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RESEARCH MEMORANDUM

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RANGE PERFORMANCE OF BOMBERS POWERED BY TURBINE-PROPELLER POWER PLANTS

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

RANGE PERFORMANCE OF BOMBERS POWERED BY TURBINE-PROPELLER POWER PLANTS

By Charles W. Cline

SUMMARY

Calculations have been made to find ranges attainable by bombers of gross weights from 140,000 to 300,000 pounds powered by turbine-propeller power plants. Only conventional configurations were considered and emphasis was placed upon using data for structural and aerodynamic characteristics which are typical of modern military airplanes. An effort was made to limit the various parameters involved in the airplane configuration to practical values. Therefore, extremely high wing loadings, large amounts of sweepback, and very high aspect ratios have not been considered. Power-plant performance was based upon the performance of a typical turbine-propeller engine equipped with propellers designed to maintain high efficiencies at high-subsonic speeds.

Results indicated, in general, that the greatest range, for a given gross weight, is obtained by airplanes of high wing loading, unless the higher cruising speeds associated with the high-wing-loading airplanes require the use of thinner wing sections. Further results showed the effect of cruising at high speeds, of operation at very high altitudes, and of carrying large bomb loads.

INTRODUCTION

In view of recent progress in turbine-propeller power-plant development and especially the research indicating the high propeller efficiencies available at transonic speeds, it seems likely that the problem of long range at high speeds might be somewhat more optimistically undertaken using bombers powered by turbine-propeller power plants. Accordingly, this paper gives the results of calculations or ranges for typical medium to heavy bombers powered by turbine-propeller engines whose performance is representative of present-day turbo-propeller design. These engines drive the thin-bladed, high-efficiency propellers described in reference 1,

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which give unusually high efficiencies in the transonic speed range. Because of the importance of power-plant performance on range, the calculations for both engine performance and propeller efficiencies have been more rigorous than is usual for generalized performance study. Friction losses in the intake duct and in the tail pipe have been considered, as well as power-plant operation at conditions other than ideal.

Emphasis has been placed upon employing values for the aerodynamic and structural characteristics of the bombers that are typical of present-day airplanes in order that the usefulness of the turbine-propeller power plant may be properly evaluated. No design features which might be difficult to apply to actual bombers have been considered, such as very large amounts of sweepback or extremely high wing loading; furthermore, the bombers for which the calculations have been made are relatively high-speed airplanes, requiring thin wing sections and high power-plant weight. Consequently, the ratios $\frac{\text{Initial weight}}{\text{Final weight}}$ and $\frac{\text{Lift}}{\text{Drag}}$ are not as high as would be desirable for bombers designed solely for long range. An example of the range of values of these parameters for the bombers of this report is given in the following table:

Gross weight (lb)	Number of engines	Wing loading (lb/sq ft)	$\frac{\text{Initial weight}}{\text{Final weight}}$	$\left(\frac{\text{Lift}}{\text{Drag}}\right)_{max}$
200,000 to 140,000	2	100 to 60	1.79 to 1.63	21.5 to 19.7
200,000 to 140,000	4	100 to 60	1.62 to 1.44	16.4 to 17.6

SYMBOLS

a speed of sound, feet per second

A duct cross-sectional area, square feet

AR aspect ratio

b wing span, feet

c specific fuel consumption, $\frac{\text{pounds fuel per hour}}{\text{shaft horsepower}}$

C_D	drag coefficient
C_{D_M}	increment of drag coefficient due to compressibility
C_{D_o}	profile-drag coefficient
C_L	lift coefficient
D	drag, pounds
d_f	maximum fuselage diameter, feet
d_p	propeller diameter, feet
e	induced-drag efficiency factor
f_c	duct friction coefficient ($\Delta H/q$)
f_{ce_1}	effective inlet-duct friction coefficient
F_g	gross jet thrust, pounds
F_N	net jet thrust, pounds
g	acceleration due to gravity, feet per second per second
ΔH	loss in total pressure in a duct, pounds per square foot
JHP	jet horsepower
l_f	fuselage length, feet
L	lift, pounds
m	mass flow of gas through tail pipe, slugs per second
M	airplane Mach number (u/a)
n	engine speed, revolutions per minute
N	design load factor
p	pressure, pounds per square foot

q dynamic pressure, pounds per square foot $\left(\frac{1}{2}\rho u^2\right)$

R gas constant, $\frac{\text{feet}^2}{\text{second}^2}\ {}^{\circ}\text{F}$

S area upon which drag coefficient is based, square feet

SHP engine shaft power, horsepower

t wing taper ratio $\left(\frac{\text{Root chord}}{\text{Tip chord}}\right)$

T temperature, $^{\circ}$F absolute

u fluid velocity, feet per second (without subscript, u is airplane velocity in ft/sec)

V airplane velocity, miles per hour

w air flow through engine, pounds per second

W airplane weight, pounds

W_D weight of load and structures carried in or on wing, pounds

W_E airplane gross weight minus weight of fuel and bombs, pounds

$W_F = W_E - W_w$

W_G airplane gross weight, pounds

W_W wing weight, pounds

γ ratio of specific heats of a gas

δ relative pressure ratio $\left(p_{T_2}/2116\right)$

η propeller efficiency

$\eta_e = \eta + \frac{\text{JHP}}{\text{SHP}}$

θ relative temperature ratio $\left(T_{T_2}/518.4\right)$

ρ fluid density, slugs per cubic foot

τ wing thickness ratio $\left(\frac{\text{Wing thickness}}{\text{Wing chord}}\right)$

Subscripts refer to:

1 engine diffusor inlet

2 engine compressor inlet

3 engine turbine inlet

7 engine tail-pipe entrance

a free stream

f fuselage

n nacelles

t tail

T total

W wing

nl with tail-pipe losses nl

Wl without tail-pipe losses Wl

ANALYSIS

All ranges calculated are optimum ranges, during which the airplane is assumed to be flown at the speeds and altitudes corresponding to the most efficient operation possible (within its operating capabilities). For the most part, this entails operation at the speed giving maximum lift-drag ratio at as high an altitude as possible, using rated power in all engines. Consequently, the airplane is in a very slight climb as fuel consumption steadily reduces its weight. However, since satisfactory engine operation at altitudes greater than 40,000 feet is not guaranteed in the engine specifications, no ranges or portions of ranges were flown above 40,000 feet. This restriction causes an increase in

specific fuel consumption and a consequent loss in efficiency of operation. No allowances have been made for fuel required for take-off, combat maneuvering, or landing reserve; therefore, the ranges for any conventional military operation may be expected to be lower than those shown in this paper.

No unusual airplane configurations or radical departures from conventional design procedure have been considered. In the determination of specific bomber configurations, aspect ratios, which have a strong influence on both the aerodynamic and structural weight properties of the airplanes, have been selected to give an optimum compromise, in each case, between structural weight and lift-drag ratio. Wing thicknesses τ are either 10, $12\frac{1}{2}$, or 15 percent, and were selected by one of two different criteria: For two-engine bombers, which are relatively low-powered and are designed primarily for long range, the wing thicknesses were selected so as to keep the wings free from compressibility drag rise at the highest bomber cruising speeds. For four-engine bombers, which are relatively high-powered and are designed primarily for high performance, the wing thicknesses were selected to keep the wings free from, or at least to minimize, compressibility drag rise at the top bomber speeds at target weight at an altitude of 35,000 feet. In all cases, the bombers were considered to have wings of 30° sweepback.

Data required for the range calculations can be classified into three general divisions: (1) structure and useful-load weights, (2) aerodynamics, and (3) power-plant performance. For this paper the structure and useful-load-weights data were obtained from an analysis of existing modern military airplanes, plus some estimations of certain useful-load requirements, such as crew size (5 to 7 men), armament provisions (2.2 percent of gross weight), and number of free guns (5 to 7).

Aerodynamic data have been obtained from a general survey of wind-tunnel, flight, and bomb-drop tests on wings, nacelles, and bodies of revolution. Attention has been given to adjusting the results of model tests to realistic values attainable with production-manufactured airplanes.

Power-plant performance has been calculated by using general performance charts of a typical turbine-propeller engine and theoretical analyses of propeller efficiencies at high-subsonic speeds. The engine performance was calculated for inlet losses of 10 percent of the inlet dynamic pressure and tail-pipe losses of 4 percent of the tail-pipe dynamic pressure. Some typical curves of engine performance are shown in figure 1. In calculating propeller performance full advantage has been taken of the high efficiencies shown by reference 1 to be attainable at high-subsonic speeds. The findings of reference 1 indicate, for example, propeller efficiencies of 0.85 at forward Mach numbers as

high as 0.75. Since, in order to realize these high efficiencies, the propeller must be operated at the proper advance ratio, it has been assumed that the power-plant gearbox could be altered from one installation to another to give the required design propeller speed without incurring additional power-plant weight. It should be noted that this procedure gave the optimum propeller operation for a specific configuration but it did not give peak efficiency for all cruise conditions of a given configuration.

Details of these calculations are presented in appendix A.

RESULTS AND DISCUSSION

The results of the calculations are shown in figures 2 to 8. Unless otherwise specified, all bomb loads are 5000 pounds. While this is a very small load, by present-day standards, it has been used in the present calculations because the emphasis of the paper has been in the attainment of long ranges with high-performance airplanes. The effect of bomb load upon range is shown subsequently. Figure 2 shows the ranges attainable by bombers powered by two engines. The discontinuity in the curve for the airplanes of 100-pound-per-square-foot wing loading is caused by a shift in wing thickness from 0.125 to 0.150, made possible by the lower speed at which the heavier bombers cruise. The dotted line is an extension of the curve for a wing thickness of 0.125, which shows the ranges which would have been attained if the shift in wing thickness had not been made. Comparison of these two curves for the 100-pound-per-square-foot wing loading indicates the important effect of wing thickness on range. The longest range shown in figure 2, 8730 miles, is attained by the bomber with 200,000-pound gross weight and 100-pound-per-square-foot wing loading at an average cruising speed of 425 miles per hour. This bomber has a target altitude of 39,000 feet, and a top speed at target weight, 35,000-foot altitude, using rated power, of 485 miles per hour. As the gross weight decreases, the range decreases, but other performance parameters improve. The bomber with 160,000-pound gross weight and 100-pound-per-square-foot wing loading, capable of a range of 8000 miles at an average cruising speed of 455 miles per hour, has a target altitude of 40,000 feet, and a top speed at target weight, 35,000-foot altitude, using rated power, of 521 miles per hour. This same airplane, if it had used a wing thickness of 0.125 would have had 400 miles less range, and a top speed, at target weight, 35,000-foot altitude, using rated power, only 7 miles per hour higher.

The explanation for the longer ranges attained by the airplanes of higher wing loading at most of the gross weights shown lies primarily in the lower structural weight and the lower specific fuel consumption of these bombers. The lower structural weight is a result of the smaller

wing and tail areas used on the high-wing-loading bombers, and the lower specific fuel consumption is due to the higher cruising speed and to the fact that the high-wing-loading bombers reach 40,000 feet at a lower weight than the low-wing-loading bombers and consequently spend less time cruising at reduced power settings.

Figure 3 shows the ranges attained by bombers powered by four engines. It is apparent that, similar to the two-engine bombers, the longest ranges are attained by the bombers of highest wing loading. The much lower ranges achieved by the four-engine bombers are due to the heavier power-plant and structural weights resulting from the addition of two more engines and the reduction of the wing thickness ratio to 10 percent. However, the performance of the four-engine bombers in other respects is far superior to that of the two-engine airplanes. For instance, the bomber with the 200,000-pound gross weight and 100-pound-per-square-foot wing loading is capable of a top speed at target weight, 35,000-foot altitude, using rated power, of 579 miles per hour, and the bomber with the 160,000-pound gross weight and 100-pound-per-square-foot wing loading is capable of a top speed, under the same conditions, of 592 miles per hour. Further indication of the advantage in performance, other than range, that the four-engine airplanes enjoy over the two-engine airplanes is shown in figure 4, which compares the rates of climb of various bombers at 25,000-foot altitude.

Figure 5 indicates the ranges attained by the bombers with four engines and 300,000-pound gross weights, and their corresponding average cruising speeds. At this gross weight, the maximum range is given by the 80-pound-per-square-foot wing loading. This optimum wing loading is a result of a compromise between two factors; as the wing loading increased, it was necessary to decrease the design wing thickness in order to avoid compressibility losses at high speed. This resulted in higher structural weight for the bombers of high wing loading. However, the increased cruising speeds associated with the high-wing-loading airplanes yield lower specific fuel consumption. At 80 pounds per square foot the optimum compromise between these factors is reached and the range becomes maximum.

As has been previously mentioned, all bombers were required to level off after reaching a 40,000-foot altitude. This procedure resulted in operation at reduced power settings and therefore at higher specific fuel consumption. An example of the penalty in range which this procedure involves is shown in figure 6, which shows the ranges attained by bombers with four engines and 200,000-pound gross weights with and without the 40,000-foot altitude limitation. Note that the bomber with the 100-pound-per-square-foot wing loading shows a gain in range of about 120 miles in spite of the fact that it encountered compressibility drag rise at the higher altitudes. Figure 6 also indicates that the gain in average cruising speed for these bombers is around 45 miles per hour,

unless compressibility effects are encountered. For some bombers the gains in going to higher altitude may be expected to be greater than those shown in figure 6, but for others, particularly the high-powered, high-wing-loading airplanes, little or no gain may result, due to the compressibility drag rise encountered at the higher altitudes.

The effect of cruising at high speed is shown in figure 7. Figure 7(a) is a plot of the ranges of three different four-engine bombers as a function of the cruising speed. The end points of the curves are the top speeds, at rated power and full gross weight of the particular bomber represented. It should be noted that the drastic reduction in range which accompanies the increased cruising speeds is due almost entirely to the loss in L/D of the airplane, rather than to any powerplant difficulties. This point is brought out in figure 7(b), which shows the average L/D of the same three airplanes represented in figure 7(a), as a function of cruising speed. Furthermore, the large decrease in L/D which accompanies the higher cruising speeds could be minimized by redesigning the bomber for each speed. The new designs would incorporate such things as thinner wing sections, larger sweep angles, and higher wing loadings than have been considered in this paper.

In all the previously discussed results the bomb load has been 5000 pounds. It is desirable to estimate what the effect of increased bomb load is on the range of the bombers. The decrease in range for a given addition of load will naturally be much more severe for a light bomber than for a heavy bomber. In the preparation of figure 8, the two bombers were selected to show the maximum and minimum effect of bomb load on range of all the bombers discussed in this paper. The airplanes represented in figure 8 were not redesigned for each new bomb load, but continued to have sufficient tankage to accommodate all the fuel carried in the 5000-pound-bomb-load condition.

CONCLUDING REMARKS

1. The calculations show the possibility that a range of 8700 miles at an average cruising speed of 425 miles per hour can be attained by a bomber of 200,000-pound gross weight.

2. Medium bombers capable of ranges of 4000 miles have top speeds, at rated power, in the neighborhood of 600 miles per hour.

3. The longest range, for a given gross weight, is generally achieved by bombers of highest practical wing loading, except when the high speeds associated with high-wing-loading aircraft require the use of thinner wing sections.

4. Large decreases in range accompany very high cruising speeds, but these decreases are due almost entirely to a decrease in airplane lift-drag ratio rather than difficulties with the turbine-propeller power plant.

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APPENDIX A

DETAILED REVIEW OF METHOD OF CALCULATIONS

All ranges were computed using the equation:

$$\text{Range} = 375 \int_{W_E}^{W_G} \frac{1}{D} \frac{\eta + \frac{JHP}{SHP}}{c} \, dW$$

which yields a value in miles. The behavior of the various parameters involved in the equation indicates the procedure for obtaining the maximum range for a given airplane. Specific fuel consumption c decreases with increasing altitude, velocity, and percent rated power. Minimum drag is a constant (barring compressibility effects), but the velocity required to fly at minimum drag increases with increasing altitude. The term $\left(\eta + \frac{JHP}{SHP}\right)$ remains fairly constant with changing altitude and flight speed. Consequently, in order to maximize the expression $\frac{1}{D} \frac{\eta + \frac{JHP}{SHP}}{c}$, all bomber ranges were considered to be flown at the highest altitude at which the velocity required for flight at minimum drag could be attained using rated power in all engines. However, since the engine specifications used in the computations state that the engines were not guaranteed to operate properly above an altitude of 40,000 feet, this procedure was not followed whenever it required that a plane be flown above that altitude. Instead, that portion of the range which would have been flown above an altitude of 40,000 feet was flown at 40,000 feet at the velocity which gave a maximum value of $\frac{1}{D} \frac{\eta + \frac{JHP}{SHP}}{c}$ for that altitude.

Because fuel consumption continuously decreased the weight of the bomber throughout the range, that portion of the range flown below 40,000 feet was flown in a continuous climb. This steady climb required an increment of power over that used to overcome the drag, but since the maximum rate of climb found necessary in any of the range computations was less than 35 feet per minute, this increment of power was neglected in the calculations.

Certain ranges were computed in which the bomber was not required to level off at 40,000 feet, in order to determine the additional range possible if the engines were able to function properly above their guaranteed altitude.

In computing the ranges for the chart of range against cruising speed, the airplanes were considered to be flown at the maximum altitude at which the particular speed was attainable using rated power (but not higher than 40,000 ft). These ranges were flown at constant speed and not necessarily at minimum drag.

In all cases, only absolute ranges have been calculated; no allowance has been made for fuel required for take-off, climb, combat, or landing reserve.

Rates of climb for various bombers at an altitude of 25,000 feet, using rated power, have also been computed, using the equation:

$$\text{Rate of climb} = \frac{\left(\eta_e \text{SHP} - \frac{DV}{375}\right)33,000}{W}$$

which yields a value in feet per minute.

All performance was calculated for an NACA standard atmosphere.

Further examination of the equation used to compute range indicates that three separate divisions of data are required: η, JHP, SHP, and c are power-plant data, W_G and W_E are structural data, and D is aerodynamic data. The methods used in obtaining and applying the data of these three divisions are discussed in the following sections.

Power Plant

Engine performance was calculated from the general performance curves presented in reference 2 for the XT35-W-3 turbine-propeller engine. Both intake-duct and tail-pipe friction losses were included and were handled in the computations as explained in appendix B. The intake-duct losses were considered to be a constant 10 percent of the inlet dynamic pressure, and the tail-pipe losses were 4 percent of the tail-pipe dynamic pressure. The engines were considered to be operated at maximum allowable turbine-inlet temperature since this gives a specific fuel consumption close to the minimum, as shown in appendix C. No allowance was made for power to run the accessories.

The propellers used are the thin-bladed, high-efficiency propellers reported in reference 1 and the efficiencies η were calculated using

references 1 and 3. The assumption was used that the engine-to-propeller gear ratio could be changed from that given in reference 2 without incurring a weight penalty, and the gear ratio for any particular installation was fixed to give propeller advance ratios producing the best average propeller efficiency for that airplane.

It should be noted that the propeller profile efficiencies calculated in reference 1 are optimum efficiencies and it is necessary to operate at a constant value of propeller disk loading to continuously realize these optimum profile efficiencies throughout the variation of power settings, altitudes, and velocities flown at over the entire range of a given airplane. However, since the maximum total variation in propeller disk loading throughout any range problem calculated in this report was less than 15 percent, and since the effect of a variation of this magnitude on profile efficiency is very slight, the effect was considered negligible and was not considered in the calculations.

Similarly, the induced propeller efficiencies given in reference 3 are for propellers having optimum blade loading, and therefore will not be continuously realized as the loading varies from the design condition. Nevertheless, this effect is likewise small and has been neglected in the computations. The combined maximum error incurred in making the above approximations has not been evaluated but is believed to be between 0 and 2 percent.

For ease of calculation, and because the structural problems associated with extremely thin propeller blades are largely unexplored, large-diameter, single-rotating propellers have been used in all cases rather than contrarotating propellers. In most cases these large-diameter propellers would be perfectly satisfactory, but for the light four-engine bombers, the propeller diameters necessary to give satisfactory efficiencies are unduly large, and these bombers would necessarily have to be equipped with dual-rotating propellers of a smaller diameter. Provided that these thin-bladed dual-rotating propellers can be satisfactorily developed, little or no loss in range would result.

Structures and Weights

The structures and weights of all the bomber configurations studied in this report were arrived at through analysis of a number of present-day military aircraft. In order to arrive at a reasonable estimate for the gross and empty weights of a specific configuration, the weights of

the following components were analyzed separately and their weights summed to give values for the airplane weight for any particular condition:

(a) Wing
(b) Fuselage
(c) Tail
(d) Engine and accessories
(e) Propellers
(f) Nacelles
(g) Landing gear
(h) Hydraulic, electric, and communication system, and cabin furnishings
(i) Cabin pressurizing equipment
(j) Surface controls
(k) Anti-icing equipment
(l) Armament provisions
(m) Fuel system
(n) Oil system
(o) Oxygen equipment
(p) Crew
(q) Instruments
(r) Gunfire armament
(s) Bomb
(t) Fuel
(u) Oil

Component (a).- The wing weights of various military aircraft are presented in figure 9. The parameter against which the wing weights are presented is an empirical modification of a parameter which arises in the theoretical solution for the weight of a uniformly loaded, tapered, cantilever beam. The line drawn among the points is the variation of wing weight used in the calculations. For all bomber configurations in the present analysis the design load factor is 3.25, the taper ratio is 2.5, and the distributed weight W_D is the weight of the power plants, plus 12.5 percent of the landing-gear weight (tandem-type landing gear), plus 10 percent of the fuel weight. The aspect ratio is varied to give a maximum value of $\left(\frac{L}{D}\right)_{max} \log \frac{W_G}{W_E}$, as explained in appendix D, and the thickness ratio τ is selected according to the criteria explained in the analysis. No systematic variation of wing weight with sweepback angle could be found and therefore none has been incorporated into this paper. All bomber configurations analyzed here are considered to have 30° sweepback.

Component (b).- Fuselage geometry appears to depend somewhat upon wing loading as seen in figure 10, a plot of fuselage length over wing span against wing loading. However, for purposes of the present analysis, it was felt that a more logical comparison between airplanes of different wing loading could be made if fuselage dimensions, and hence volumetric capacity, depended only upon the gross weight of the airplane and not upon the wing loading. Consequently a $\frac{\text{Fuselage length}}{\text{Wing span}}$ ratio

of 0.832 $\left(\text{corresponding to } \frac{W_G}{S_W} = 80 \text{ lb/ft}^2 \text{ in fig. 2}\right)$, and AR = 9, and a $\frac{\text{Fuselage length}}{\text{Fuselage max. diameter}}$ variation as shown in figure 11 was used to calculate fuselage dimensions for each gross weight. These dimensions, together with figure 12, were used to arrive at fuselage weights. The dimensions and weights are shown in the following table:

W_G (lb)	l_f (ft)	d_f (ft)	Fuselage weight (lb)
300,000	152.8	12.40	16,900
200,000	124.5	11.81	12,130
180,000	118.1	11.63	11,250
160,000	111.3	11.46	10,360
140,000	104.3	11.22	9,360

Component (c).- Tail area was chosen to be 35 percent of the wing area, as shown by the straight line of figure 13, and tail weight to be 4.15 pounds per square foot, represented by the line labeled "High-speed aircraft" in figure 14.

Component (d).- The engine dry weight is given by the engine specifications (reference 2) as 5950 pounds, and an arbitrary 250 pounds has been added for accessories, totaling 6200 pounds per engine.

Component (e).- Propeller diameters, because of the high-power, high-altitude operation of these bombers, are unusually large. Since no aircraft propellers of this size exist at the present time the weights used are estimates. All propellers used are six-bladed single-rotating. The weights and diameters are as follows:

W_G (lb)	d_p (ft)	Propeller weight (lb)
300,000	23.0	1675
200,000	23.0	1675
180,000	22.7	1635
160,000	22.3	1600
140,000	22.0	1575

Component (f).- Nacelles used to accommodate the XT35-W-3 have a length of 24 feet and a frontal area of 25 square feet. The weight was computed using figure 15.

Component (g).- The landing-gear weight is 5.62 percent of the gross weight as shown by the straight line of figure 16.

Component (h).- Hydraulic, communications, and electrical systems, and cabin furnishings comprise 5.1 percent of the gross weight as seen in figure 17.

Component (i).- Cabin pressurizing equipment weight is 0.24 percent of the gross weight, from figure 18.

Component (j).- Surface-control weight is given by the equation: Weight of surface controls = $(740 + 0.103S_W)$ pounds, which is the line labeled "With hydraulic boost" in figure 19.

Component (k).- Anti-icing equipment weight is given by $(-345 + 6.75B)$ pounds. This is the straight line shown in figure 20.

Component (l).- Armament provisions are 2.2 percent of gross weight.

Component (m).- Fuel-system weight is 1 pound per gallon of fuel.

Component (n).- Oil-system weight is 2.5 pounds per gallon of oil.

Component (o).- Oxygen equipment weighs 40 pounds per crew member.

Component (p).- Crew weight is 250 pounds per man; all bombers of gross weight 200,000 pounds or less carry a five-man crew; all bombers of 300,000-pound gross weight carry a seven-man crew.

Component (q).- Instruments weigh 180 pounds.

Component (r).- The weight of the gunfire armament is 2060 pounds for bombers of 200,000 pounds or less gross weight, and 2900 pounds for 300,000-pound gross weight. This is equivalent to five free .50 caliber machine guns and 5300 rounds of ammunition, and to seven guns and 7400 rounds of ammunition, respectively.

Component (s).- Bomb load is 5000 pounds unless otherwise specified.

Component (t).- Fuel weighs 6.0 pounds per gallon.

Component (u).- Oil weighs 7.5 pounds per gallon and the ratio of gallons of oil to gallons of fuel is 0.009.

It should be noted that the calculations of this paper required extrapolation of some of the weight curves, and the maximum extrapolation value of the ordinates as a percent of the ordinate at the abscissa of the last data point are listed below:

Figure	Subject	Extrapolated	
		$W_G \leqq 200,000$ lb	$W_G = 300,000$ lb
9	Wing weight	136	158
14	Tail weight	234	344
17	Hydraulic, etc.	160	240
18	Cabin pressurizing	160	240
19	Surface controls	116	138
20	Anti-icing	106	166

Aerodynamics

Following a method similar to that used in the estimation of structural weights, the aerodynamic characteristics of the various bomber configurations studied were arrived at by summing the separate aerodynamic characteristics of the wing, nacelles, fuselage, and tail, and assuming that the interference drag resulting from the assembly of these components could be minimized to a negligible quantity by proper design. It was also assumed that the drag coefficient was given by

$$C_D = C_{D_o} + \frac{C_L^2}{\pi A R e} + C_{D_M}$$

where

$$C_{D_o} = C_{D_{o_w}} + \frac{S_n}{S_w} C_{D_{o_n}} + \frac{S_f}{S_w} C_{D_{o_f}} + \frac{S_t}{S_w} C_{D_{o_t}}$$

$$C_{D_M} = C_{D_{M_w}} + \frac{S_n}{S_w} C_{D_{M_n}} + \frac{S_f}{S_w} C_{D_{M_f}} + \frac{S_t}{S_w} C_{D_{M_t}}$$

$$\frac{1}{e} = \left(\frac{1}{e}\right)_w + \left(\frac{1}{e}\right)_n + \left(\frac{1}{e}\right)_f + \left(\frac{1}{e}\right)_t$$

The aerodynamic characteristics of the separate components were arrived at, whenever possible, by analysis of experimental data, as explained in the following paragraphs.

Wing.- The profile-drag coefficients for the wings of all the bombers calculated is 0.0065. No account of the variation of profile drag with wing thickness or plan form was made. The factor e_w is 0.85 in all cases and the aspect ratio is varied as explained in appendix C. The $C_{D_{M_w}}$ is shown in figures 21, 22, and 23 for wing thicknesses of 10, $12\frac{1}{2}$, and 15 percent, respectively. These curves are based upon extrapolations of high-speed tunnel tests on an NACA 65-210, AR = 9, t = 2.5, 30° sweptback wing (as shown in reference 4) to wings of different thicknesses. The extrapolations are somewhat arbitrary but are guided by a survey of experimental data on the effect of wing thickness on compressibility drag rise.

Nacelles.- The nacelle profile-drag coefficient for all nacelles is 0.045, based upon the experiments reported in reference 5. The effect of angle of attack on profile drag was considered negligible. In the absence of reliable data on three-dimensional, ducted bodies in transonic flow, the assumption was made that $C_{D_{M_n}} = C_{D_{M_W}}$.

Fuselage.- Fuselage profile-drag coefficient varies with the fineness ratio of the fuselage l_f/d_f according to the equation $C_{D_{o_f}} = 0.055\left[1 + 0.152\left(\frac{l_f}{d_f} - 6\right)\right]$, which is a good approximation, within the range of fineness ratios used, to low-speed data on bodies of revolution. The effect of angle of attack on fuselage drag was calculated by the method of reference 6, which gives the value of $1/e_f$ as a function of fuselage frontal area and wing aspect ratio. The compressibility drag rise on the fuselage $C_{D_{M_f}}$ is shown in figure 24. These curves are based on bomb-drop tests of streamline bodies of revolution, as reported in reference 7.

Tail.- Tail profile-drag coefficient is taken as 0.0065. The induced drag of the tail is accounted for by letting $\frac{1}{e_t} = 0.029$, which gives a combined induced efficiency factor for both the wing and tail $e_{w+t} = 0.83$. The coefficient $C_{D_{M_t}}$ is 70 percent of $C_{D_{M_W}}$, since the tail airfoil sections can be built thinner than the wing airfoil sections.

APPENDIX B

METHOD OF CALCULATING ENGINE DUCT FRICTION LOSSES

Because most duct tests indicate that the friction coefficient f_c is essentially a constant up to Mach numbers very close to 1, friction losses in the diffusor and tail pipe were calculated on the basis of $f_{c_1} = 0.10$ for the diffusor and $f_{c_7} = 0.04$ for the tail pipe. On this basis, an implicit relationship between p_{T_2}/p_7, ram pressure ratio with friction losses, and engine air flow w, can be developed using the equations of one-dimensional, isentropic flow:

$$\frac{p_{T_2}}{p_a} = \frac{p_{T_1}}{p_a}\left(1 - \frac{f_{ce_1}\frac{\gamma}{2}M^2}{p_{T_1}/p_a}\right)$$

where

$$\frac{p_{T_1}}{p_a} = \left(1 + \frac{\gamma - 1}{2}M^2\right)^{\frac{\gamma}{\gamma-1}}$$

and

$$f_{ce_1} = f_{c_1}\left(\frac{u_1}{u}\right)^2\left[1 + \left(1 - \left(\frac{u_2}{u}\right)^2\right)\frac{\gamma - 1}{2}M^2\right]^{\frac{1}{\gamma-1}}$$

$$\frac{W/g}{\rho u A_1} = \frac{u_1}{u}\left[1 + \left(1 - \left(\frac{u_1}{u}\right)^2\right)\frac{\gamma - 1}{2}M^2\right]^{\frac{1}{\gamma-1}}$$

Using the above equations, and knowing the altitude, airplane speed, diffusor inlet area, and engine air flow, p_{T_2}/p_a may be computed. p_{T_2}/p_7 is then obtained in the following manner

$$p_7 = p_a + f_{c_7}q_7$$

(assuming a constant velocity in the tailpipe) and since, by definition, the gross jet thrust F_g is the momentum flux of the exhaust gases out of the tail pipe,

$$F_g = \rho_7 u_7 A_7 u_7 = \rho_7 u_7^2 A_7 = 2A_7 q_7$$

or

$$q_7 = \frac{F_g}{2A_7}$$

Therefore

$$p_7 = p_a + \frac{f_{c_7}}{2A_7} F_g$$

Consequently, with the additional knowledge of the tail-pipe area and engine gross thrust the ratio p_{T_2}/p_7 can be computed.

From the foregoing, and from the consideration that increasing ram pressure ratio increases shaft power, it is seen that tail-pipe losses decrease the shaft power. If, however, the assumption is made that available energy in the gases immediately ahead of the turbine is independent of the tail-pipe losses and that all of this energy is converted to shaft power and kinetic energy of the exhaust gas regardless of tail-pipe losses, then any decrease in shaft power due to lowering of the ram pressure ratio from p_{T_2}/p_a to p_{T_2}/p_7 must be accompanied by an increase in kinetic energy of the exhaust gas. That is:

$$550(SHP)_{nl} + \frac{1}{2}mu_7^2 = 550(SHP)_{wl} + \frac{1}{2}m(u_7 + \Delta u_7)^2 \tag{1}$$

and

$$(F_g)_{nl} = mu_7 \tag{2}$$

$$(F_g)_{wl} = m(u_7 + \Delta u_7) \tag{3}$$

Combining equations (1), (2), and (3) and making the assumption that the mass flow of the exhaust gases is equal to the engine mass air flow (neglecting the fuel added),

$$1100\frac{W}{g}\left[(SHP)_{nl} - (SHP)_{wl}\right] + \left(F_g\right)_{nl}^2 = \left(F_g\right)_{wl}^2 \tag{4}$$

The tail-pipe losses also constitute a drag on the airplane, equal to the difference in total pressure between the entrance and exit to the tail pipe times the tail-pipe cross-sectional area (assuming a constant velocity throughout the pipe).

$$\text{Drag on tail pipe} = f_{c_7}A_7\bar{q}_7$$

and since

$$q_7 = \frac{1}{2}\rho_7 u_7^2 = \frac{\frac{1}{2}F_g}{A_7}$$

the drag can be written

$$\text{Drag on tail pipe} = \frac{1}{2}f_{c_7}F_g$$

Therefore, the net jet thrust of the engine is given by:

$$F_N = \left(F_g\right)_{wl} - \frac{W}{g}u - \frac{f_{c_7}\left(F_g\right)_{wl}}{2} \tag{5}$$

where $\left(F_g\right)_{wl}$ is obtained from equation (4). However, it was found, during the calculations, that the net thrust computed using equation (5) so closely approximated the net thrust calculated using the equation

$$F_N = \left(F_g\right)_{nl} - \frac{W}{g}u \tag{6}$$

for all typical operating conditions, that the error involved in using equation (6) to compute net thrust was negligible and consequently (6) was used throughout the remainder of the calculations.

APPENDIX C

APPROXIMATION USED IN CALCULATING SPECIFIC FUEL CONSUMPTION AT REDUCED POWER SETTINGS

Using the general performance curves given in the engine specifications (reference 2) for the XT35-W-3 turbine-propeller engine, a chart of specific fuel consumption c against $SHP/S\sqrt{\theta}$ for various turbine inlet temperatures can be produced for a given ram pressure ratio. Figure 25 is such a chart. Notice that the curve for each turbine inlet temperature reaches a minimum and then rises. The envelope of these curves is therefore a curve of minimum specific fuel consumption for any particular value of shaft power being used. Slightly above this minimum line is the surge line, which represents the line of maximum satisfactory pumping ability of the compressor. The surge line is also a line of maximum allowable T_3/θ up to rated T_3/θ, where rated $T_3 = 2160^\circ$ Fabs. Since the surge line is presented on all the general performance curves given in the engine specifications, considerably less labor is involved in using it than in using the line of minimum c. For this reason, and because the error involved is less than 2.2 percent everywhere within the range of values of $SHP/S\sqrt{\theta}$ used, the surge line was used for calculating engine performance at any operating power less than that requiring rated T_3/θ. A typical graph of specific fuel consumption against percent rated power is shown in figure 1.

APPENDIX D

METHOD OF SELECTING ASPECT RATIOS

Examination of the range equation,

$$\text{Range} = 375 \int_{W_E}^{W_G} \frac{L}{D} \frac{\eta_e}{c} \frac{dW}{W}$$

(where $L = w$) shows that, if η_e/c is constant, maximum range occurs at maximum L/D, which, if the drag coefficient is given by

$$C_D = C_{D_o} + \frac{C_L{}^2}{\pi ARe}$$

is equal to

$$\left(\frac{L}{D}\right)_{max} = \frac{1}{2}\sqrt{\frac{\pi ARe}{C_{D_o}}}$$

which is a constant. The range is then given by the Breguet solution:

$$\text{Range} = \frac{375}{2} \frac{\eta_e}{c} \sqrt{\frac{\pi e}{C_{D_o}}} \sqrt{AR} \ln \frac{W_G}{W_E} \tag{7}$$

Now if W_F is defined as the gross weight of the bomber minus the weights of the fuel, bombs, and wing, then:

$$W_E = W_F + W_W \tag{8}$$

The sum of the tail, fuselage, engines, crew, etc. W_F can be determined independently of the aspect ratio once the gross weight, number of engines, and wing loading have been decided upon. The weight W_W, however is strongly dependent upon aspect ratio. Once the taper ratio, wing loading, wing thickness, and wing area are determined, W_W is a function of aspect ratio alone. This function can be determined

empirically, using the straight-line graph of figure 9 and the approximation $W_D = 0.136W_F$, to be

$$W_W = \frac{2.95S + \dfrac{16.07\sqrt{S_W}\left(0.9428W_G\right)}{10^6\,\tau}AR^{3/2}}{1 + \dfrac{16.07\sqrt{S_W}}{10^6\,\tau}AR^{3/2}}$$

This equation, combined with equations (7) and (8), yields an equation for range of the form

$$\text{Range} = \text{Constant}\ \sqrt{AR}\ \log \frac{\text{Constant}}{\text{Constant} + f(AR)}$$

which, differentiated and set equal to zero gives the following equation in aspect ratio:

$$0 = \log \frac{1 + E(AR)^{3/2}}{(B + C) + (EB + D)AR^{3/2}} - \frac{3(D - EC)AR^{3/2}}{(B + C) + (2EB + EC + D)AR^{3/2} + E(EB + D)AR^3}$$

where

$$B = \frac{W_G}{W_F} \qquad C = \frac{2.95}{W_G/S_W}$$

$$D = 0.9428\,\frac{16.07S_W}{10^6\tau} \qquad E = \frac{16.07S_W}{10^6\,\tau}$$

This equation must be solved for optimum aspect ratio for each particular bomber configuration. This method, while not as accurate as actually computing ranges for a number of aspect ratios to find the maximum for each case, gives a reasonable approximation and saves much labor.

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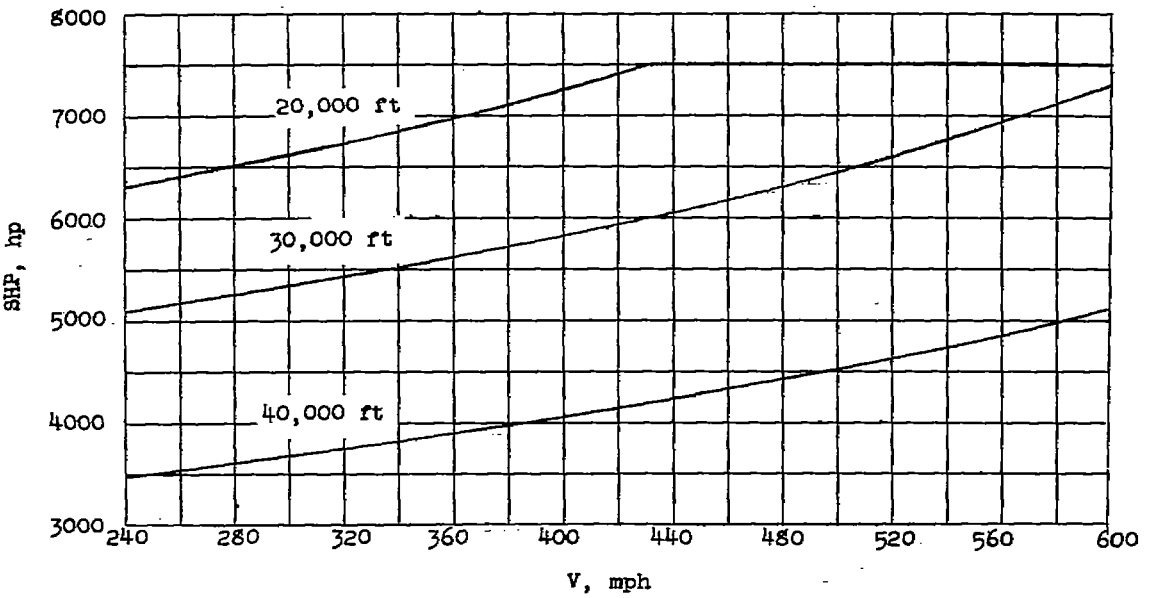
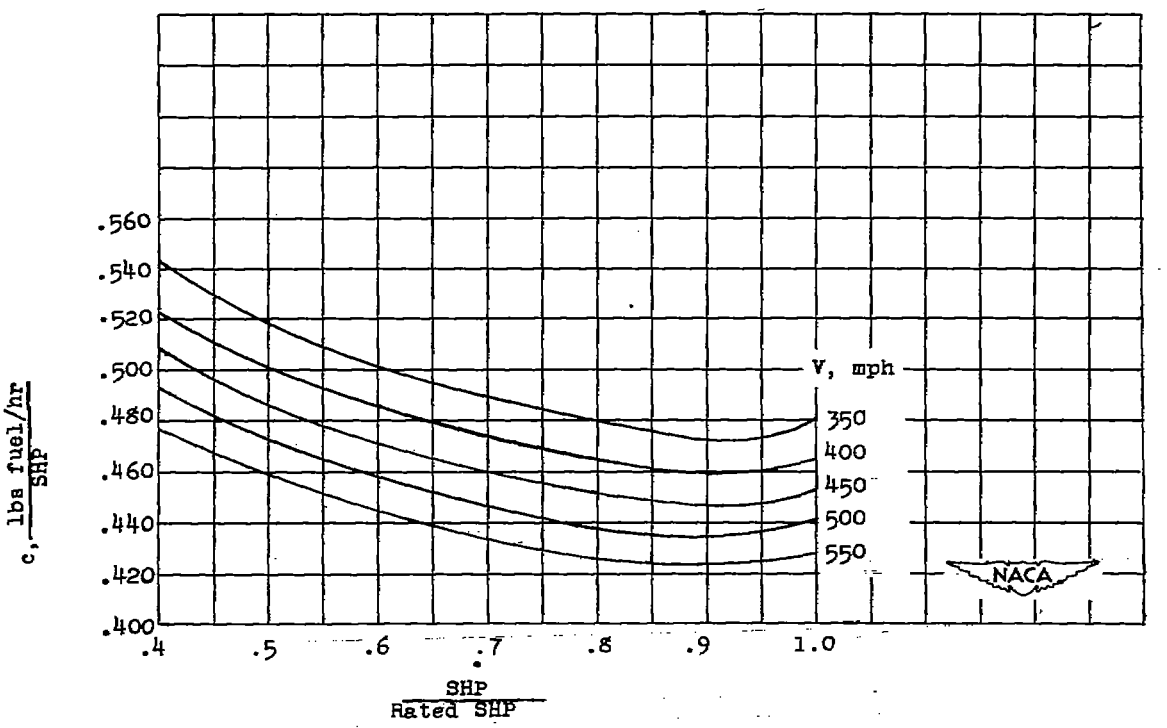


(a) Rated shaft power at various altitudes.



(b) Specific fuel consumption at 40,000-foot altitude.

Figure 1.- Engine performance curves.

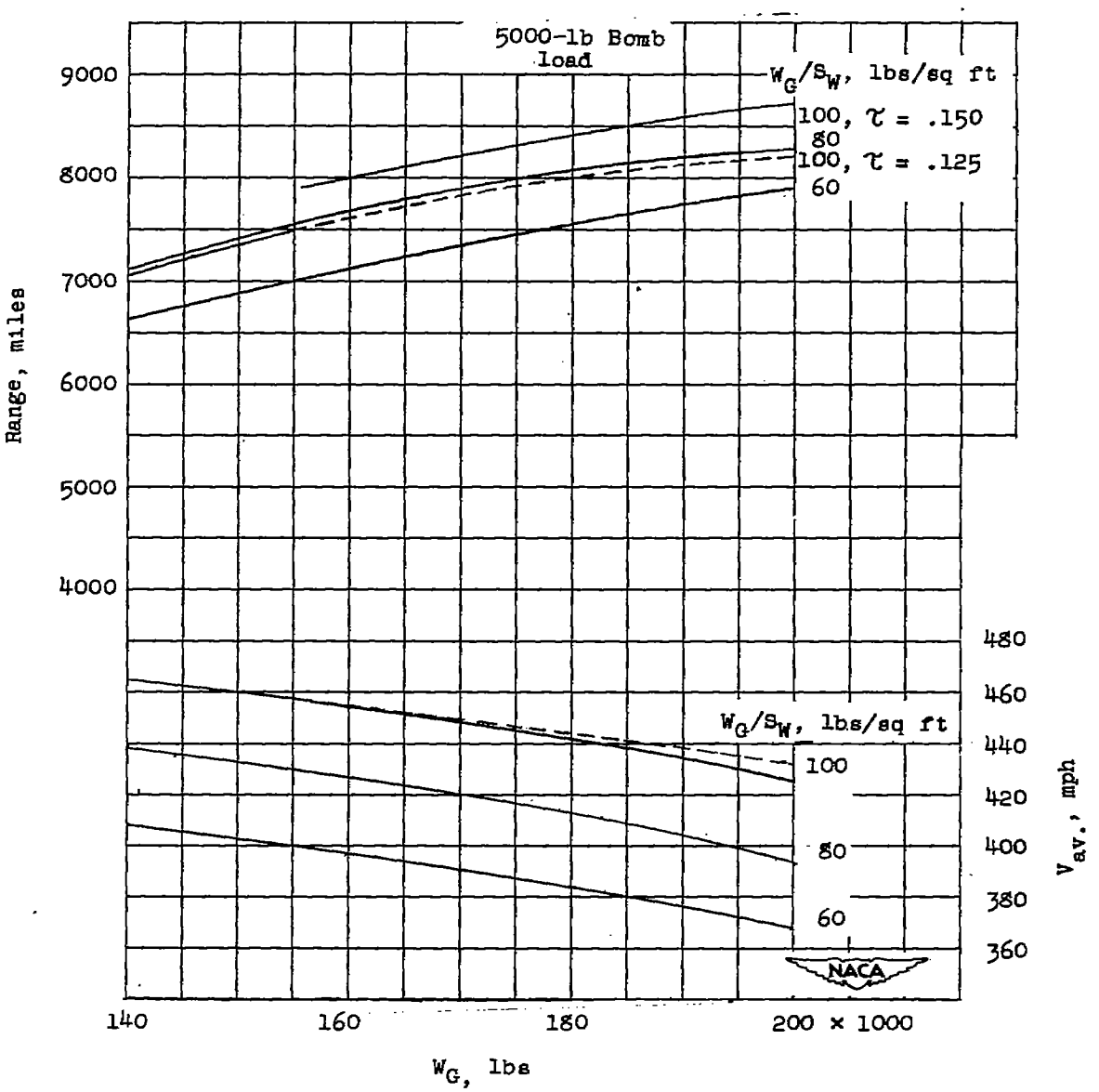


Figure 2.- Ranges and average cruising speeds of two-engine bombers.

[redacted]

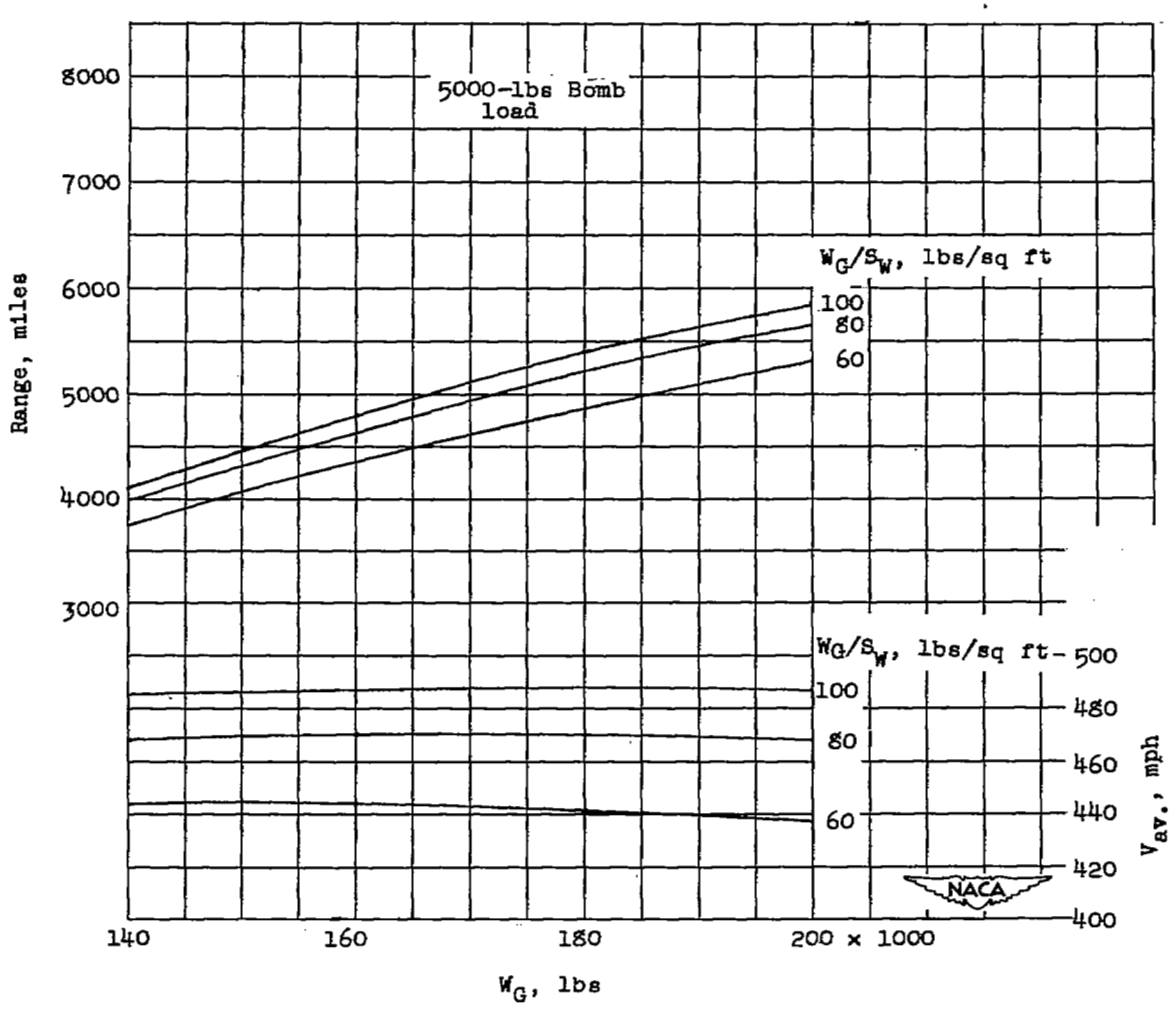


Figure 3.- Ranges and average cruising speeds of four-engine bombers.

[redacted]

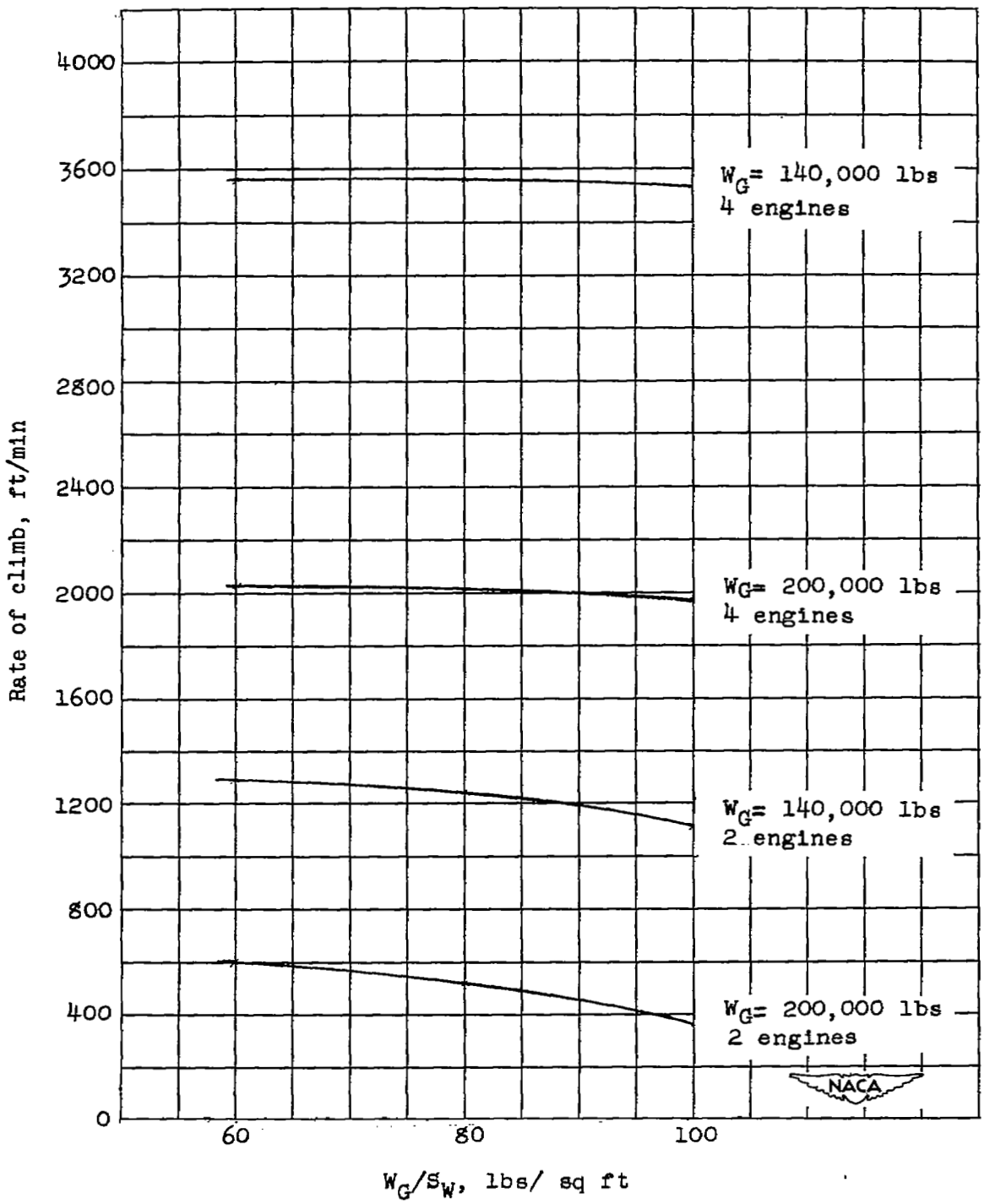


Figure 4.- Rates of climb of various bombers at full gross weight using rated power at 25,000-foot altitude.

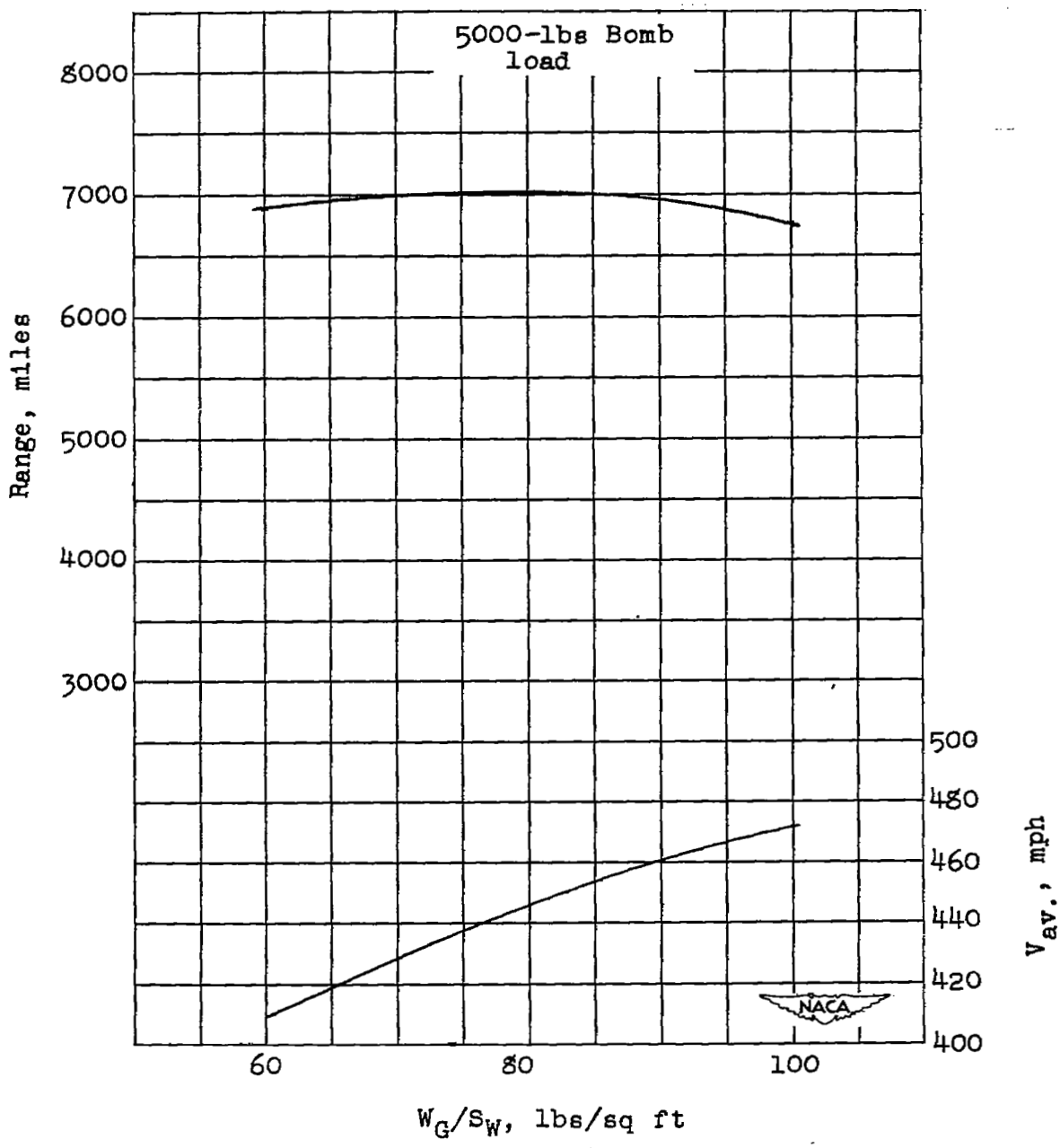


Figure 5.- Ranges and average cruising speeds of 300,000 pound gross weight, four-engine bombers.

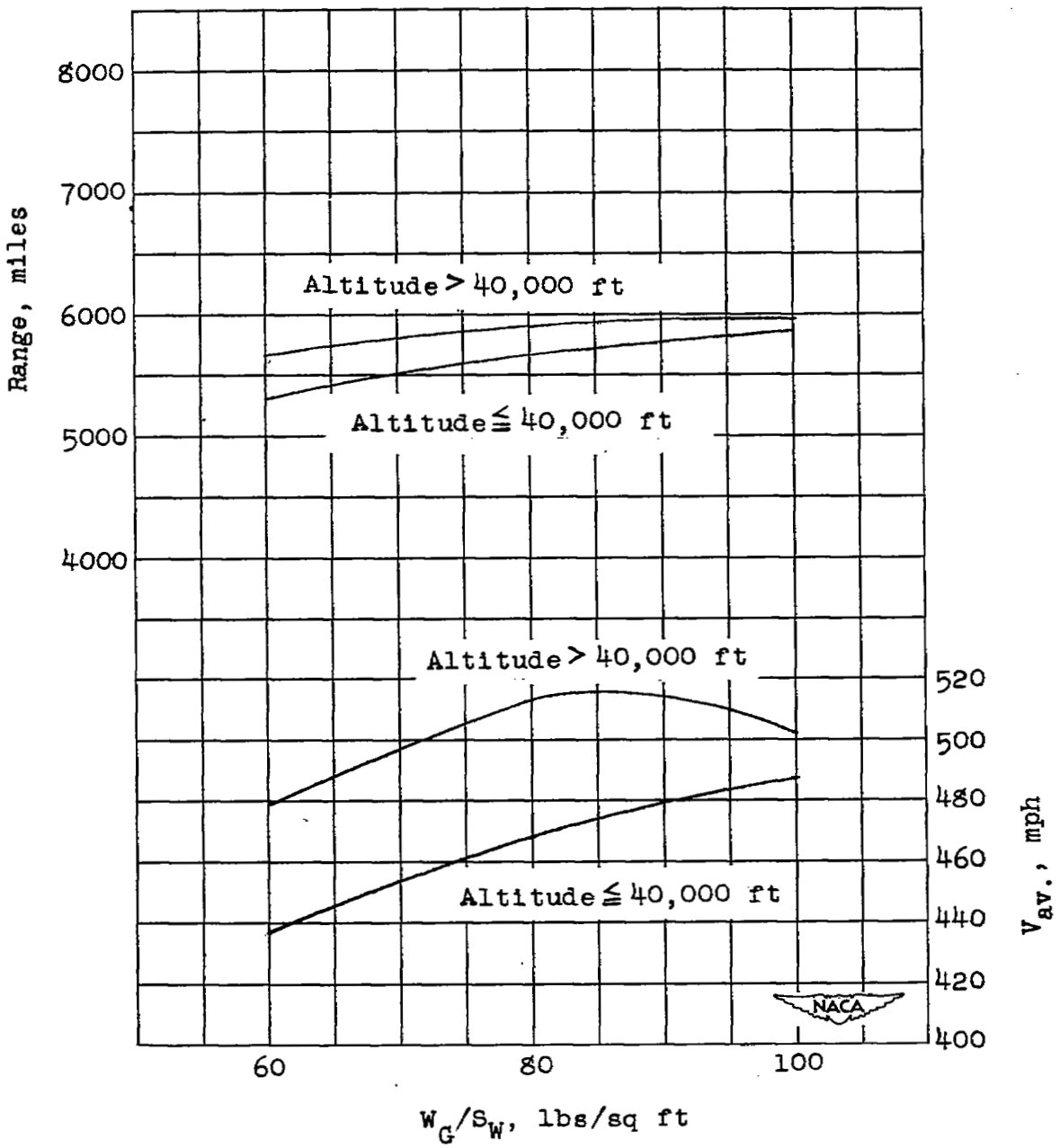
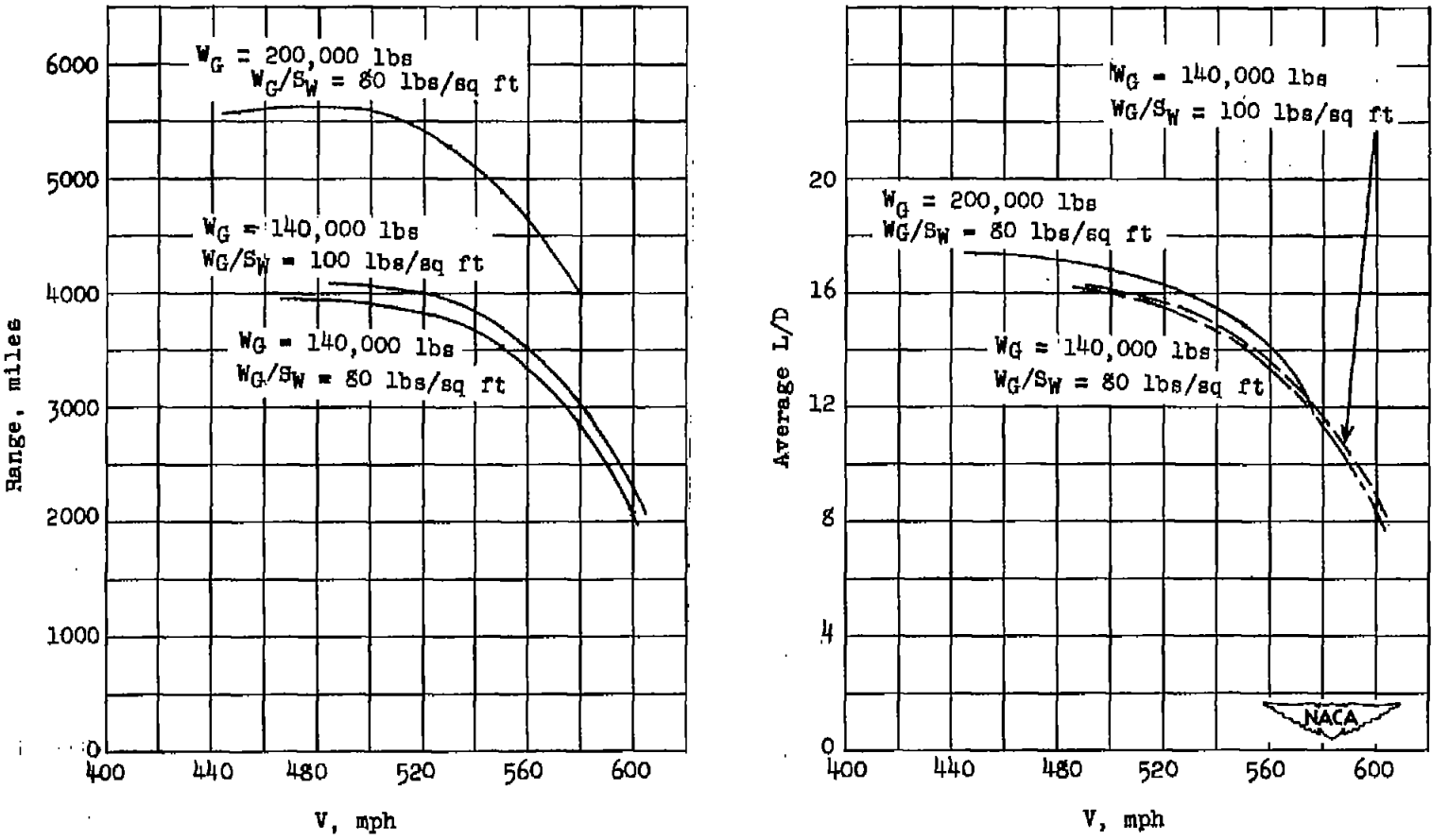


Figure 6.- Effect of the 40,000-foot-altitude limitation on the range and average cruising speeds of bombers with four engines and 200,000-pound gross weights.



(a) Effect of cruising speed on the range of three different four-engine bombers.

(b) Effect of cruising speed on the average L/D of three different four-engine bombers.

Figure 7.- Effect of cruising speed on the range and average L/D of three different four-engine bombers.

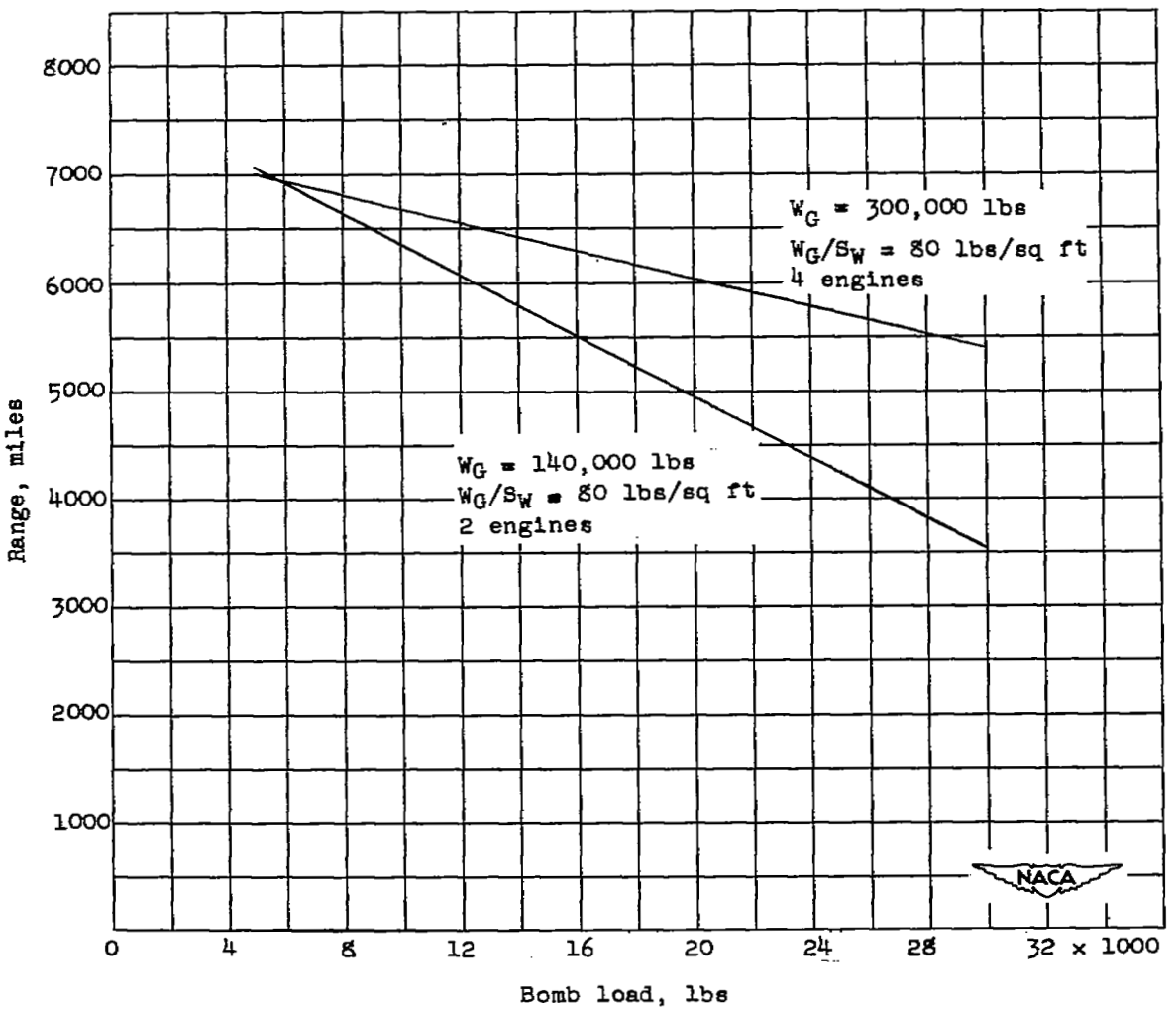


Figure 8.- Effect of bomb load on the ranges of two selected bombers.

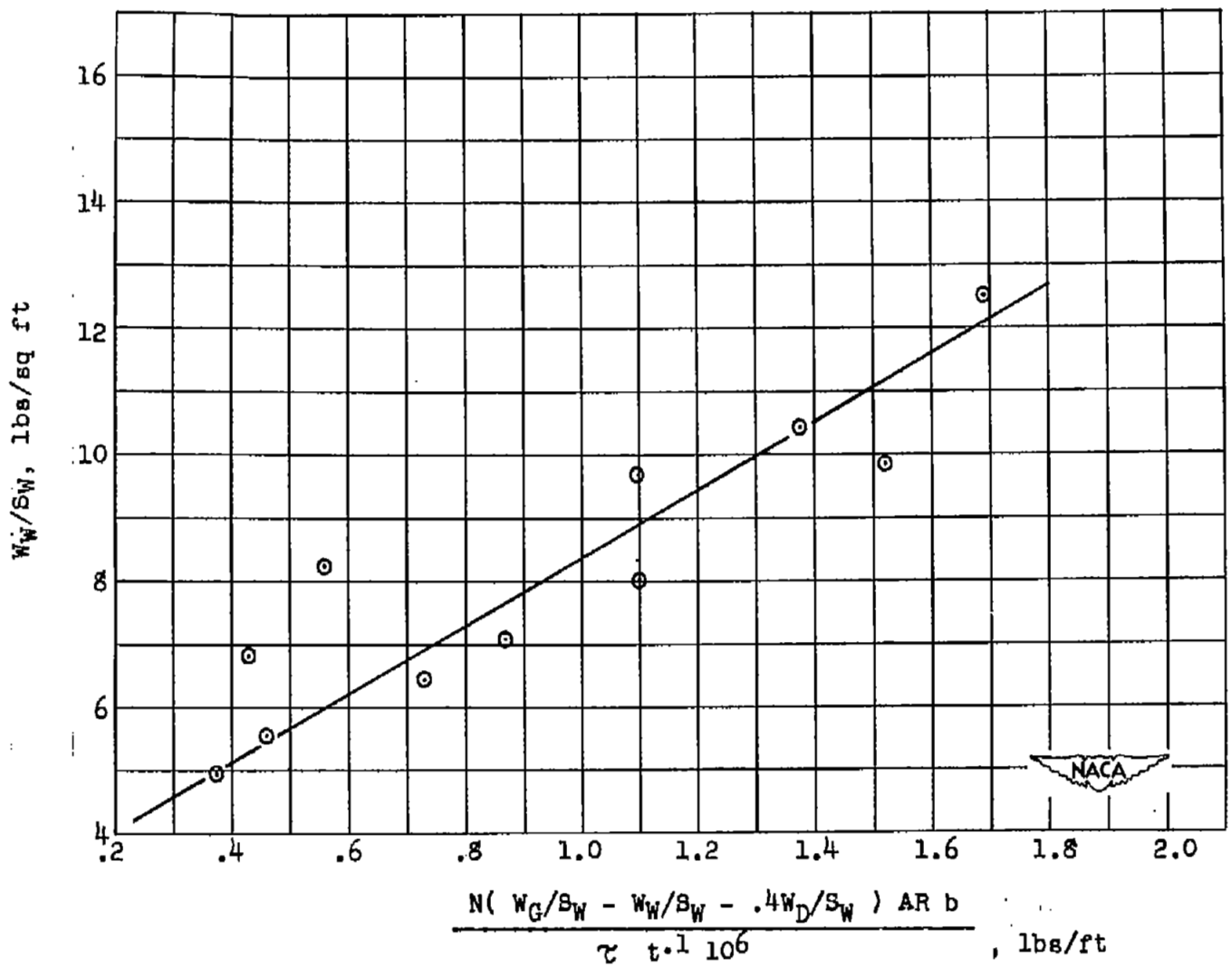


Figure 9.- Wing weights of recent military aircraft.

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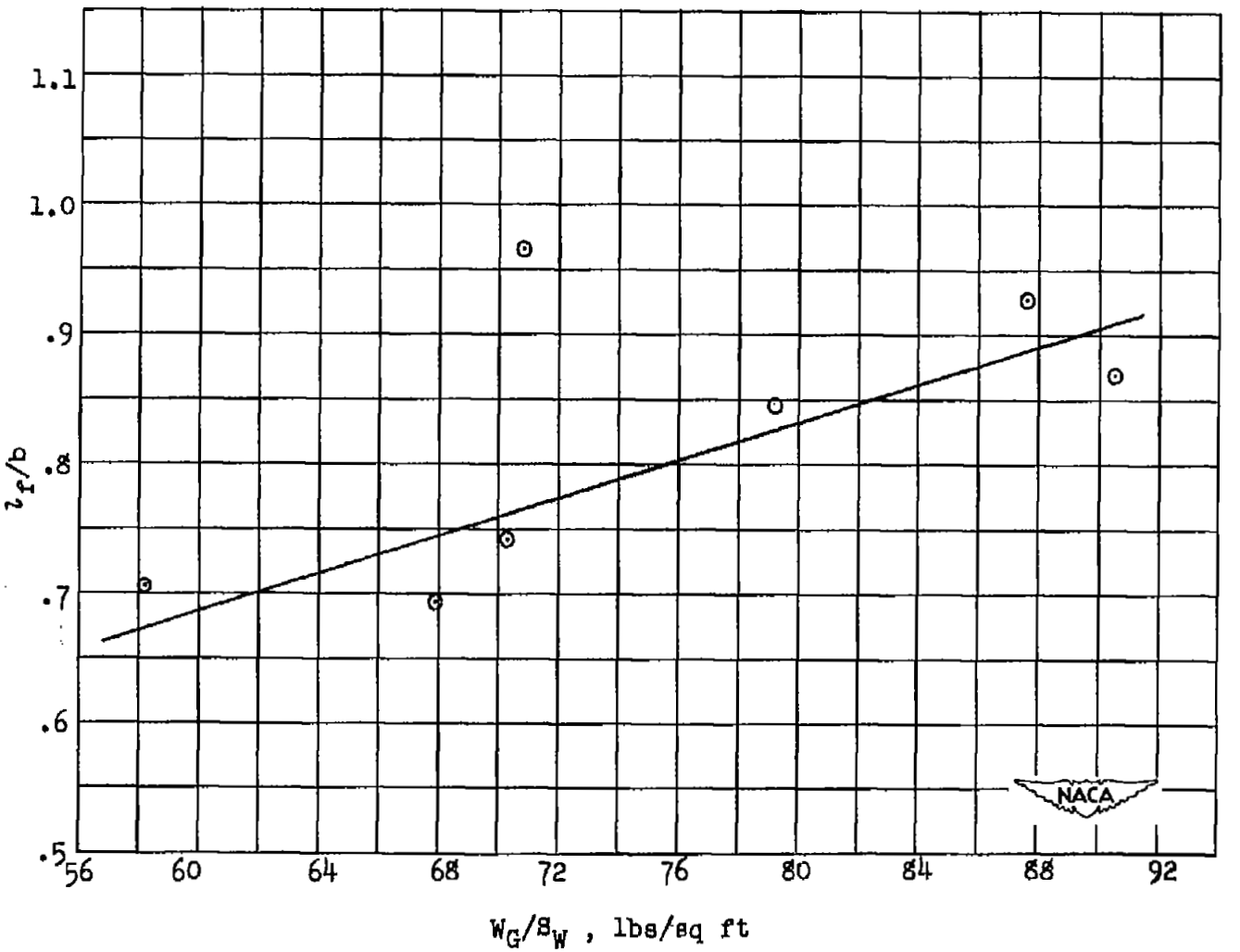

Figure 10.- Variation of length/span ratio of recent military aircraft.

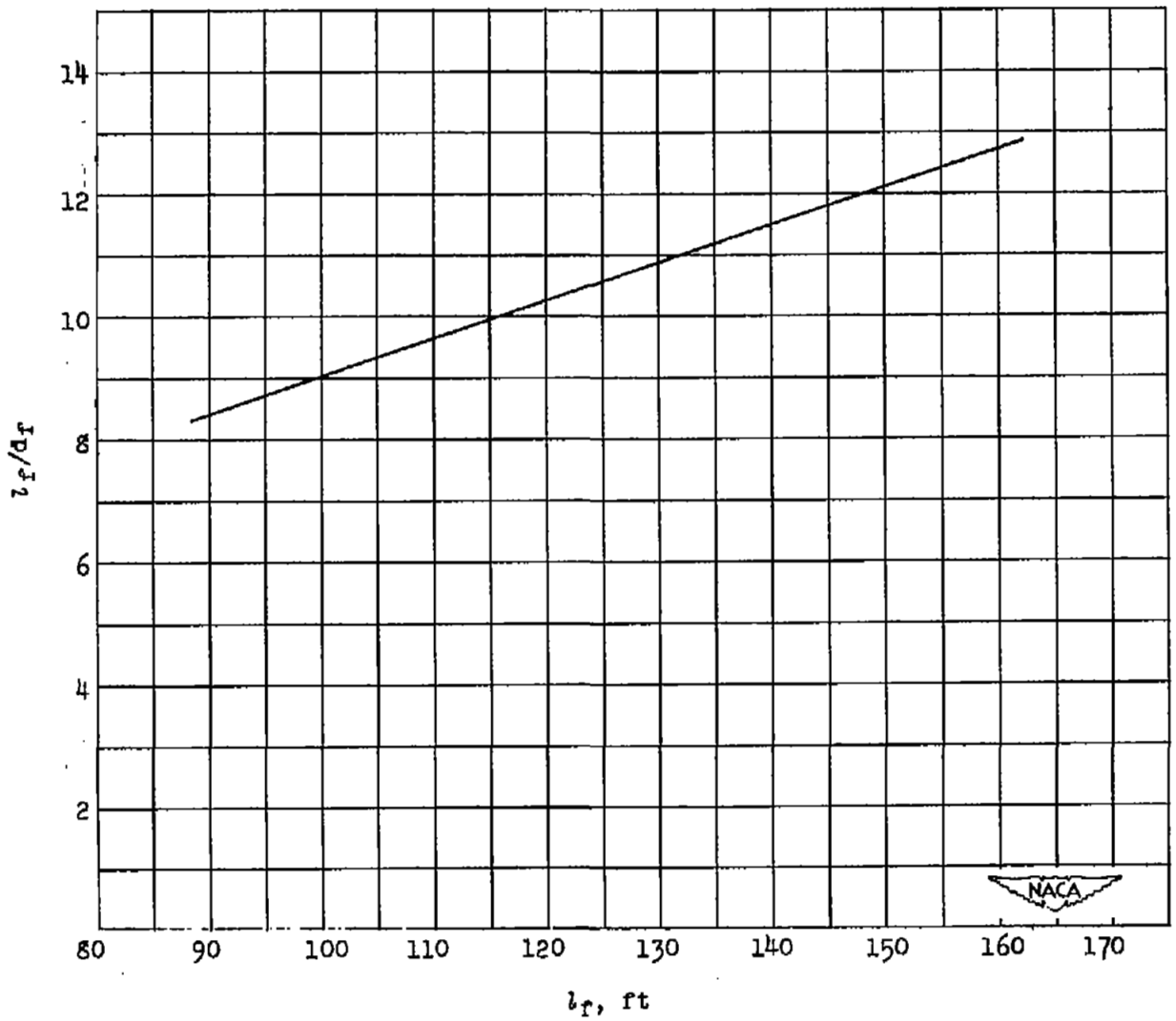

Figure 11.- Variation of fuselage fineness ratio used in calculation of fuselage sizes.

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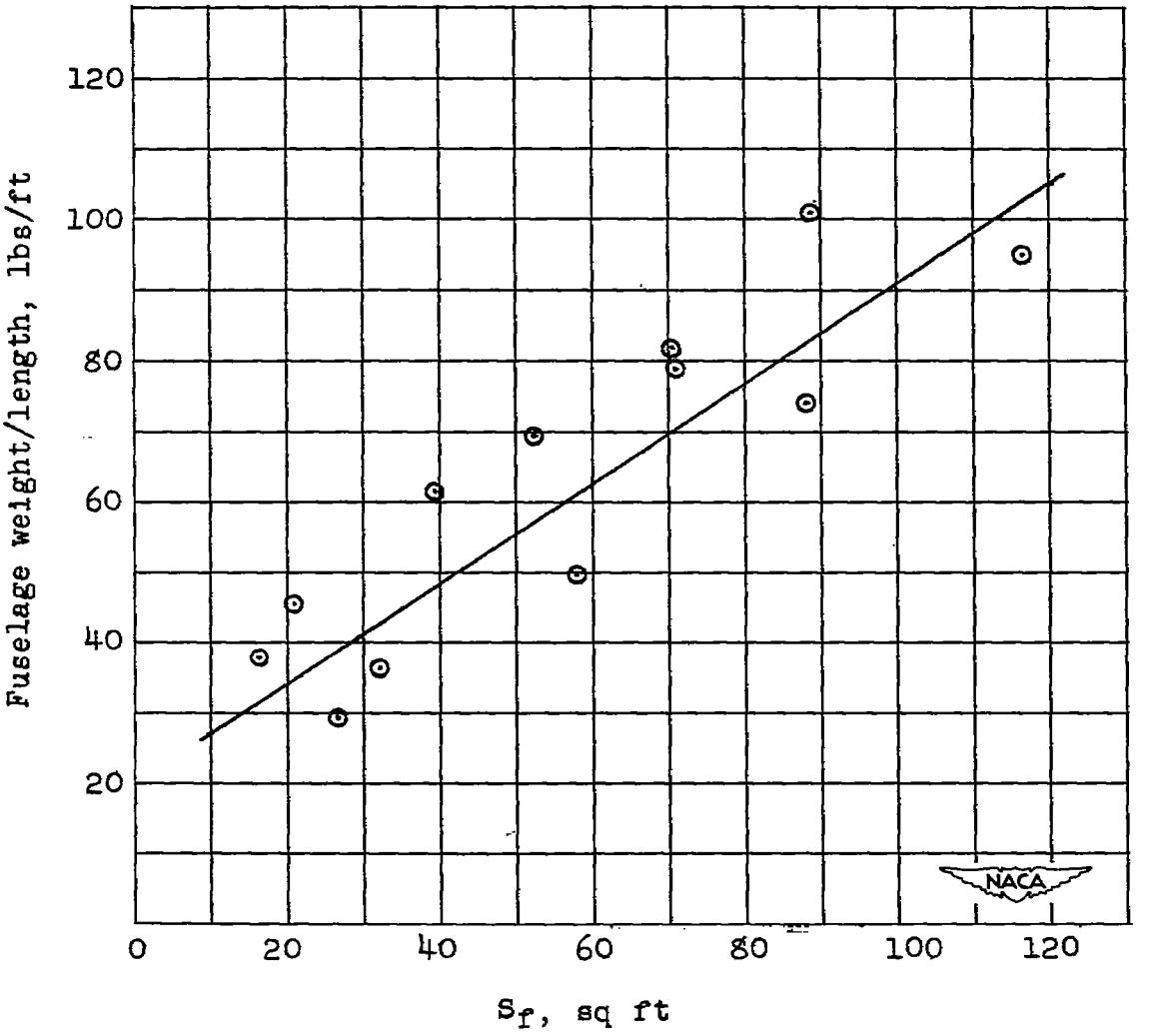

Figure 12.- Fuselage weights of recent military aircraft.

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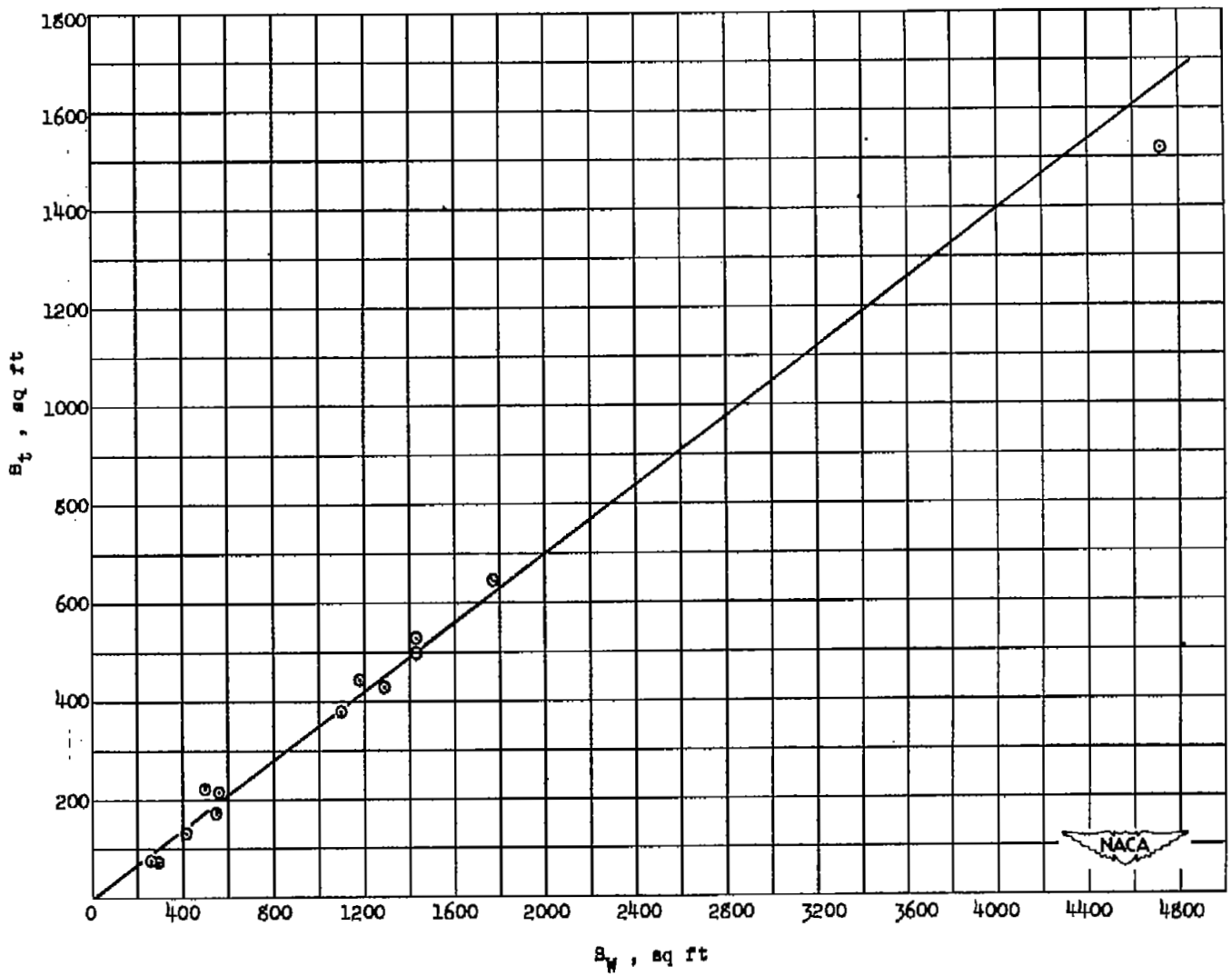

Figure 13.- Tail areas of recent military aircraft.

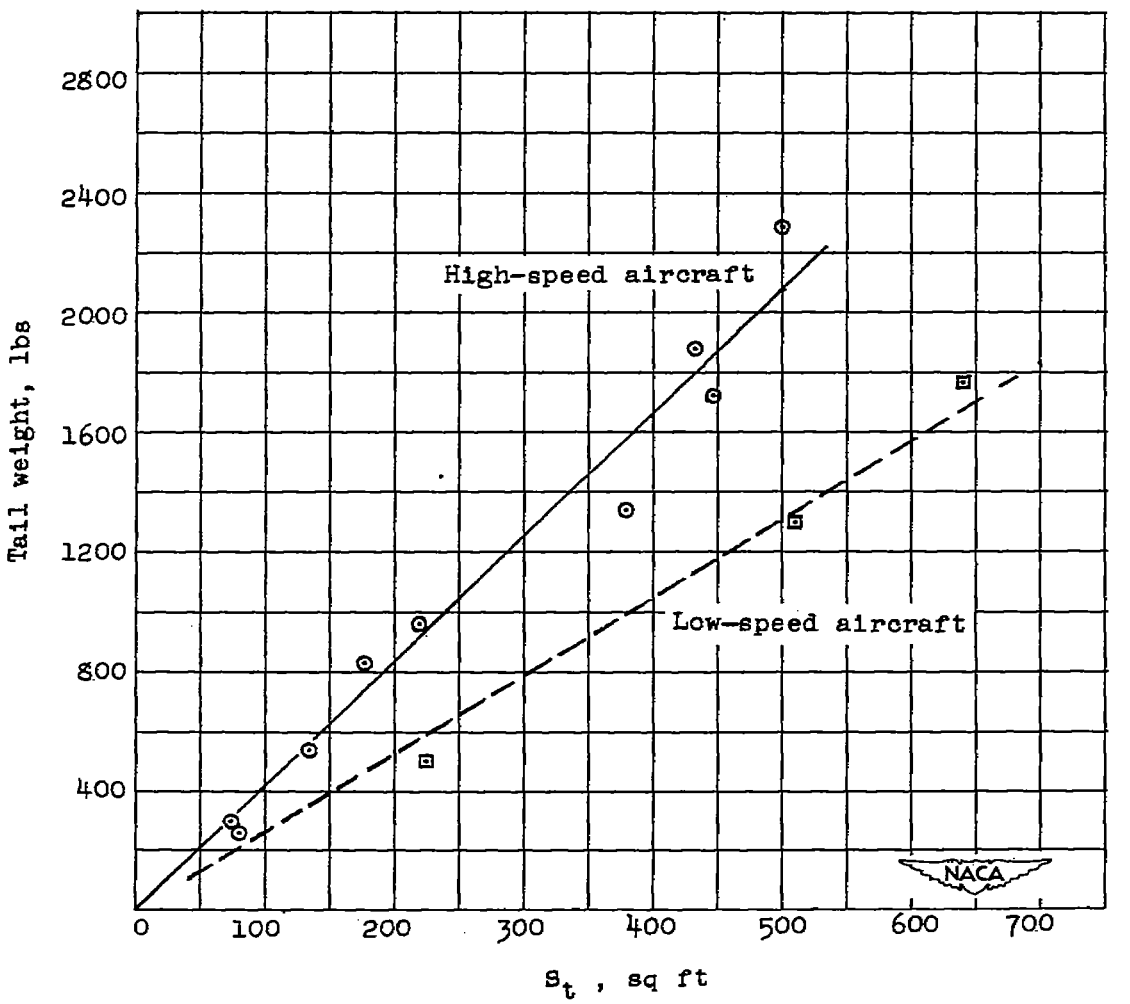

Figure 14.- Tail weights of recent military aircraft.

[redacted]

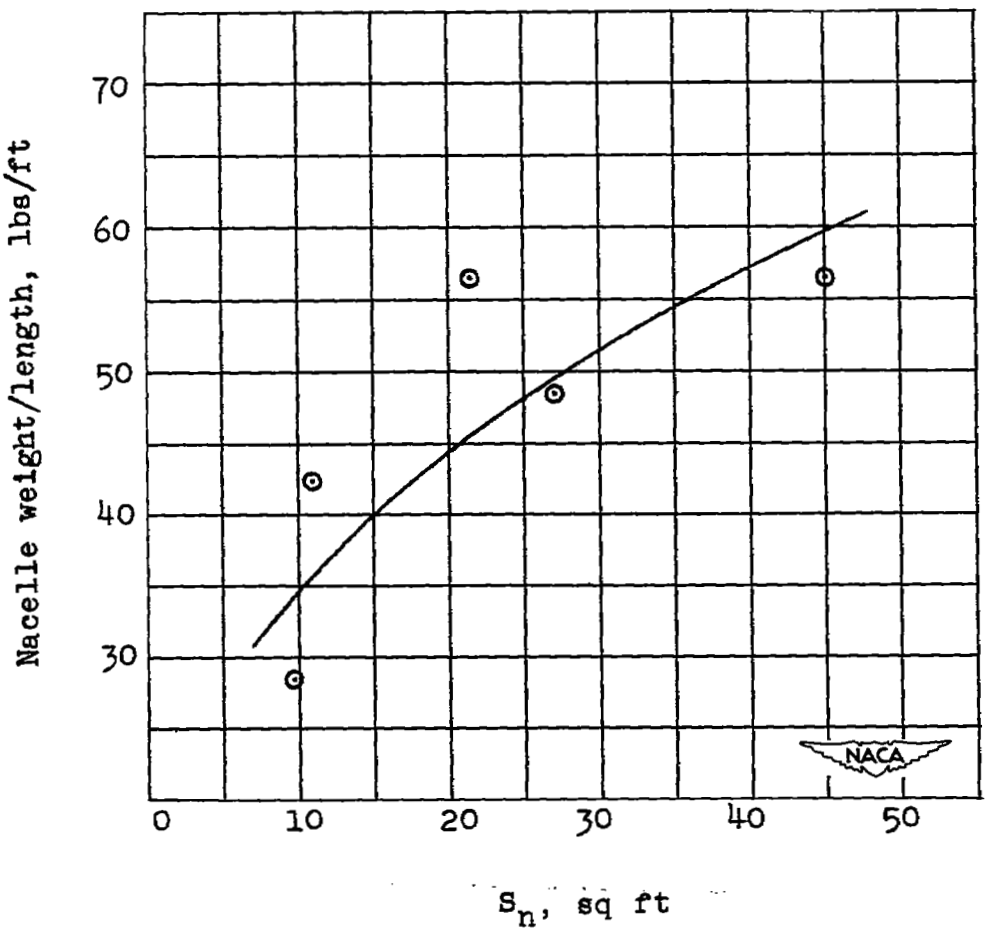

Figure 15.- Nacelle weights of recent military aircraft.

[redacted]

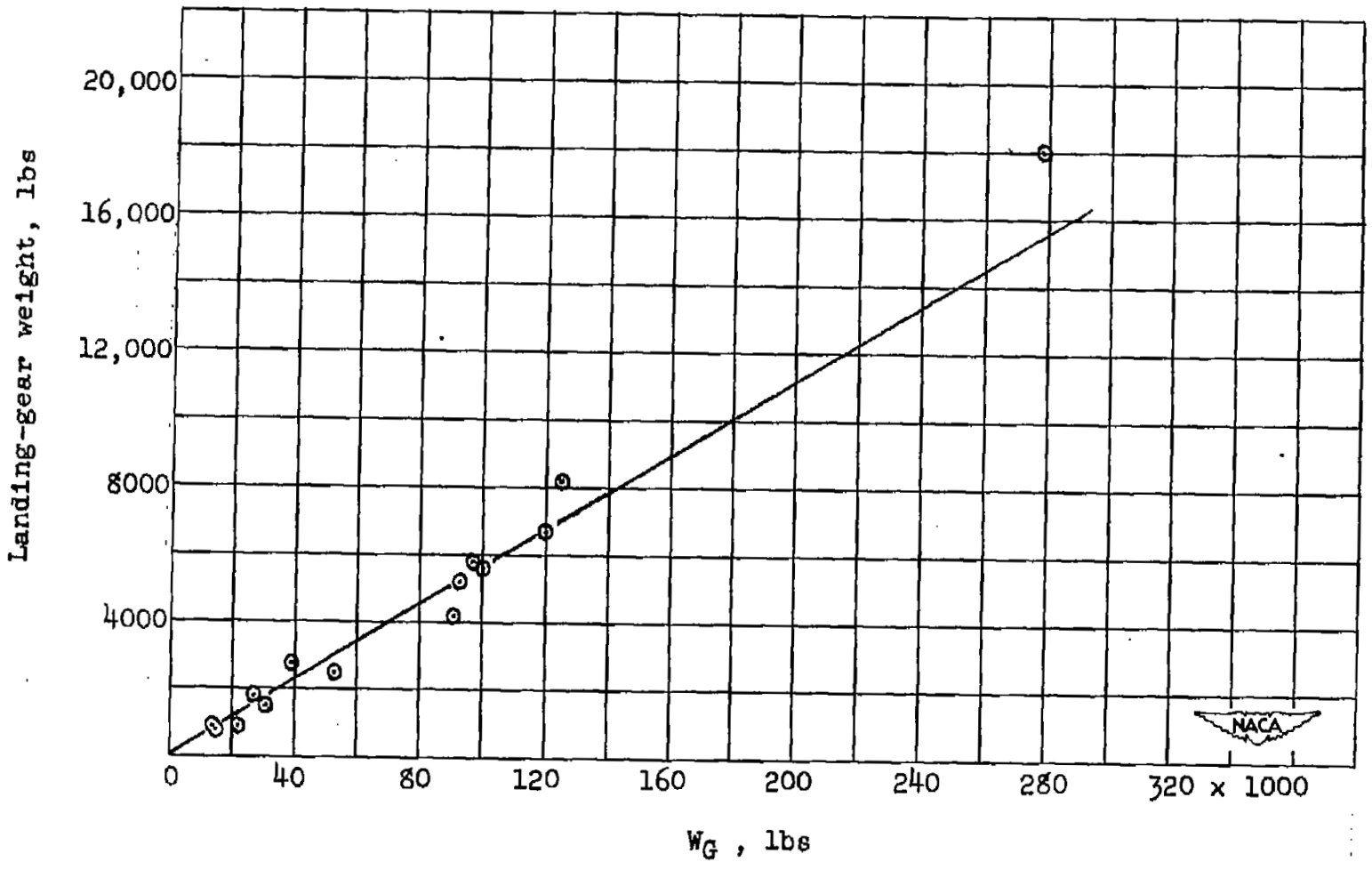

Figure 16.- Landing-gear weights of recent military aircraft.

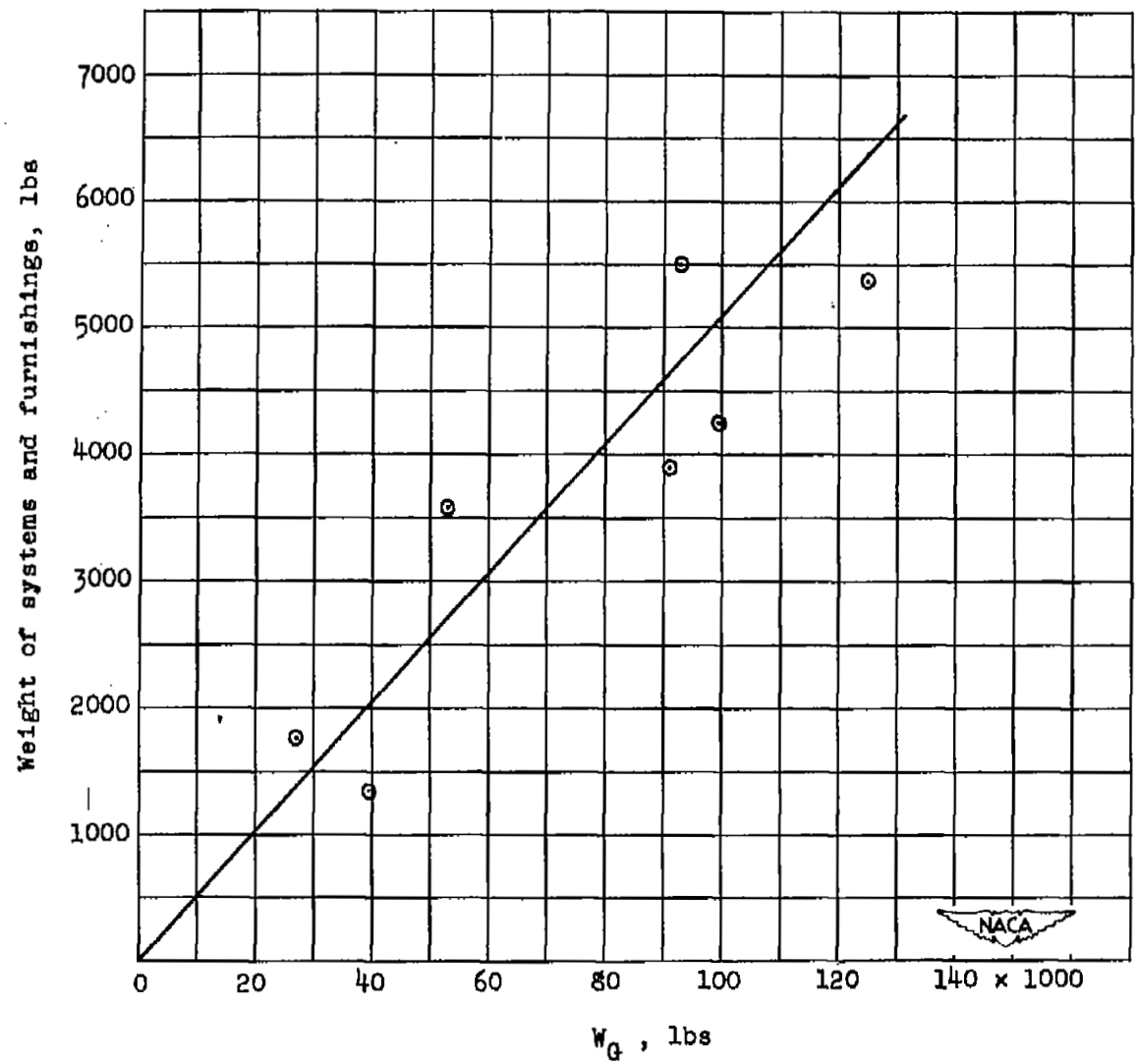

Figure 17.- Weight of hydraulic, electrical, and communication systems, and cabin furnishings of recent military aircraft.

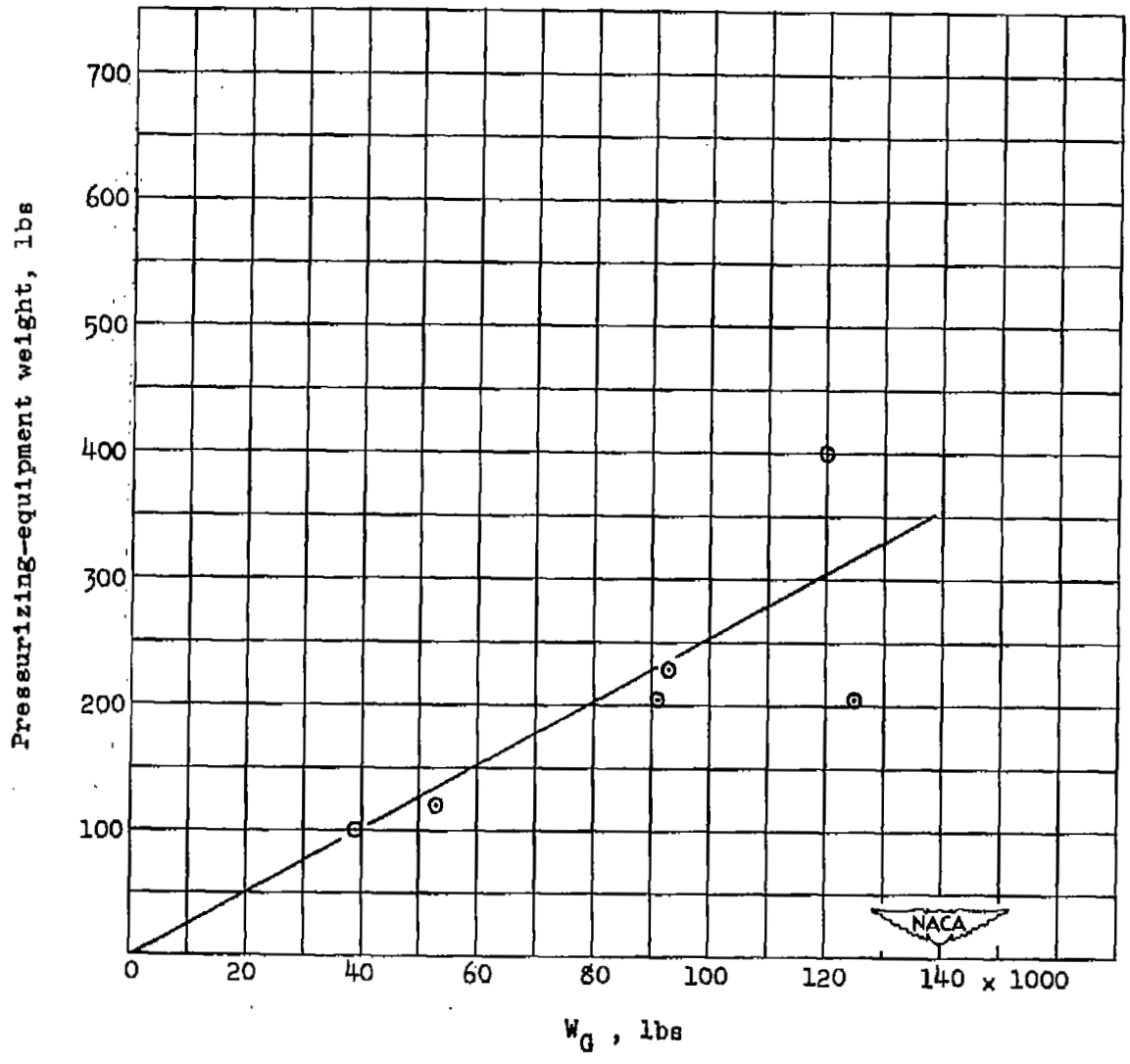

Figure 18.- Weight of cabin-pressurizing equipment of recent military aircraft.

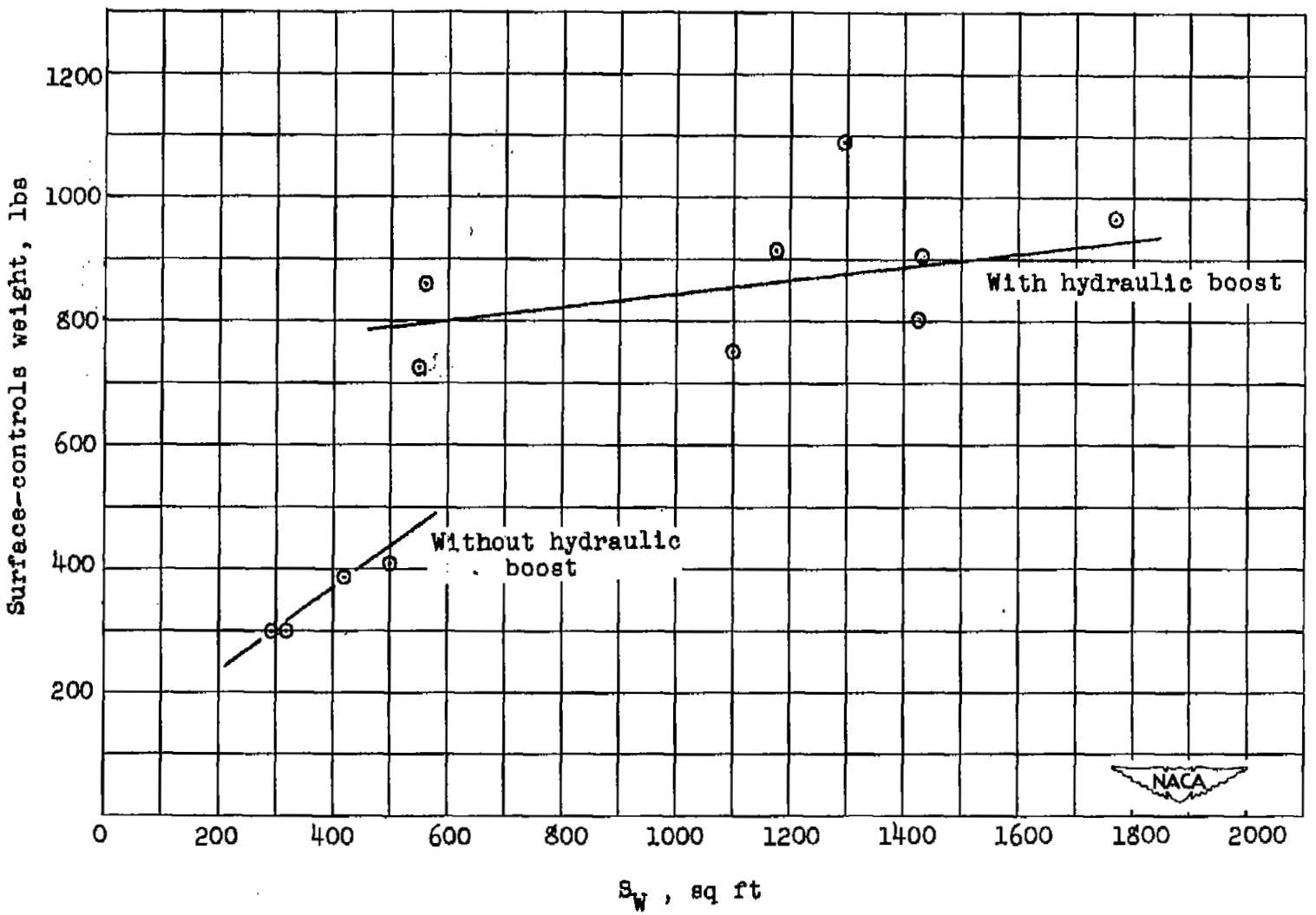

Figure 19.- Weight of surface controls of recent military aircraft.

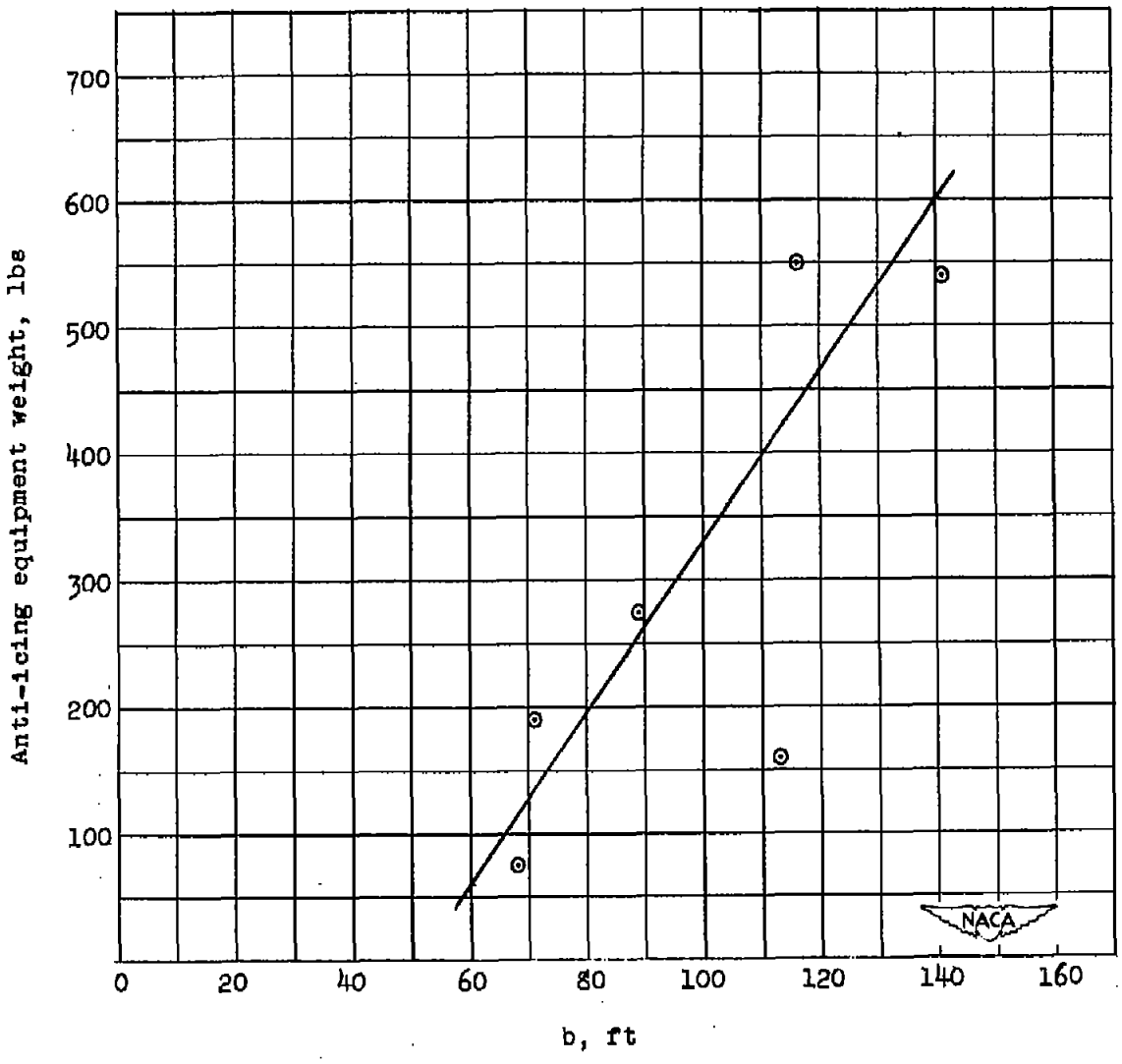

Figure 20.- Weight of anti-icing equipment of recent military aircraft.

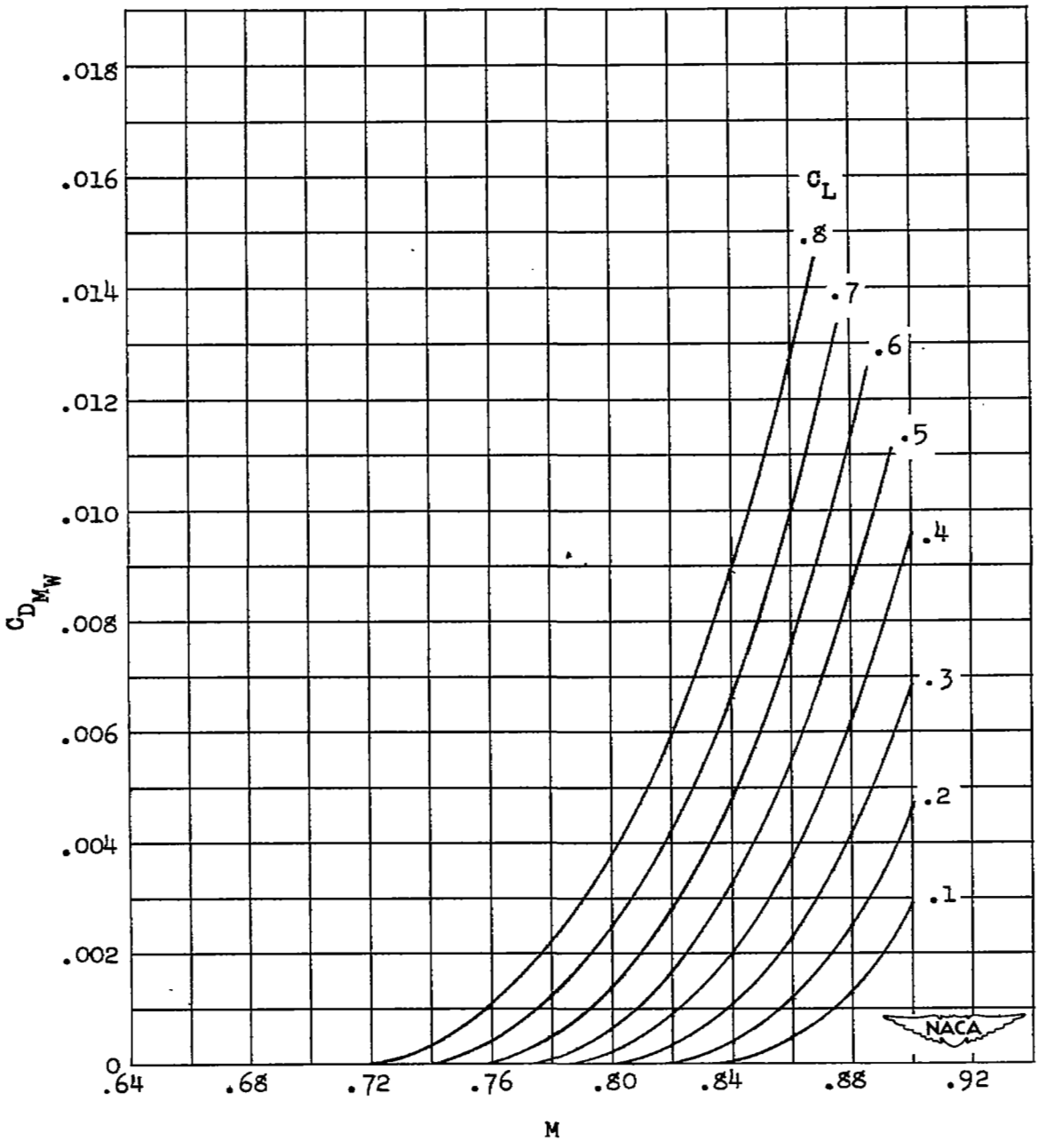

Figure 21.- Compressibility drag rise for a 10-percent-thick wing.

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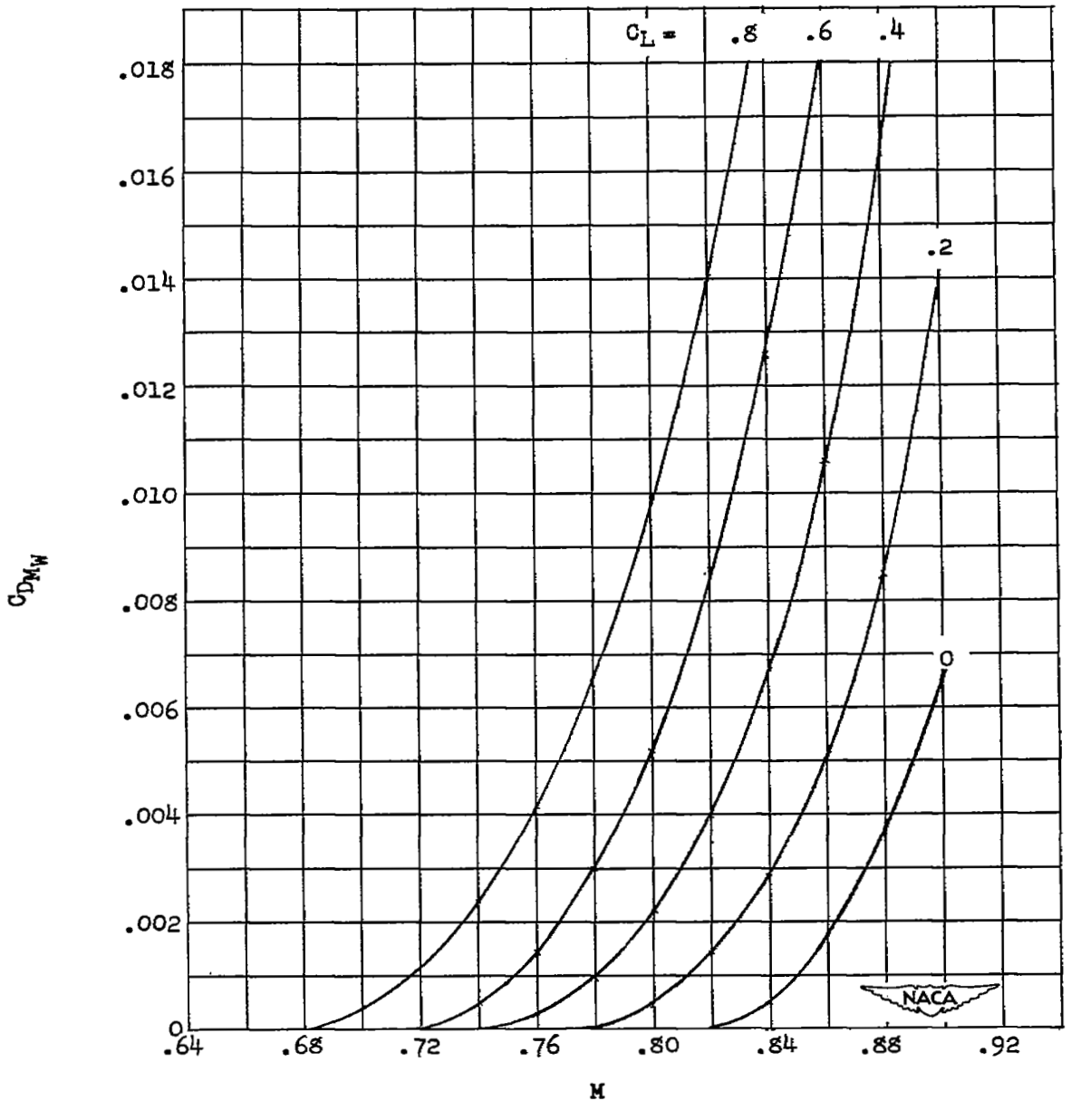

Figure 22.- Compressibility drag rise for a $12\frac{1}{2}$-percent-thick wing.

[redacted]

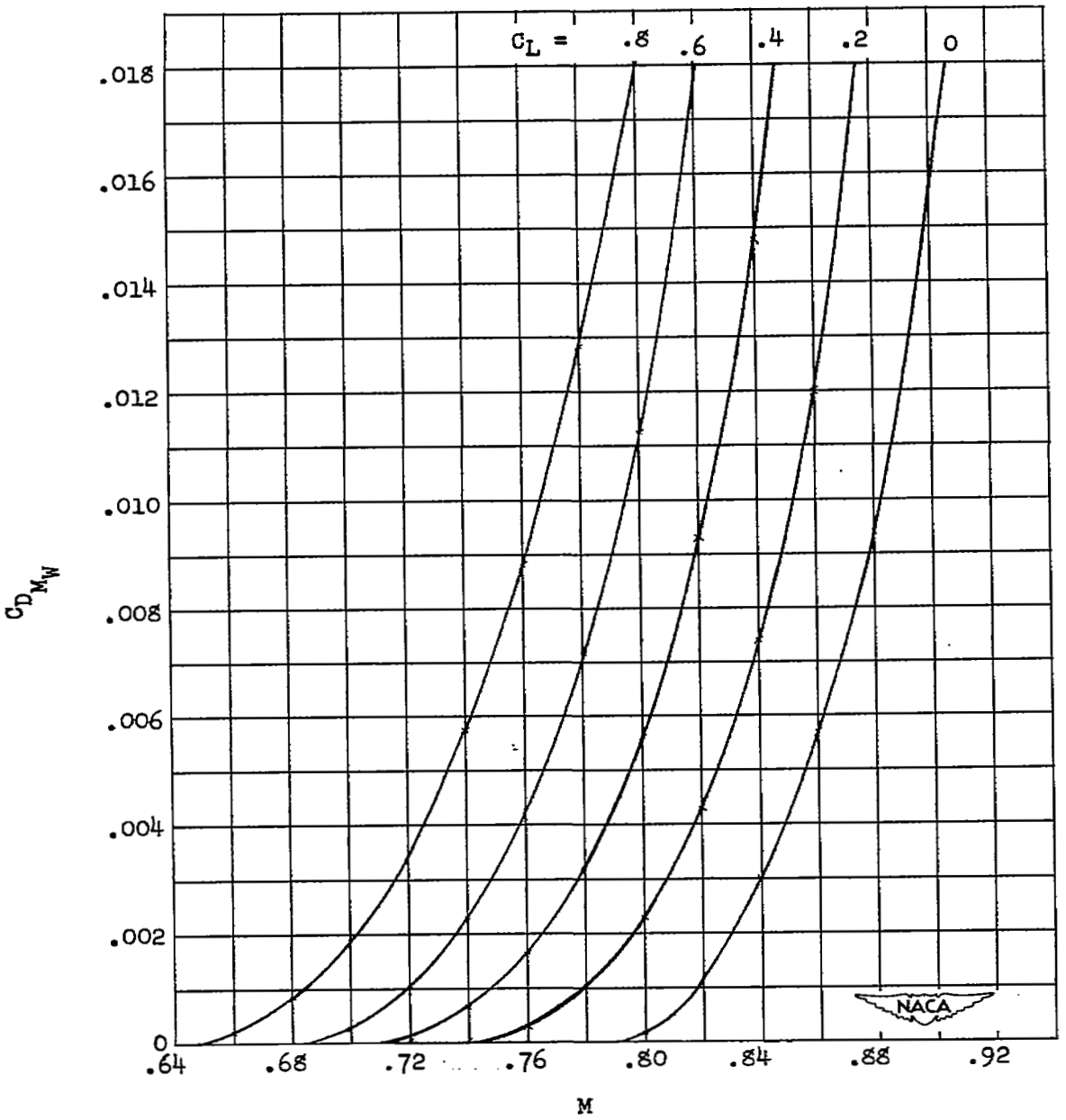

Figure 23.- Compressibility drag rise for a 15-percent-thick wing.

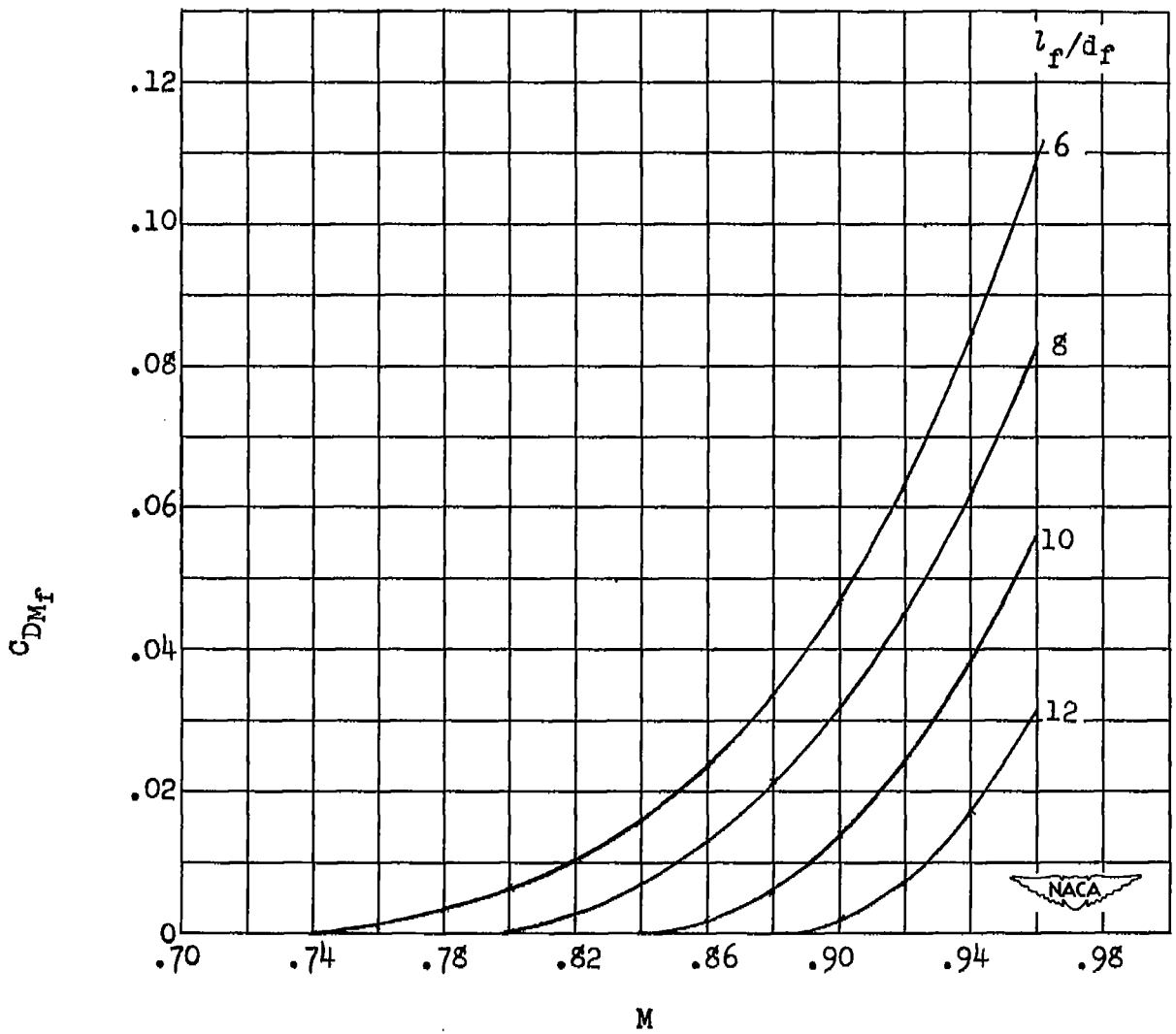

Figure 24.- Compressibility drag rise on fuselages.

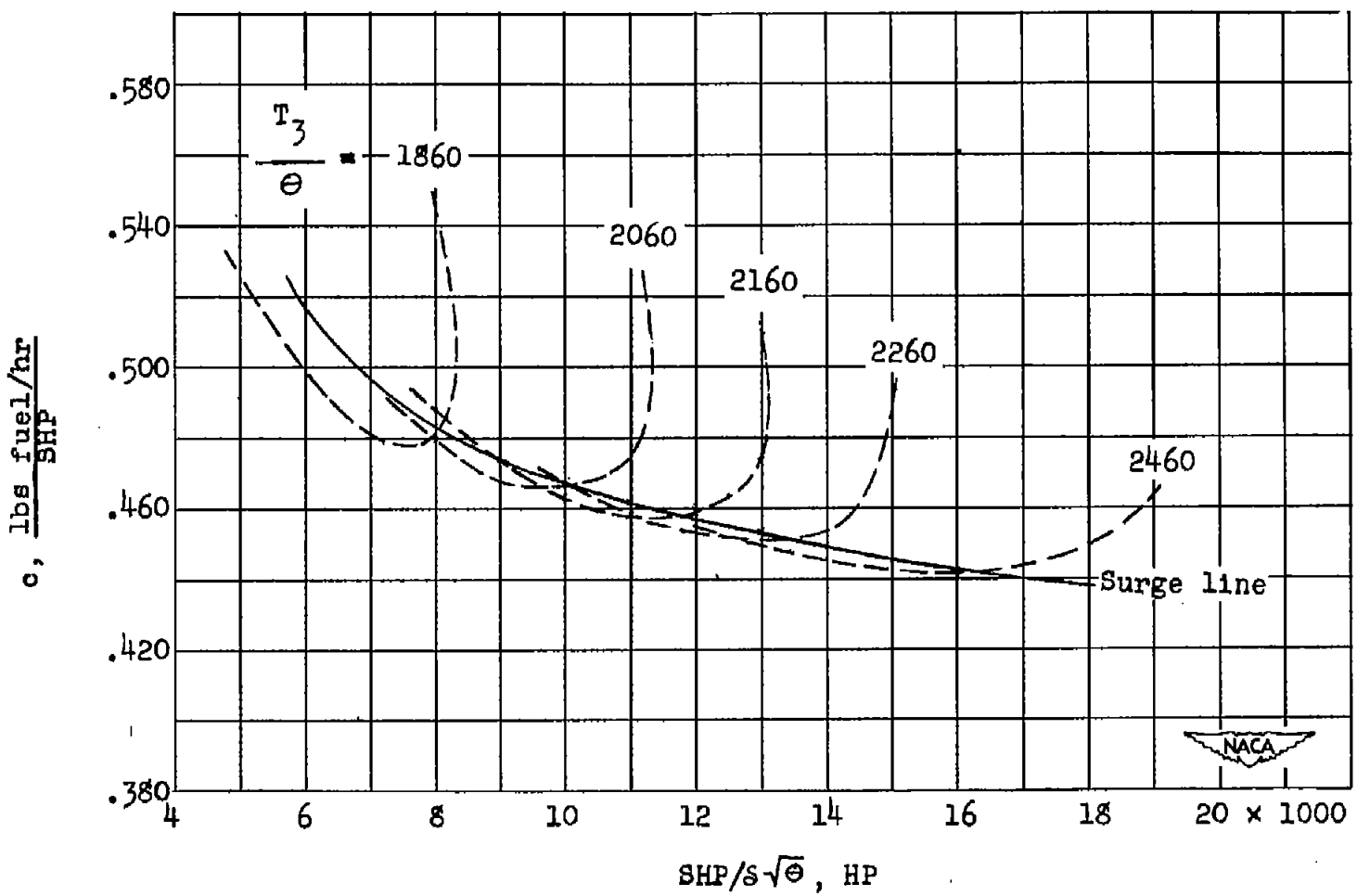

Figure 25.- Effect of shaft power and turbine inlet temperature on specific fuel consumption. Ram pressure ratio = 1.5.

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ABSTRACT

Calculations have been made to find ranges attainable by bombers of gross weights from 140,000 pounds to 300,000 pounds, powered by a typical turbine-propeller power plant. Results show the effect of wing loading, of cruising at high speeds, of operation at very high altitudes, and of carrying large bomb loads.

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