising altitude decreases as gross weight increases.

2.16 Calculation of Range

When all of the above assumptions regarding weight and drag are gathered together, the following range equation can be written down:

$$\text{Range (miles)} = \frac{\frac{0.6Ma}{c}\left\{0.87 - \frac{25000}{W_o} - \frac{K_e}{\sigma_o}\left(\frac{D}{W}\right)_c - \frac{0.809n\sqrt{AS}}{45000\cos\Lambda}\left[4.95 + \frac{.065A}{\tau\cos\Lambda}\right]\right.}{\left(\frac{D}{W}\right)_c}$$

where W_o = design gross weight

$K_e = \frac{\text{engine weight} \times \sigma}{\text{thrust}}$

$\left(\frac{D}{W}\right)_c$ = cruising drag-weight ratio

$= 2.5\sqrt{1.3\frac{dC_D}{dC_L{}^2} \times C_{D_{o_w}}}$

$$\frac{dC_D}{dC_L^2} = 0.30\sqrt{M^2-1}$$ (except in case of subsonic and cambered supersonic bombers)

c = specific fuel consumption (lb./lb.-hr.)

$$S \doteq \text{wing area} - \frac{2C_{D_b}S_b}{1.3C_{D_{o_w}}}$$

$$\sigma_o = \frac{W_o\sqrt{1.3\frac{dC_D}{dC_L^2} \times C_{D_{o_w}}}}{C_{D_b}S_b\rho_o M^2 a^2}$$

This equation was evaluated for a range of weights from 100,000 to 400,000 lb. and a range of Mach numbers from 0.9 to 2.0.

3.0 DISCUSSION OF RESULTS

3.01 Aircraft Configurations

According to the methods outlined above for arriving at near-optimum configurations, the wing configuration and wing area are independent of design gross weight. Furthermore the fuselage dimensions were assumed to be fixed for all aircraft. Hence it is possible to sketch the plan views of the aircraft which result from the analysis, and these sketches will be a function only of design cruising Mach number. The aircraft configurations are shown in Figure 2. Two are drawn for a Mach number of 1.2. The upper one is the bomber which makes full use of the area rule for reducing profile drag and of wing camber for reducing drag due to lift. This is the largest aircraft of the group in terms of wing area, a fact which is explained by the equation for wing area developed in paragraph 2.15.

Some of these configurations have a peculiar appearance, to say the least. In practice they would probably vary considerably from those shown, because if these shapes produce nearly maximum range for a given gross weight it follows that range should not change greatly with relatively large changes in configuration.

3.02 Range

The calculated range as a function of design cruising Mach number and gross weight is plotted in Fig. 1. These ranges are based on turbo-jet engines having the following characteristics as suggested by Mr. Tyler of the Gas Dynamics Laboratory:

Mach Number	1.2	1.4	1.6	1.8	2.0
Engine weight per pound thrust at 50,000 ft:	1.385	1.16	1.01	0.93	0.97
Specific fuel consumption:	1.145	1.17	1.22	1.28	1.35

It will be noticed at once that the supersonic bombers which make no use of the area rule or wing camber have much lower still air ranges than the subsonic bomber. In general this is due to wave drag. As design cruising Mach number is increased above 1.2 there is at first an increase in range for a given gross weight. As the Mach number approaches 2, however, range again begins to decrease. It should be noted that the analysis took no account of the effects of aerodynamic heating on structure weight. These effects would become noticeable at Mach numbers slightly above 2.0 and at the same time the reduction of thrust of a turbo-jet engine would further decrease range in this area. It therefore appears that a Mach number of 2.0 represents an upper limit to the design of turbo-jet powered bombers in the foreseeable future.

The range of the "simple" supersonic bombers is of the order of 3000 miles at the largest weights, and it increases very little as gross weight is increased above 300,000 to 400,000 lb.

The question arises as to whether the range of a bomber could be increased if it flies most of the distance at subsonic speeds. It is reasonable to suppose, because of the relatively short range of intercepting devices, that the bomber may have little to fear over most of its mission and hence it would be sufficient to provide a burst of speed only during a few hundred miles. This question has not been examined at length in the present analysis, but it is clear that the optimum design for efficient supersonic flight is usually much different from that required for economical sub-sonic cruising. Although the maximum lift-drag ratios of the "simple" supersonic configurations are considerably lower than that of a good subsonic aircraft, the configurations designed for Mach numbers above about 1.6 would in themselves have poor subsonic efficiency.

The area rule bomber would have a high lift-drag ratio at subsonic speeds but the calculations indicate that its supersonic range may be nearly as great as that of a subsonic bomber in any case.

It is therefore concluded that the supersonic ranges shown in Fig. 1 could not be greatly increased by flying most of the distance at subsonic speeds.

The very large benefits due to employing camber and area rule modifications are clear from Figure 1. Although the calculations must be taken as representative of an ideal case, it is felt that they are not unrealistic. Wind tunnel results are available for a bomber configuration generally similar to the one considered here (Reference 5). At a Mach number of 1.15, these tests gave a maximum lift-drag ratio of 14.5. The methods of drag estimation used here predict a maximum lift-drag ratio of 15 for the cambered area rule bomber shown in Fig. 2, at a Mach number of 1.2.

3.03 Altitude over the Target

Although the cruising altitude over the target would normally be taken as one of the design specifications of a bomber, it has been chosen here only from the point of view of maximizing still-air range. In any practical case, therefore, if the required cruising altitude varies greatly from that shown in Fig. 1 for a specified range and cruising Mach number, the design would have to be compromised in such a way that gross weight would increase.

For the supersonic bombers, the altitudes over the target are generally greater than 50,000 ft. and in some cases (short ranges and high Mach numbers), over 70,000 ft.

For a given range the area rule bomber designed for M = 1.2 is heavier than the subsonic bomber, but can cruise much higher over the target.

4.0 CONCLUSIONS

The following conclusions are drawn from the above analysis, which considers the range possibilities for supersonic turbo-jet bombers carrying a payload of 10,000 lb.

(a) Still air ranges from 3000 to 5000 miles appear to be possible from the top of the initial climb for bombers designed to cruise at Mach numbers between 1.2 and 2.0. The operational radii would be about one-half of these values.

(b) If the benefits of the area rule and of wing camber are not made use of, the maximum still-air range remains approximately constant at about 3000 miles throughout the supersonic speed range.

(c) If, on the other hand, these recent aerodynamic refinements are fully applied, the range can be increased to about 5000 miles at least for design cruising Mach numbers of about 1.2.

(d) Since the potential benefits to be expected from these refinements tend to become small at Mach numbers above about 1.6, it is doubtful if the range can be increased greatly above 3000 miles at the upper end of the speed scale (up to M = 2.0).

(e) At Mach numbers above 2.0 two factors begin to come into effect which tend to reduce range. These are the effect of aerodynamic heating on structure weight, and the increase in turbo-jet specific weight.

(f) Very little increase in range is evident in all cases for increases in gross weight above 300,000 to 400,000 lb.

(g) Altitudes over the target tend to decrease as range (or gross weight) is increased, and as design cruising Mach number is decreased.

(h) Altitudes over the target are generally of the order of 50,000 feet for the supersonic bombers, and in some cases may be as high as 70,000 ft.

(i) The advantages to be gained by the full use of wing camber and the application of the area rule appear to be so great, at least for low supersonic Mach numbers, that this would seem to be not only a possible, but a very probable future trend in the development of long range bombers. An increase in cruising speed from, say, 0.9 to 1.2 increases greatly the difficulty of interception, at least by manned interceptors.

(j) Although no detailed consideration has been given here to the possibility of increasing bomber range by flying only a few hundred miles at supersonic speed, rough considerations indicate that little is to be gained. This question, however, possibly requires some analysis.

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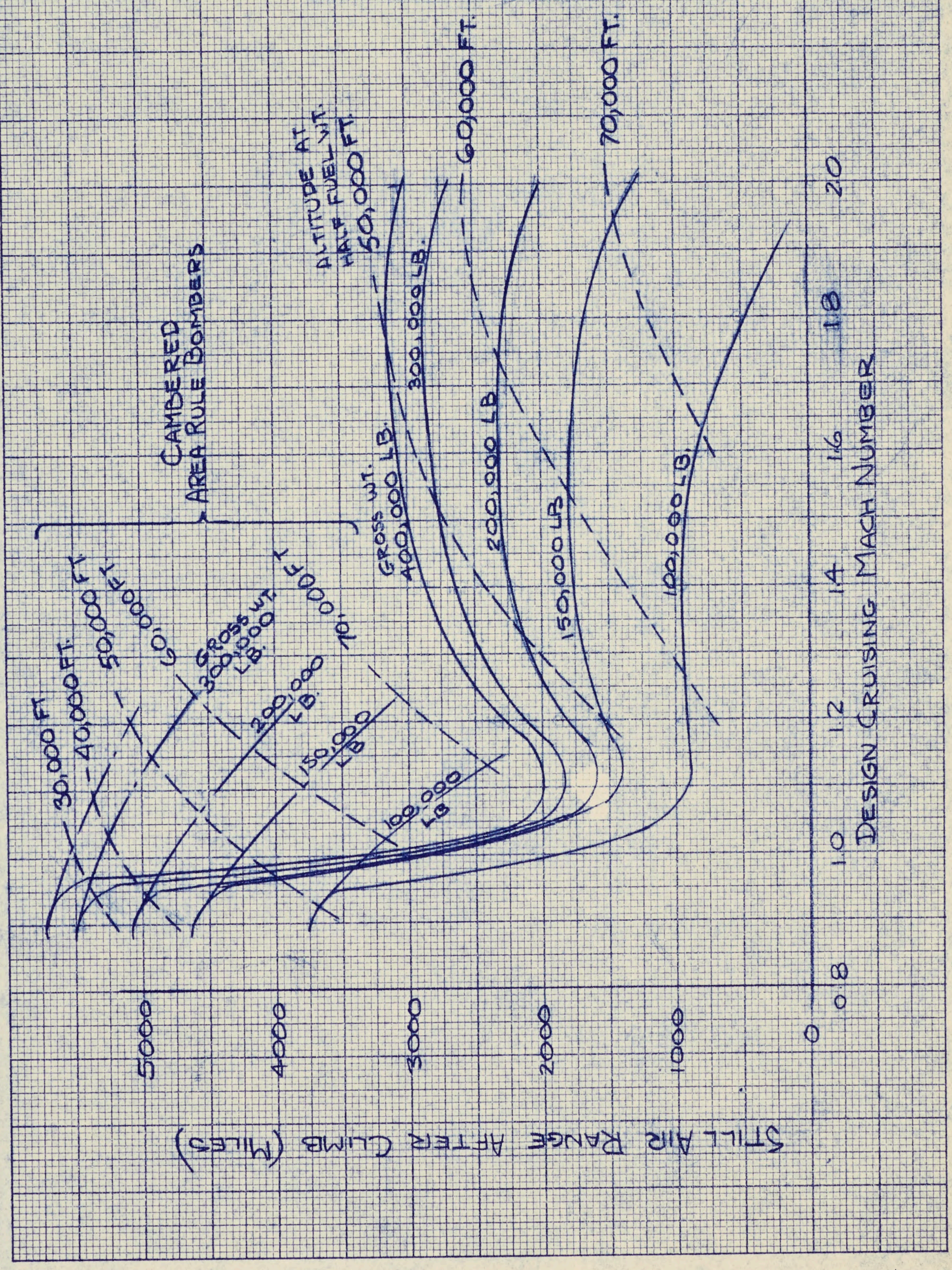
RANGE, WEIGHT, CRUISING ALTITUDE OF SUPERSONIC BOMBERS

FIG 1

PAYLOAD = 10,000 LB.



BOMBER CONFIGURATIONS

FIG. 2