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RESEARCH MEMORANDUM

EVALUATION OF PISTON-TYPE GAS-GENERATOR

ENGINE FOR SUBSONIC TRANSPORT OPERATION

By A. F. Lietzke and Hugh M. Henneberry

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Cleveland, Ohio

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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EVALUATION OF PISTON-TYPE GAS-GENERATOR ENGINE FOR SUBSONIC TRANSPORT OPERATION

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SUMMARY

A piston-type gas-generator engine was evaluated by comparing the performance of a transport airplane powered by this engine with the performance of an airplane having the same characteristics powered by other types of engine. The engines chosen for comparison were a turbojet engine, a turbine-propeller engine, a compound engine, and a turbosupercharged reciprocating engine utilizing a variable-area exhaust jet nozzle. The comparison was made at turbine-inlet total temperatures of 1400° and 1800° F for the turbojet, turbine-propeller, and gas-generator engines. Increases in engine pressure ratio and air-handling capacity were assumed to accompany the increase in turbine-inlet temperature. The comparison is based on the pay-load ton-miles per hour of operation per ton take-off gross weight.

The relative merit of the piston-type gas-generator engine based on this comparison was found to increase as the flight range increased. The performance of the piston-type gas-generator engine was found to exceed the performance of the other engines at all flight speeds investigated for flight ranges greater than 1600 miles at a turbine-inlet temperature of 1400° F and for flight ranges greater than 2000 miles at a turbine-inlet temperature of 1800° F. For the purpose of obtaining the maximum value of pay-load miles per hour, the most desirable cruising altitudes for the gas-generator engine in this analysis are between 35,000 and 45,000 feet for all flight velocities considered.

INTRODUCTION

The high efficiency that is expected from compounding the reciprocating engine is of particular interest for long-range flights. A specific compound engine called a piston-type gas-generator engine is proposed in reference 1. This engine diagrammatically shown in figure 1, utilizes a high degree of compounding in which the piston component does only the work of compression. Reference 2 includes a pay-load-range study of a reciprocating engine with varying degrees of compounding. This reference, however, compares the highly compounded reciprocating engine only with a conventional turbosupercharged reciprocating engine and the comparison includes only one altitude and one velocity.

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The piston-type gas-generator engine is further evaluated herein by comparing it with other engines when installed in a transport airplane. This evaluation was made at the NACA Lewis laboratory. The basis for comparison is the product of pay load per unit airplane gross weight and block speed or the pay-load miles per hour of operation per unit airplane gross weight.

The performance estimates of reference 1 are for a series of gas-generator engines, each designed for a particular altitude. Although considerable development might be necessary to produce a single engine design capable of approximating the performance indicated in reference 1, determination of the potentialities of this engine is desirable in order to determine if further research should be directed to this field.

The piston-type gas-generator engine of reference 1 is compared herein with other engines at flight ranges from 500 miles to ultimate range and velocities from 200 to 500 miles per hour. The engines chosen for comparison are a turbojet, a turbine-propeller, a compound, and a turbosupercharged reciprocating engine utilizing a variable-area exhaust jet nozzle. The comparison was made at turbine-inlet total temperatures of 1400° and 1800° F for the turbojet, turbine-propeller, and gas-generator engines. Increases in engine pressure ratio and air-handling capacity were assumed to accompany the increase in turbine-inlet temperature.

SYMBOLS

The following symbols are used in this analysis:

A effective aspect ratio

$C_{D,min}$ airplane minimum drag coefficient

L/D airplane lift-drag ratio

M resultant specific engine weight, installed engine plus propeller weight per pound of resultant thrust (Term "resultant" refers to net performance of complete power-plant unit including engine, propeller, nacelle, and heat exchangers.)

N resultant specific fuel consumption, pounds of fuel plus oil per resultant thrust horsepower-hour

Q instantaneous fuel rate, $\frac{(D/L)\ N}{375}$, pounds per mile per pound instantaneous gross weight

Q_0 initial fuel rate, pounds per mile per pound take-off gross weight

q_c compressible dynamic pressure, pounds per square foot

R range or length of trip, miles

r ratio of total fuel carried per trip to fuel burned per trip

S wing area, square feet

T_b block time per trip, $\frac{R}{V} + T_o$, hours

T_o net time lost per trip in climb, descent, and taxi, hours

t ratio of total fuel weight plus tank weight to total fuel weight

V cruise velocity, miles per hour

W_G airplane take-off gross weight, pounds.

w_e engine weight, pounds of installed engine plus propeller weight per pound take-off gross weight

w_f fuel plus oil burned per trip, pounds per pound take-off gross weight

w_p pay load carried per trip, pounds per pound take-off gross weight

w_s airplane empty weight less engines and fuel tanks, pounds per pound take-off gross weight

METHOD OF EVALUATION

A piston-type gas-generator engine was evaluated by comparing the performance of a transport airplane powered by this engine with the performance of an airplane having the same characteristics powered by other engines.

Pay-load ton-miles per hour of operation was used as the criterion because it is a measure of the rate at which transport work is accomplished. In order to make the analysis independent of airplane size, this criterion was divided by airplane take-off gross weight.

The conditions for the airplane and the engines are given in appendix A and are summarized in figures 2 to 4. Performance data for the piston-type gas-generator engine were obtained for the 1800° F turbine-inlet total-temperature conditions from reference 1 and were calculated for the 1400° F conditions according to the methods of the same reference. Reference 1 also includes a weight estimate for a particular gas-generator engine, which was not considered applicable to the present report for reasons given in appendix B. Other weight estimates were therefore required and are shown in appendix B along with a determination of specific frontal area for the gas-generator engine.

Transport Performance Equations

The transport performance expressed as pay-load ton-miles per hour per ton take-off gross weight is given by the following equation:

$$\frac{\text{ton-miles}}{(\text{hr})(\text{ton})} = \frac{w_p R}{\frac{R}{V} + T_o} \tag{1}$$

Pay-load weight per unit take-off gross weight w_p is the difference between take-off gross weight (unity) and the combined weight of airplane structure, engine, and fuel plus tanks.

$$w_p = 1 - \left(w_s + w_e + w_f r t\right) \tag{2}$$

Engine weight w_e can be determined from

$$w_e = \frac{D}{L} M$$

Practical limitations on wing loading require the airplane to fly at a lift-drag ratio less than maximum at high dynamic pressures but permit the airplane to fly at maximum lift-drag ratio at low dynamic pressures by a reduction in wing loading below the maximum allowable.

$$\frac{D}{L} = \frac{C_{D,min}\, q_c}{\left(\frac{W_G}{S}\right)_{max}} + \frac{\left(\frac{W_G}{S}\right)_{max}}{q_c \pi A} \quad \text{when} \quad q_c > \frac{\left(\frac{W_G}{S}\right)_{max}}{\sqrt{\pi A C_{D,min}}}$$

$$\frac{D}{L} = 2\sqrt{\frac{C_{D,min}}{\pi A}} \quad \text{when} \quad q_c < \frac{\left(\frac{W_G}{S}\right)_{max}}{\sqrt{\pi A C_{D,min}}}$$

where $\left(\frac{W_g}{S}\right)_{max}$ is maximum allowable wing loading.

The fuel weight w_f depends on the flight plan. Range and velocity are treated as independent variables. The lift-drag ratio and the velocity during cruise are maintained constant by climbing to higher altitude as gross weight decreases because of the consumption of fuel. The resultant specific fuel consumption is assumed constant throughout cruise and the fuel required to raise the airplane is neglected because most of this energy is available as potential energy. These assumptions allow the use of the Breguet range equation from which

$$w_f = 1 - e^{-QR}$$

Determination of Optimum Altitude

The performance of an engine-airplane combination may be greatly affected by the flight altitude as can be seen by the effect of altitude on airplane lift-drag ratio. For a typical airplane operating at 500 miles per hour, the lift-drag ratio varies from 4 at sea level to 16 at 50,000 feet. This variation in lift-drag ratio affects both the power required and the fuel consumed and therefore the airplane performance. Flight altitude also affects engine specific weight and specific fuel consumption, which in some cases compensates for the effect of lift-drag-ratio variation and in other cases adds to it, depending on the characteristics of the engine. The performance of the transport airplane is most sensitive to variations in flight altitude at high velocities and long ranges.

Unless altitude is restricted by prior consideration, the engine-airplane combination should be allowed to start the flight at the altitude that gives best performance. The initial cruising altitude was chosen in each case to give maximum pay-load ton-miles per hour of operation per ton gross weight. For any one range and velocity, this altitude will give maximum pay load. This altitude will hereinafter be designated optimum altitude.

The determination of the optimum altitude for a given range and velocity was simplified by assuming that the pounds of fuel burned per mile flown was constant and equal to its value at the start of cruising. This assumption was not made in the determination of transport performance but only in the determination of optimum altitude. The errors in ton-miles per hour per ton gross weight incurred by this assumption were checked by trial-and-error calculations on unpublished data and were found to be within the accuracy of the calculations. 1110

As previously mentioned, the optimum altitude for any one range and velocity is that altitude which yields maximum pay load. Differentiation of equation (2) with respect to altitude Z gives

$$-\frac{dw_p}{dZ} = \frac{dw_e}{dZ} + rt\,\frac{dw_f}{dZ}$$

and with the derivative set equal to zero

$$\frac{dw_e}{dw_f} = -\,rt \tag{3}$$

Application of the assumption that weight of fuel burned per mile flown is constant at initial value throughout the flight yields

$$w_f = Q_0 R$$

Then, for any given range

$$dw_f = R\,dQ_0 \tag{4}$$

and combination of equations (3) and (4) gives

$$\frac{dw_e}{dQ_0} = -\,Rrt$$

The approximate optimum altitude is therefore that altitude at which the slope of the curve of engine weight plotted against initial fuel rate is equal to $-Rrt$. This altitude cannot be found by differentiation because no simple equation exists for $w_e = f(Q_0)$. The approximate optimum altitude was graphically found from a plot of w_e against Q_0. This plot can be used for any range but only for one velocity.

RESULTS AND DISCUSSION

Optimum Altitude

The optimum altitudes for each engine are shown in figure 5 for any given flight range and several velocities. In all cases, the optimum altitude increased with an increase in range or velocity. As the range was increased, the fuel weight became a greater percentage of the disposable load and the specific fuel consumption therefore became more important. The over-all efficiency for any one velocity was increased by increasing the cruising altitude. This increase in efficiency was effected by an increase in lift-drag ratio, a decrease in resultant specific fuel consumption, or both, and at a sacrifice in engine weight.

Transport Performance

The variation of ton-miles per hour per ton take-off gross weight with range at several velocities is shown in figures 6 and 7 for the airplane powered by each of the engine types considered. The turbine-inlet temperatures for the turbojet, the turbine-propeller, and gas-turbine engines in figures 6 and 7 are 1400^{c} and 1800° F, respectively. The gas-generator engine has a marked advantage over the other engines at long range at all the velocities considered. The advantage is due to the relatively low specific fuel consumption of this type of engine. At short range, this advantage decreases because efficiency is of less importance. The high efficiency of the gas-generator engine is a result of operating at a high peak-cycle temperature and a correspondingly high peak-cycle pressure. These peak-cycle conditions are possible because they occur in the reciprocating component and not at the inlet to the turbine as in a turbine-propeller engine. The turbine can operate at the same limiting conditions in both engines, but in the gas-generator engine the work of compression has already been done, which leaves the turbine to accomplish only the useful work of the cycle. As allowable turbine-inlet temperatures increase, the advantage of the gas-generator engine over the turbine-propeller engine can be expected to decrease. This expectation is confirmed by a comparison of figures 6 and 7. The relative merit of the piston-type gas-generator engine based on this comparison increased as the flight range increased. For a 500-mile-per-hour condition, the calculated ultimate range of the piston-type gas-generator engine exceeded the ultimate ranges of the other engine types by the following percentages: spark-ignition compound engine, 25 percent; turbine-propeller engine, 55 percent; turbojet engine, 61 percent; and conventional reciprocating engine, 94 percent.

Inasmuch as the estimated performance of the compound engine and the reciprocating engine were taken from two independent sources, the engine conditions are probably not comparable. These engines should not be compared with each other in order to evaluate the two methods of utilizing exhaust gas from a reciprocating engine.

A better comparison among the various engines at short ranges is shown in figure 8. The speed-range spectrum in this figure is divided into areas showing which engine has the best performance for any given range and velocity. The reciprocating and the compound engines do not appear in this figure because their performance was always exceeded by some other engine. The dotted contours of figure 8 represent the payload ton-miles per hour of operation per ton take-off gross weight for each engine in the area in which it dominates. At any one range, velocities lower than the velocity yielding the maximum transport performance will be of little importance. Any point that lies in an area where the dotted contours have a positive slope is therefore relatively unimportant.

The performance of the piston-type gas-generator engine was found to exceed the performance of the other engines at all flight speeds investigated for flight ranges greater than 1600 miles at a turbine-inlet temperature of 1400° F and for flight ranges greater than 2000 miles at a turbine-inlet temperature of 1800° F.

The turbojet engine compared more favorably with the turbine-propeller engine when operating at the low turbine-inlet temperature (figs. 6 and 8(a)) than it does when operating at the high turbine-inlet temperature (figs. 7 and 8(b)). This difference is to be expected because an increase in turbine-inlet temperature increased the cycle efficiency of both engines but decreased the propulsive efficiency of the turbojet engine, whereas the propulsive efficiency of the turbine-propeller engine remained the same.

Consideration of only those velocities above that yielding optimum performance at each range can lead to a significant conclusion concerning the necessary range of cruising altitudes over which the gas-generator engine must operate. Velocities under 400 miles per hour for short range are undesirable because the dotted contours have positive slopes (fig. 8). Application of this reasoning to figures 8 and 5(c), indicates that for this analysis the most important cruise altitudes for the gas-generator engine are between 35,000 and 45,000 feet for all ranges. As previously mentioned, the data used for the gas-generator engine were for an idealized engine; consequently, operation at an altitude other than design would incur some sacrifice in performance. This sacrifice in performance would be of small consequence, however, if the range of desirable cruising altitudes is as restricted as indicated by the foregoing considerations.

SUMMARY OF RESULTS

A comparison of a piston-type gas-generator engine with other engine types for use in subsonic transport operations led to the following results:

1. The relative merit of the piston-type gas-generator engine based on this comparison increased as the flight range increased. For a 500-mile-per-hour condition, the calculated ultimate range of the piston-type gas-generator engine exceeded the ultimate ranges of the other engine types by the following percentages: spark-ignition compound engine, 25 percent; turbine-propeller engine, 55 percent; turbojet engine, 61 percent; and conventional reciprocating engine, 94 percent.

2. The performance of the piston-type gas-generator engine was found to exceed the performance of the other engines at all flight speeds investigated for flight ranges greater than 1600 miles at a turbine-inlet temperature of 1400° F and for flight ranges greater than 2000 miles at a turbine-inlet temperature of 1800° F.

3. For the purpose of obtaining the maximum value of pay-load miles per hour, the most desirable cruising altitudes for the gas-generator engine in this analysis are between 35,000 and 45,000 feet for all flight velocities considered.

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APPENDIX A

DETAILED CONDITIONS

The detailed conditions used in the comparison of engines are presented.

Airplane.- The airplane characteristics are:

Minimum drag coefficient of airplane less nacelles	0.019
Effective aspect ratio	8.0
Nacelle drag coefficient based on maximum engine frontal area:	
Jet engines	0.04
Propeller engines	0.055
Ratio of airframe weight (without engines) to gross weight	0.45
Maximum allowable wing loading, lb/sq ft	80
Fuel-tank weight, percent of fuel weight	10

The wing loading was reduced below 80 pounds per square foot to provide maximum lift-drag ratio at dynamic pressures below 116 pounds per square foot. Changes in the ratio of total engine power to gross weight can be accomplished by variations in the number of engines and in the engine size.

Propellers. - The variation of propeller specific weight with velocity and altitude is shown in figure 2. Propulsive efficiency varies with flight Mach number, as presented in figure 3. These propeller characteristics were obtained from reference 3.

Engines. - The performance characteristics of the engines are given in figure 4. Data for the four-row air-cooled reciprocating engine of 4360-cubic-inch displacement equipped with a two-stage turbosupercharger were obtained from reference 4. Exhaust from the turbine passes through a variable-area jet nozzle to obtain additional thrust. The necessary weight additions for installation were assumed to result in a total installed weight of 5000 pounds for the reciprocating engine without propeller.

Performance of the compound engine, which is the same as that of reference 3, is based on calculations for a four-row air-cooled engine of 4360-cubic-inch displacement geared to a turbine and an auxiliary compressor. The performance characteristics of the compound engine are based on the following cruising conditions:

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Mean piston speed, ft/min.	2200
Fuel-air ratio	0.063
Inlet-manifold pressure, in. Hg absolute	40
Exhaust-manifold pressure, in. Hg absolute	40
Brake specific oil consumption, lb/hp-hr	0.01
Compressor adiabatic efficiency, percent	80
Turbine adiabatic efficiency (total to static), percent	80
Intercooler effectiveness, percent	50
Gear efficiency (between turbine and engine), percent	95
Installed cruising specific weight at sea-level static conditions, lb/bhp	2.18
Ratio of installed weight to frontal area at sea-level static conditions, lb/sq ft.	280

The installed weight in this tabulation does not include that of a propeller. In estimating the specific weight of the compound engine at each flight condition, it was assumed that the turbine, the auxiliary compressor, and the intercooler are redesigned to suit each altitude. The cruising turbine-inlet total temperature is approximately 1700° F for the compound engine.

Calculated data for the other engines were based on the assumptions given in table I. The installed weight in table I is without propeller. In calculating the performance of the turbojet and turbine-propeller engines at each altitude and velocity, the engines were assumed to be redesigned without change in weight per unit of corrected air flow in order to maintain the assumed cruising conditions. Air flow at each flight condition for the turbojet and turbine-propeller engines was calculated assuming a constant Mach number at the compressor inlet. The ratio of propeller thrust to jet thrust for the turbine-propeller engine was chosen at each flight condition to yield the maximum thrust per pound of air flow per second.

Other conditions. - The airplane was assumed to be flown in still air, assuming NACA standard atmospheric conditions. Only cruising conditions were considered. Engines and propellers were installed to fulfill cruising requirements and satisfactory take-off and climb performance were assumed attainable by the increase in engine output that usually accompanies a reduction in altitude and by an increase in engine output for short-period operation. The net time lost per trip T_o is 0.20 hour and the required fuel reserve was 5 percent of cruising fuel (r=1.05).

APPENDIX B

DETERMINATION OF SPECIFIC WEIGHT AND

FRONTAL AREA OF GAS-GENERATOR ENGINES

Specific Weight

A weight estimate, which leads to a specific-weight figure for a particular gas-generator engine, is included in reference 1. This weight estimate, however, was not considered applicable to the purposes of this report because the engine was limited to an altitude of 20,000 feet and because no data were given in reference 1 for the gas-generator engine operating at the low turbine-inlet temperature (peak cylinder pressure of 1200 lb/sq in. and a turbine-inlet total temperature of 1400° F). Furthermore, recent information indicates that some of the component-weight values used in reference 1 were too high. Accordingly, the analysis is repeated here for the high turbine-inlet temperature (peak cylinder pressure of 1600 lb/sq in. and turbine-inlet total temperature of 1800° F) with corrected component weights and with operation extended to an altitude of 50,000 feet. All values were based on an engine of 3000 brake horsepower at the upper altitude limit and the component weights were evaluated for either the sea-level or altitude conditions, whichever was greatest.

Power section. - The weight of the gas-generator power section was based on figures obtained from the Junkers 207 compression-ignition aircraft engine, which indicated that the weight of the power section of this engine was approximately 800 pounds. This weight was then increased to include such accessories as the starter and generator.

The limiting condition for the power-section weight is obviously the altitude conditions. A calculation of the performance of the gas-generator engines shows that at an altitude of 50,000 feet the brake output of the low-temperature engine is 0.0191 Btu per cycle per cubic inch of clyinder volume and that the high-temperature engine is 0.0350 Btu per cycle per cubic inch of cylinder volume. The required cylinder volumes for the two engines to deliver the specified power are therefore 2775 and 1515 cubic inches, respectively, for engines operating at 2400 cycles per minute. If the clearance volume and the fact that only about 80 percent of the stroke will be usable because of the port height are taken into consideration, the required displacements for the two engines become 2970 and 1610 cubic inches. Correcting the weight of the power section of the Junkers engine by the ratio of the required displacement to that of the Junkers engine (1115 cu in.) and adding the extra 200 pounds results in power-section weights of 2330 and 1355 pounds for the two engines.

Compressor. - The weight of the compressor was determined from weight data from a current turbojet engine. This engine had a compressor weighing 466 pounds, which was capable of handling a sea-level air flow of 50.5 pounds per second and used 11 stages to produce a pressure ratio of 4. The weight of this compressor was corrected for gas-generator application by the ratio of the air densities at sea level and 50,000 feet, the ratio of the required and actual weight-flow capacity of the compressor, and the ratio of the logarithms of the required and actual compressor pressure ratios. The required weight flows and pressure ratios were determined from the gas-generator performance data, which indicated brake specific air consumptions of 10.4 pounds per horsepower-hour for the low-temperature engine, 8.0 pounds per horsepower-hour for the high-temperature engine, and manifold pressures of 60 to 90 pounds per square inch absolute for the two engines, respectively. This calculation resulted in compressor weights of 1415 and 1245 pounds for the two engines.

Turbine. - The turbine weight was determined using the relation of stage weight against volume-flow capacity, shown in figure 9 of reference 1. The volume flow shown as the abscissa of this curve was calculated for the gas-generator engines by taking the arithmetical average of the turbine-inlet and turbine-outlet volume flows, which in turn were calculated from a knowledge of the turbine-inlet pressure and by use of the perfect gas law. Turbine-inlet pressures of 54 and 82 pounds per square inch absolute were used for the two engines. In determining the turbine-outlet temperature, expansion was assumed to proceed to ambient pressure with an adiabatic exponent of 1.35. The resultant stage weights of the two engines as determined by this calculation were 309 and 220 pounds.

A five-stage turbine would be required to operate through the gas-generator pressure ratio at an altitude of 50,000 feet. The previous stage weight, however, includes the turbine shaft and all the necessary housing, so that stages subsequent to the first stage were assumed to weight 0.7 as much as the first stage. The final turbine weights were therefore found to be 1174 and 835 pounds for the two engines.

Reduction gear. - The reduction-gear weights were calculated using a figure of 0.143 pound per transmitted horsepower, which was obtained from weight data on a current turbine-propeller engine. This gear weight was based on the brake horsepower of the gas-generator engines at sea-level altitude. This computation resulted in weights of 689 and 637 pounds for the two engines.

Compressor-drive gear. - The weight of the compressor-drive gear was calculated using a specific weight of 0.7 of that of the propeller

reduction gear because a smaller speed ratio was involved. Compressor-power requirements were calculated using the weight flows and pressure ratios at sea-level altitude, as the compressor power required was higher there than at 50,000 feet. In calculating the compressor power, an adiabatic exponent of 1.395, an efficiency of 0.85, and a specific heat for air of 0.24 Btu per pound per °R were used. The transmitted power was found to be 3520 and 2440 horsepower for the two engines, which resulted in weights of 352 and 245 pounds for the drive gear of the two engines. This weight was subsequently increased by 30 percent to values of 458 and 318 pounds to take care of the condition at which it becomes necessary to divide the compressor into two parts that operate at different speeds. For this condition, two gear boxes would be required. 1110

Heat exchangers. - Heat-exchanger weights were determined using data from reference 5, which indicated heat-rejection rates of 2500 Btu per minute per square foot per 100° F initial-temperature difference for oil coolers and 6000 Btu per minute per square foot per 100° F initial-temperature difference for radiators. Wet weights of 48 and 54 pounds per square foot of frontal area were used for these two coolers.

In determining the heat rejection of the gas-generator engines, 16 percent of the heat input of the fuel was assumed to be rejected to the coolant and 2 percent was assumed to be rejected to the oil. The heat input of the fuel was determined from the power output of the engine and the brake specific fuel consumption by use of a heating value for the fuel of 18,500 Btu per pound. Analysis showed that the sea-level condition was the limiting condition, where the heat-rejection rates for the low-temperature engine were 89,500 Btu per minute to the coolant and 11,200 Btu per minute to the oil. The corresponding figures for the high-temperature engine were 78,300 and 9,780 Btu per minute.

The radiator weights resulting from this calculation were 422 and 370 pounds and the oil-cooler weights were 126 and 110 pounds for low- and high-temperature engines, respectively.

Installation weight. - Necessary installation weight, including engine mounting, cowling, air scoop, oil, coolant, and fuel systems and reinforcement of the manifolds to withstand high pressures was assumed for both engines. The assumed figures were 840 and 620 pounds, which are in direct proportion to the total weights of the engines.

Total installed weight. - The total installed weights of the gas-generator engine are given by the sum of the components listed in the following table:

Component ↓ \ Turbine-inlet temperature °F →	Weight, (lb)	
	1400	1800
Power section	2330	1355
Compressor	1415	1245
Turbine	1174	835
Reduction gear	689	637
Compressor-drive gear	458	318
Heat exchangers	548	480
Miscellaneous for installation	841	620
Total installed weight	7455	5490

Specific weight. - The specific weight calculated for the 50,000-foot altitude or for 3000 horsepower was 2.48 for the low-temperature engine and 1.83 for the high-temperature engine.

Frontal Area

The frontal areas of the gas-generator engines were determined from a consideration of the outside dimensions of the Junkers engine together with an allowance for the compressor, the turbine, and the reduction gear.

The frontal area of the high-temperature gas-generator engine was assumed to consist of a rectangle 21.5 inches high by 52 inches wide. These are the frontal dimensions of the Junkers engine. Whereas the displacement of the Junkers engine is somewhat less than that required by the low-temperature gas-generator engine, it was assumed that the displacement could be increased by increasing the engine bore and, if necessary, adding in-line cylinders to the engine. Because the bore-stroke ratio of this engine is rather low and because the outside dimensions are determined by the crankcase dimensions, the bore may be increased without increasing the outside dimensions. Superimposed on this rectangle was a circular area of 30 inches in diameter, which was assumed to cover the maximum requirements of the compressor, the turbine, and the reduction gear. The resultant area of this configuration was 8.6 square feet.

In the case of the low-temperature engine, one Junkers engine is not sufficient to provide the necessary displacement. Two such engines

were therefore assumed adjacent to each other. The resultant frontal area configuration is therefore a rectangle 43 inches high by 52 inches wide. Because this area is larger than the diameter of the compressor, the turbine, or the reduction gear, these elements need not be considered. The resultant area of this configuration is 15.5 square feet.

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TABLE I. - ENGINE CONDITIONS

	Turbine-inlet total temperature					
	1400° F			1800° F		
	Gas generator	Turbine propeller	Turbojet	Gas generator	Turbine propeller	Turbojet
Compressor adiabatic efficiency, percent	85	85	85	85	85	85
Turbine adiabatic efficiency (total pressure), percent	----	90	90	----	90	90
Turbine adiabatic efficiency (total to static pressures), percent	85	----	----	85	----	----
Jet-nozzle adiabatic efficiency, percent	----	95	95	----	95	95
Combustion efficiency, percent	----	95	95	----	95	95
Reduction-gear efficiency, percent	95	95	----	95	95	----
Ram-pressure recovery of dynamic pressure, percent	0	90	90	0	90	90
Compressor pressure ratio	----	8	4	----	10	10
Combustion pressure drop, percent of combustor-inlet total pressure	----	3	3	----	3	3
Peak cylinder pressure, (lb/sq in. abs.)	1200	----	----	1600	----	----
Mean piston speed, (ft/min)	2530	----	----	2530	----	----
Heating value of fuel, (Btu/lb)	18,500	18,500	18,500	18,500	18,500	18,500
Brake specific oil consumption, (lb/bhp-hr)	0.005	0	0	0.005	0	0
Ratio of installed weight to air flow at sea-level static conditions (lb-sec/lb)	360	60	25	390	65	32
Ratio of installed weight to frontal area, (lb/sq ft)	480	190	300	640	325	640

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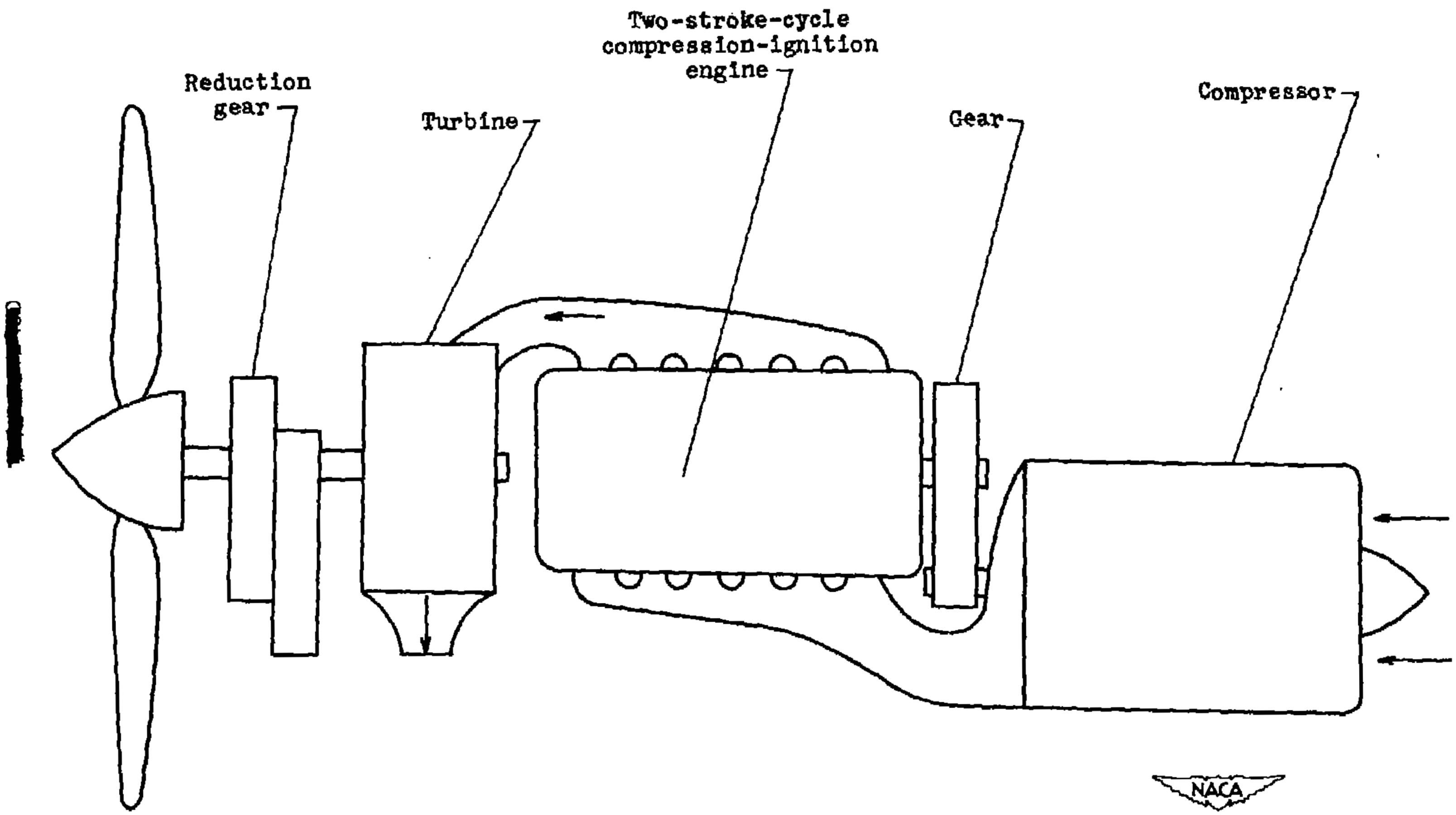

Figure 1. - Diagrammatic sketch of gas-generator engine used in analysis (reference 1).

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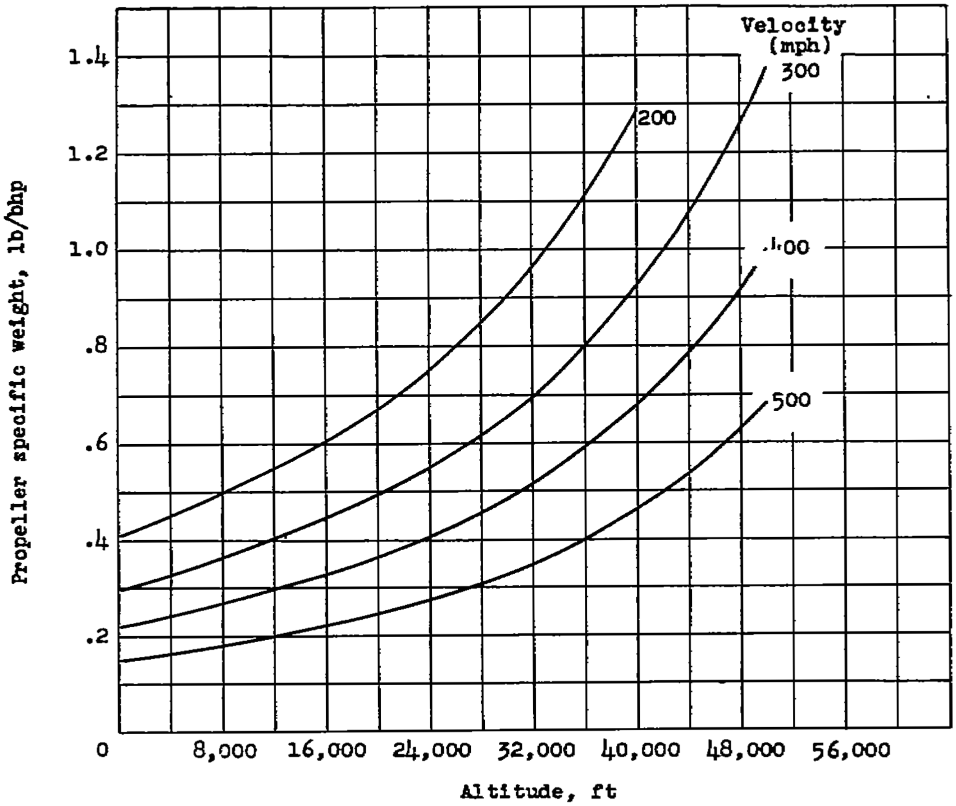


Figure 2. - Variation of propeller specific weight with altitude at several velocities.

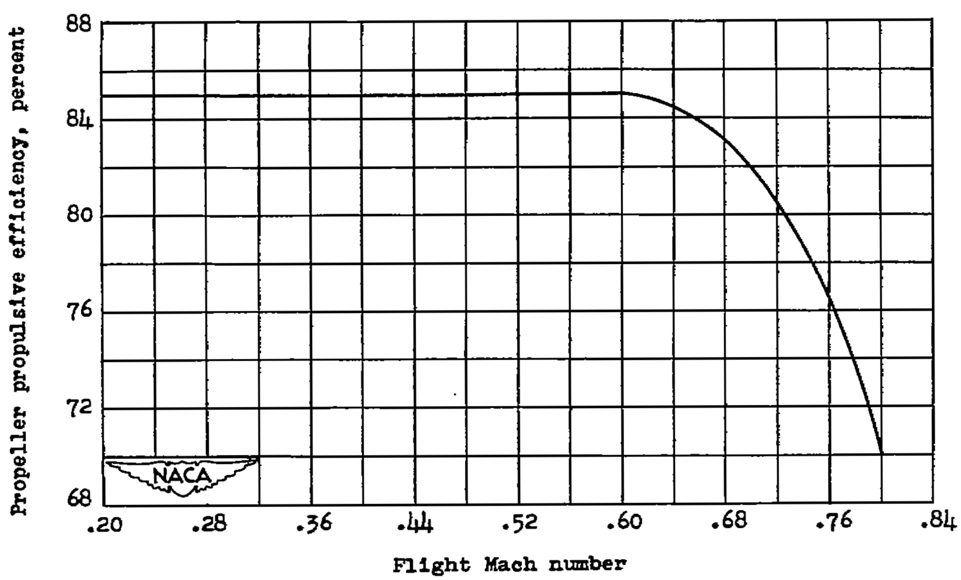
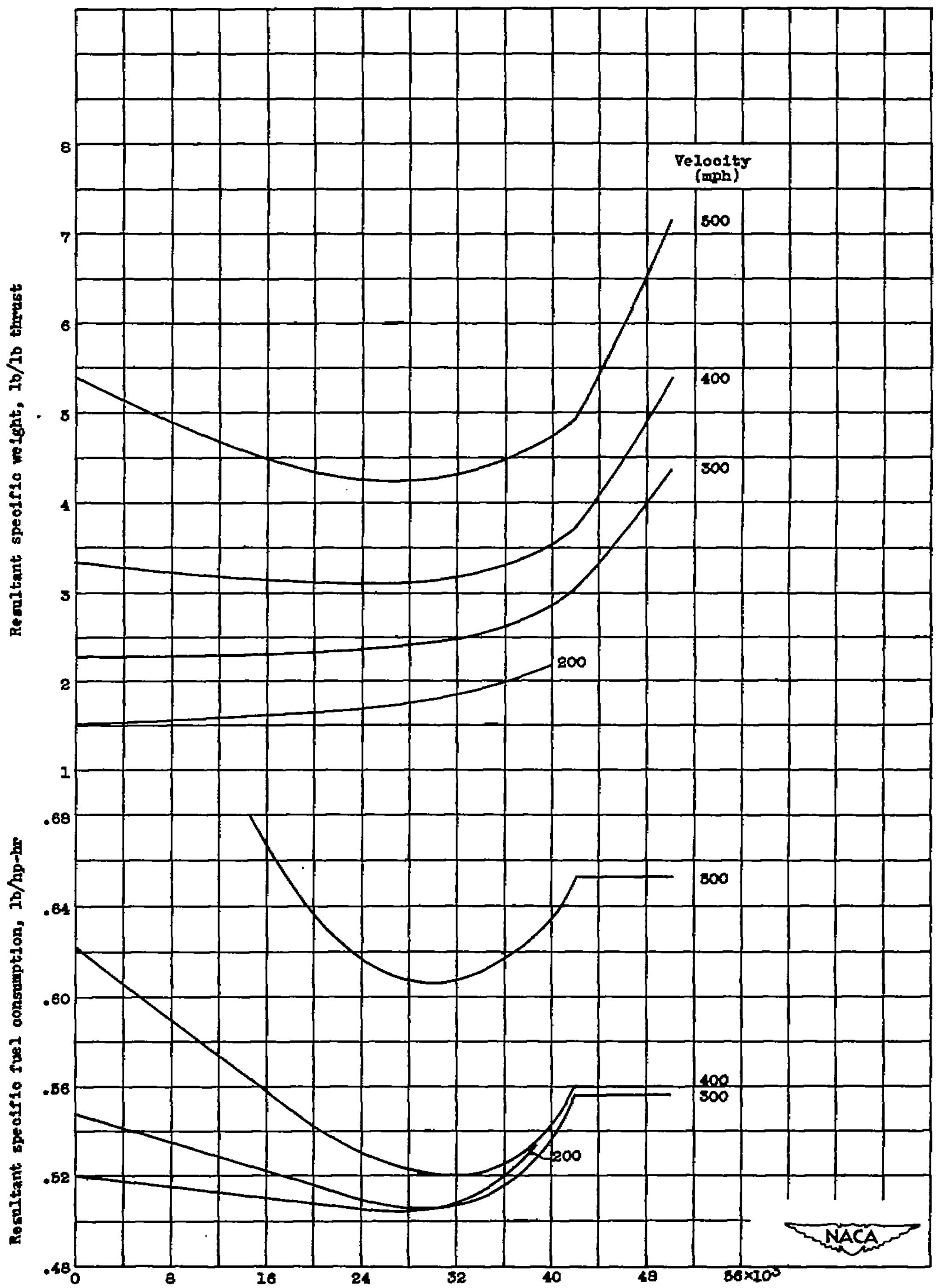


Figure 3. - Variation of propeller propulsive efficiency with flight Mach number.

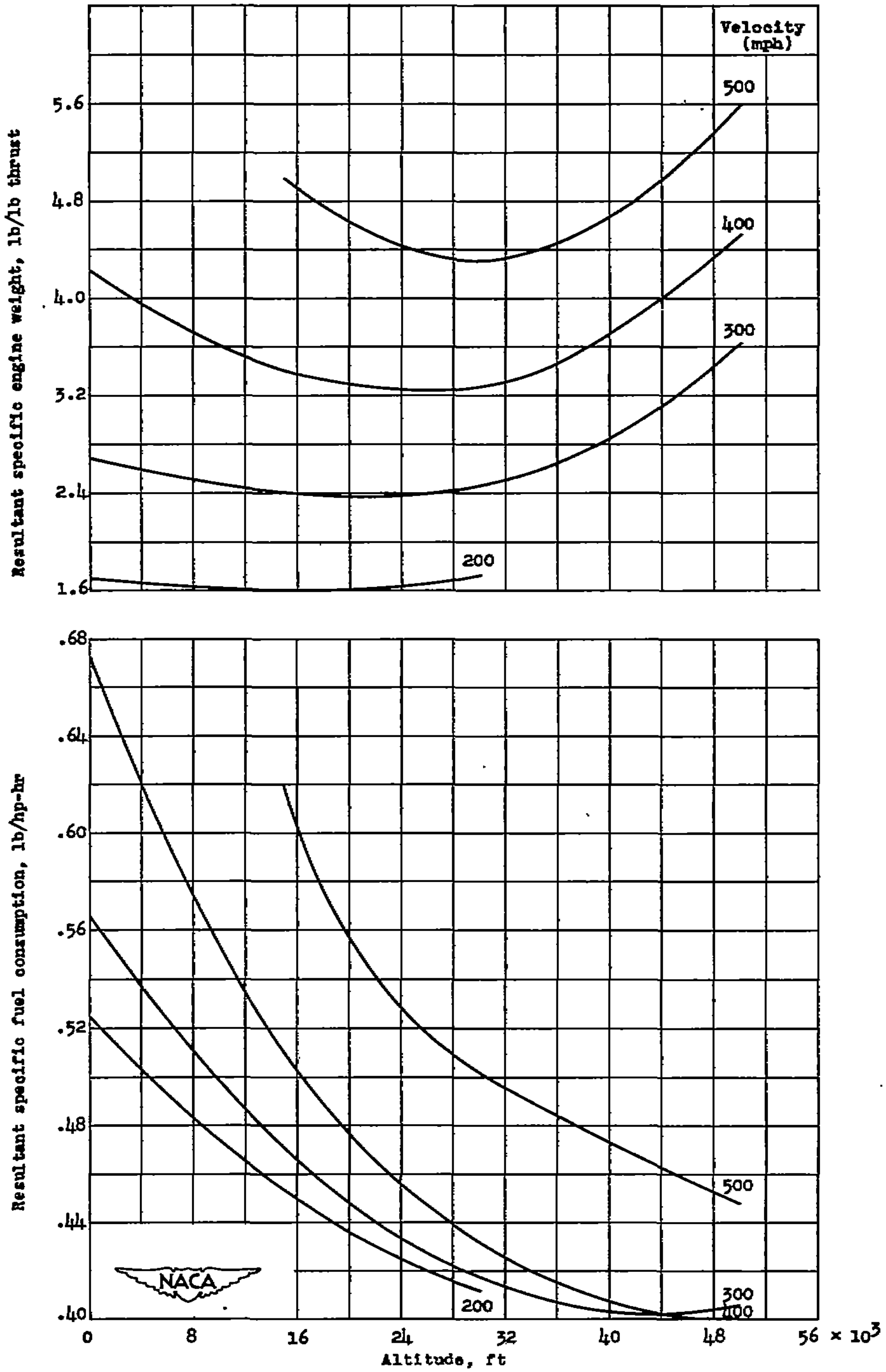


(a) Air-cooled reciprocating engine.

Figure 4. - Variation of cruising resultant specific engine weight and resultant specific fuel consumption with altitude at several velocities. Term "resultant" refers to the net performance of the complete power-plant unit including engine, propeller, nacelle, and heat exchangers.

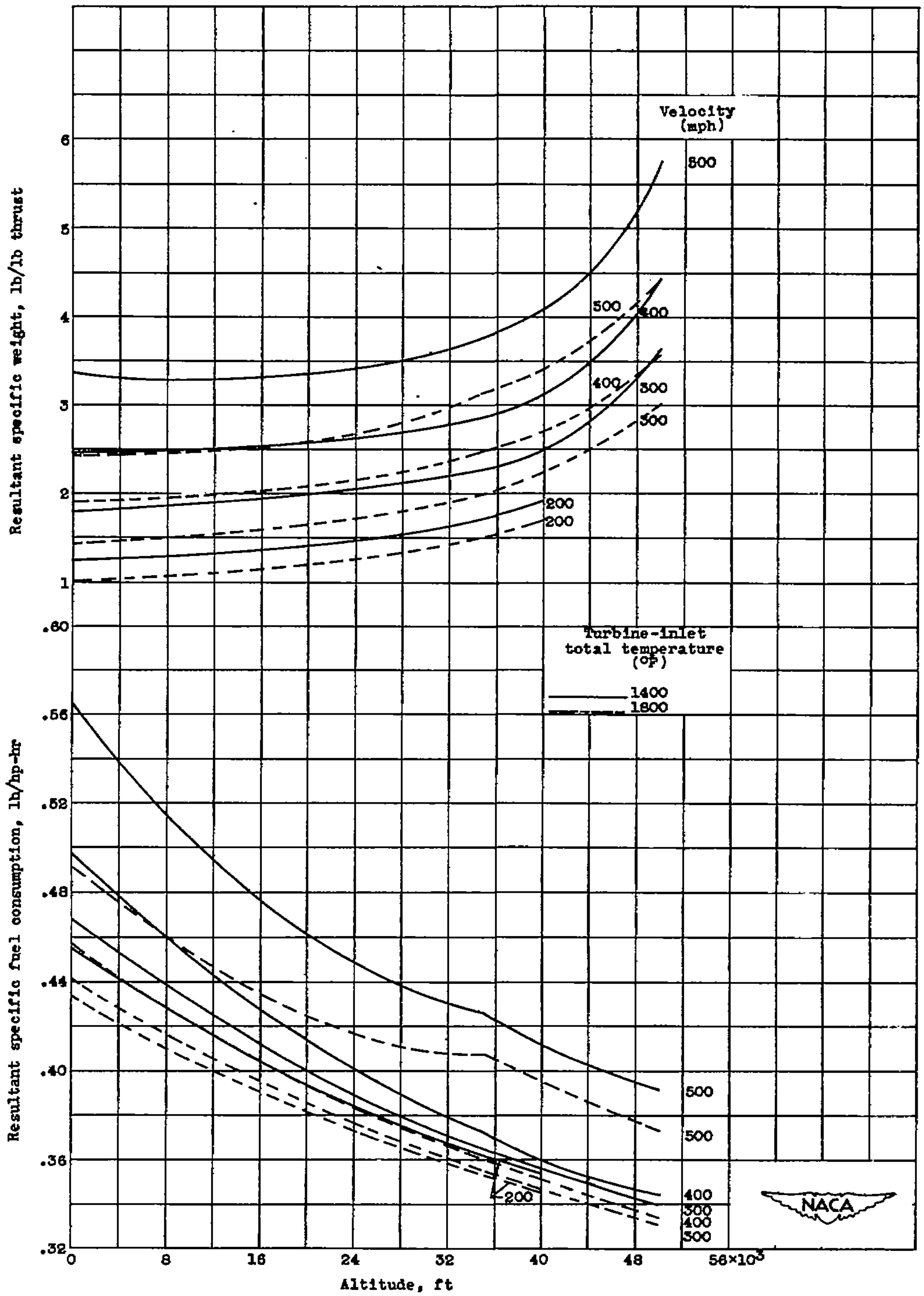
1110

1110



(b) Compound engine.

Figure 4. - Continued. Variation of cruising resultant specific engine weight and resultant specific fuel consumption with altitude at several velocities. Term "resultant" refers to net performance of complete power-plant unit including engine, propeller, nacelle, and heat exchangers.



(c) Gas-generator engine.

Figure 4. - Continued. Variation of cruising resultant specific engine weight and resultant specific fuel consumption with altitude at several velocities. Term "resultant" refers to the net performance of the complete power-plant unit including engine, propeller, nacelle, and heat exchangers.

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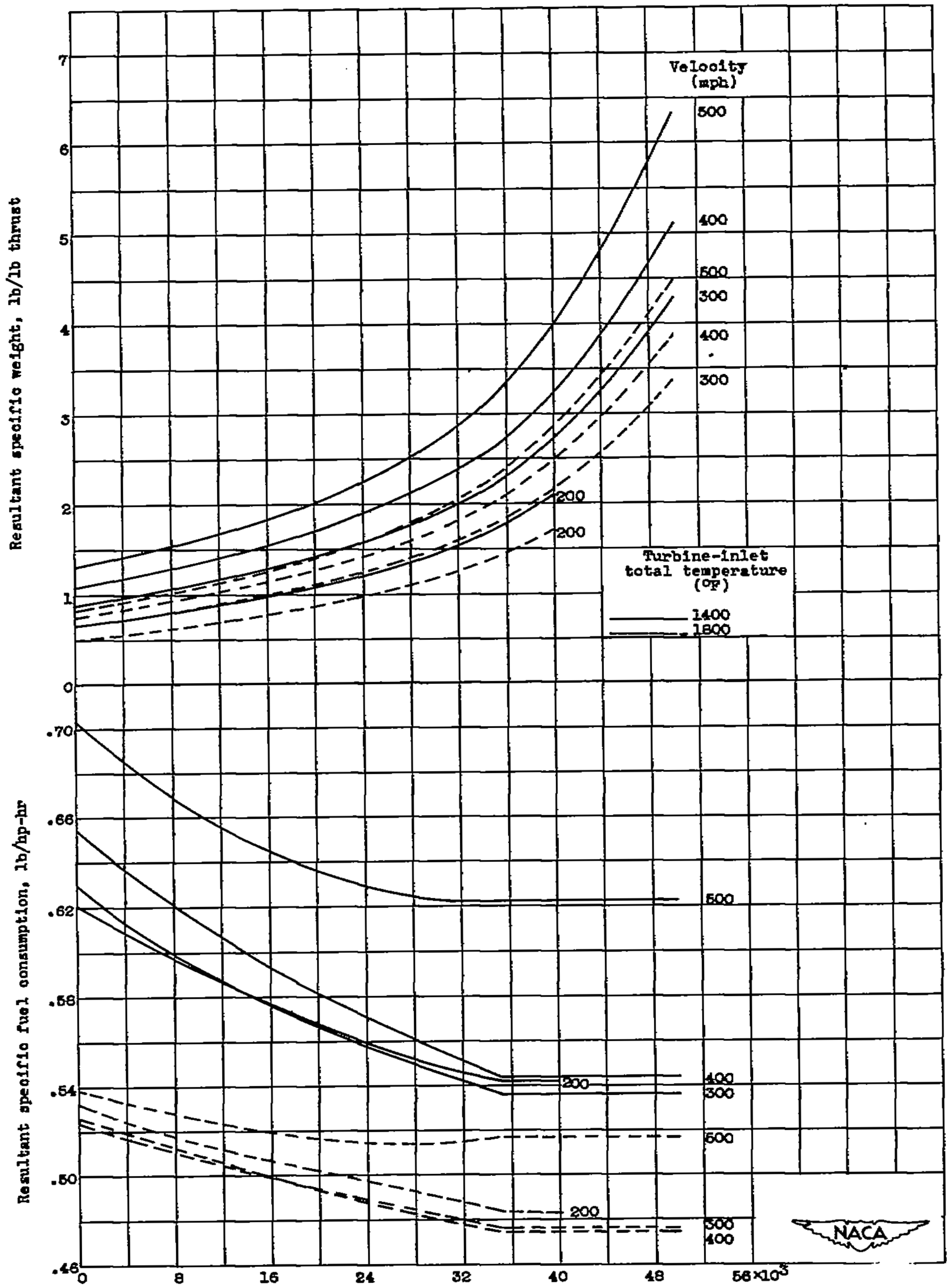
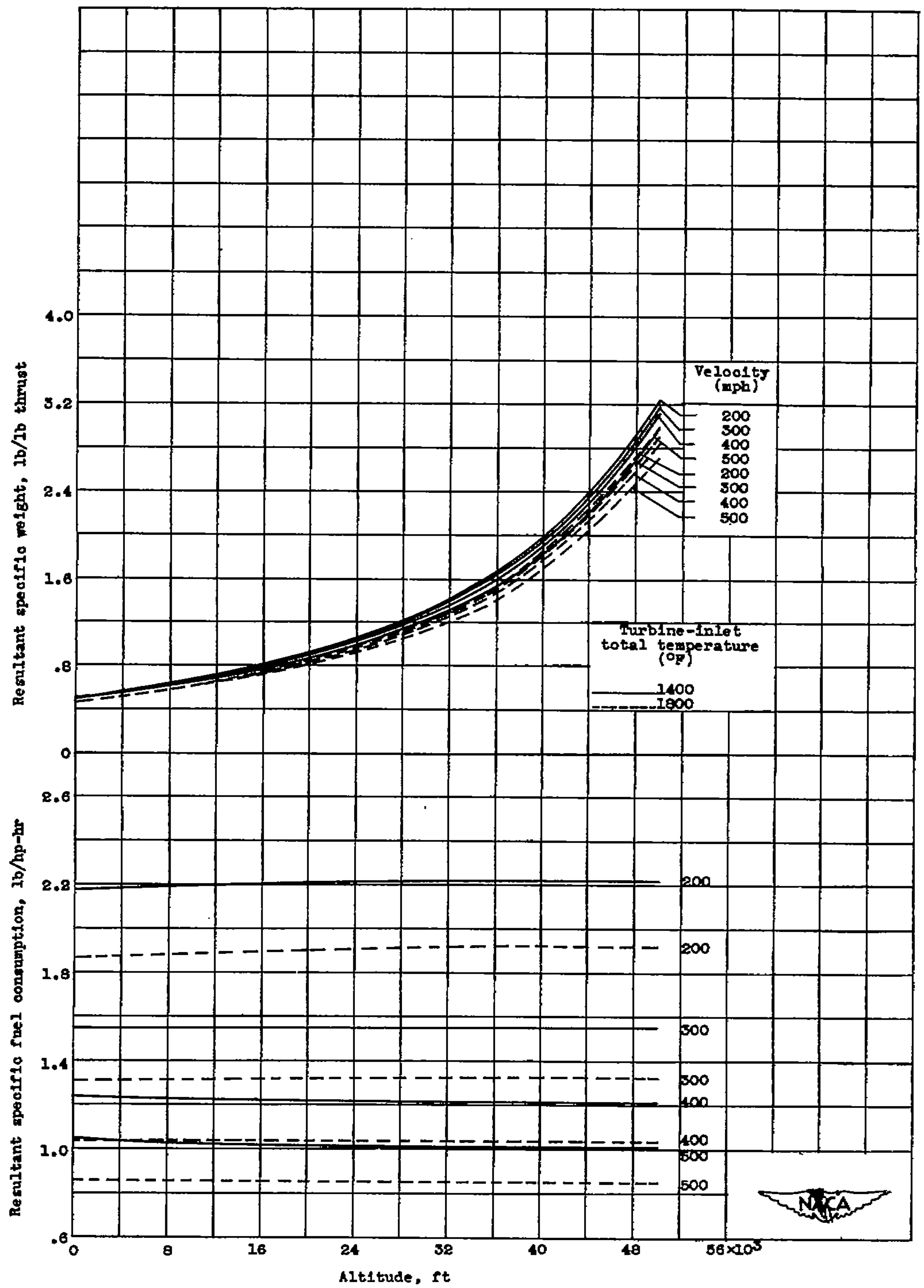


(d) Turbine propeller engine.

Figure 4. - Continued. Variation of cruising resultant specific engine weight and resultant specific fuel consumption with altitude at several velocities. Term "resultant" refers to the net performance of the complete power-plant unit including engine, propeller, nacelle, and heat exchangers.



(e) Turbojet engine.

Figure 4. - Concluded. Variation of cruising resultant specific engine weight and resultant specific fuel consumption with altitude at several velocities. Term "resultant" refers to the net performance of the complete power-plant unit including engine, propeller, nacelle, and heat exchangers.

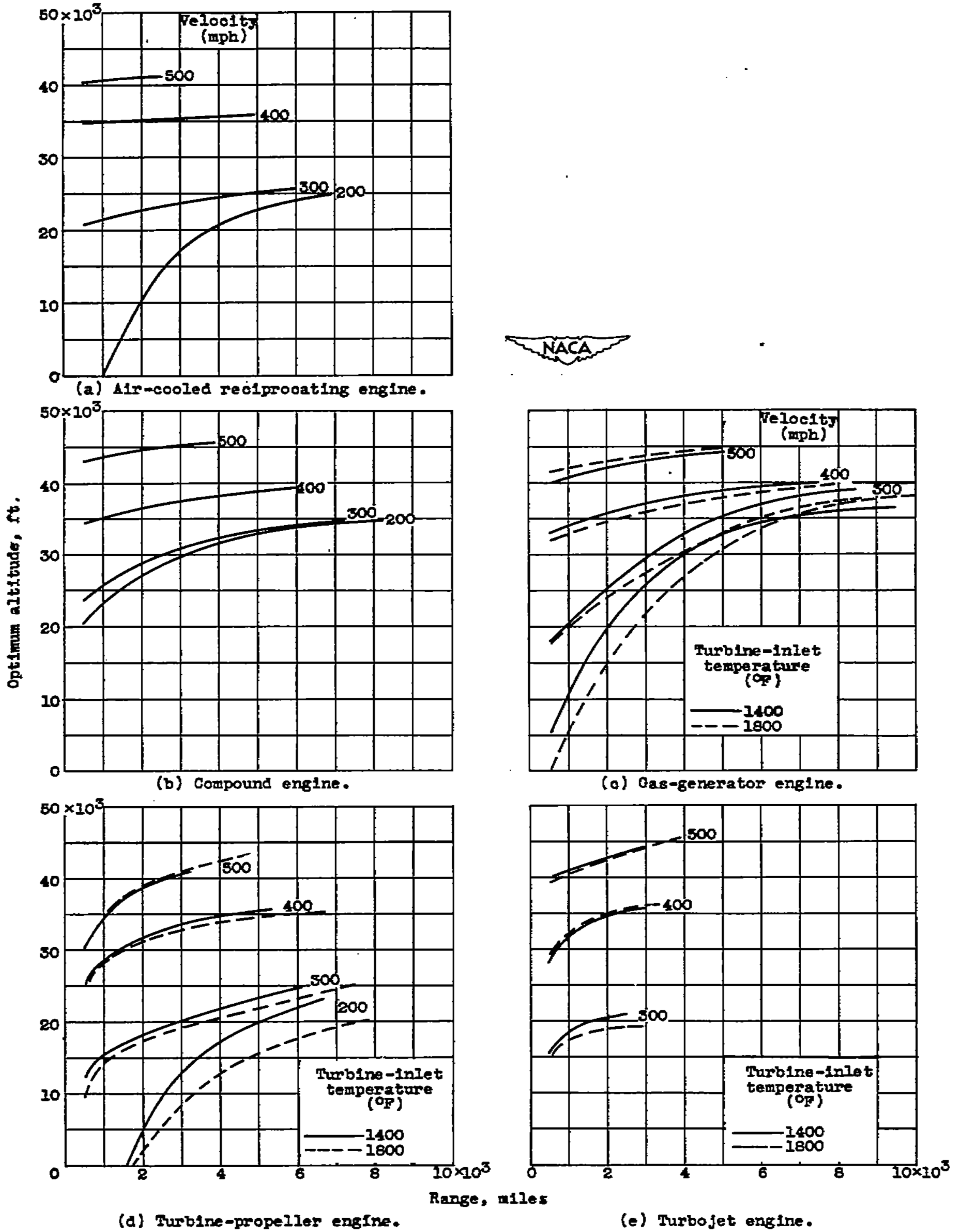
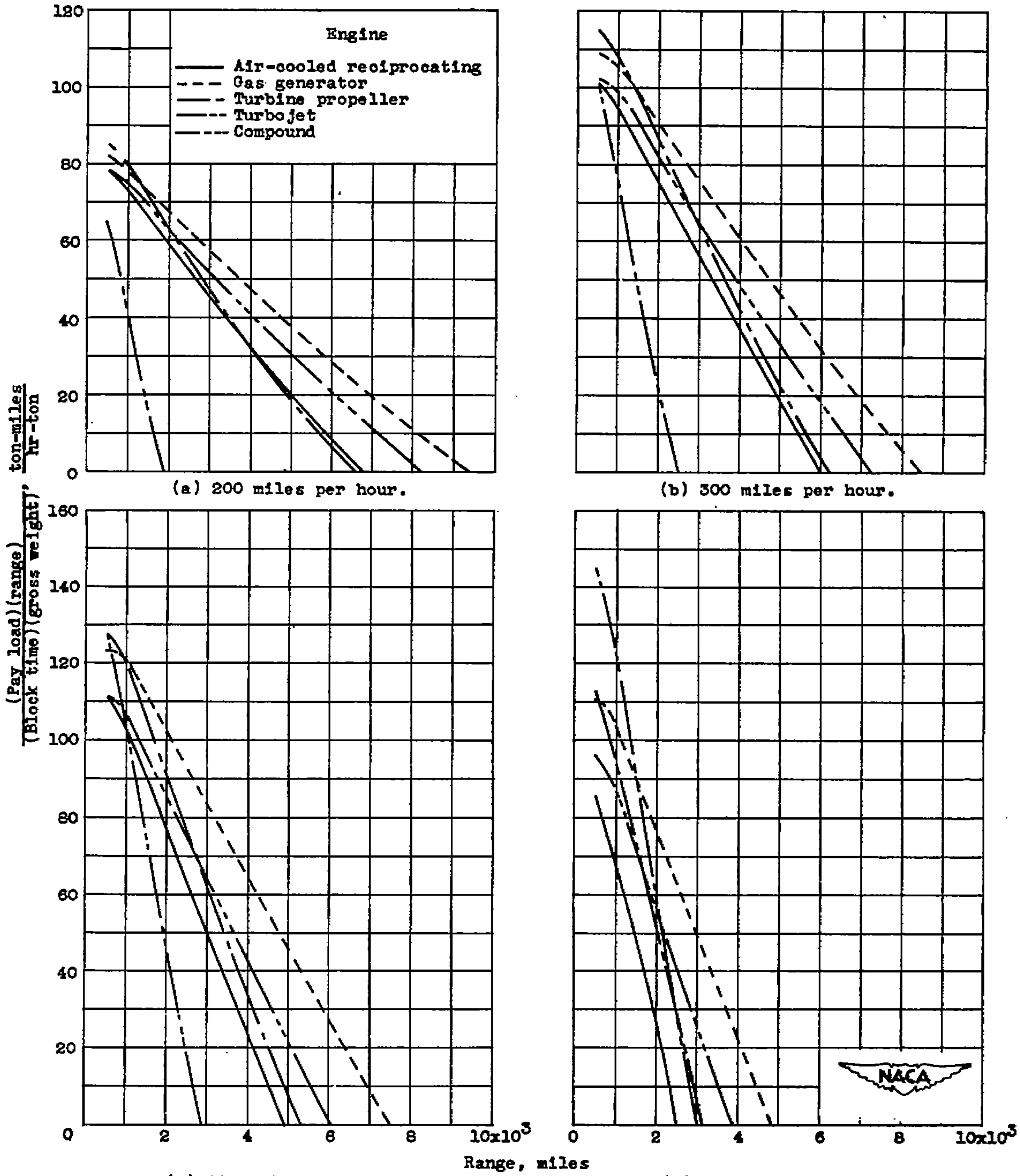


Figure 5. - Variation of optimum altitude with range for several engines at four different velocities.



(a) 200 miles per hour.

(b) 300 miles per hour.

(c) 400 miles per hour.

(d) 500 miles per hour.

Figure 6. - Variation of ton-miles per hour per ton gross weight with range for several engines at four different velocities. Turbine-inlet total temperature for turbojet, turbine-propeller, and gas-generator engines, 1400° F.

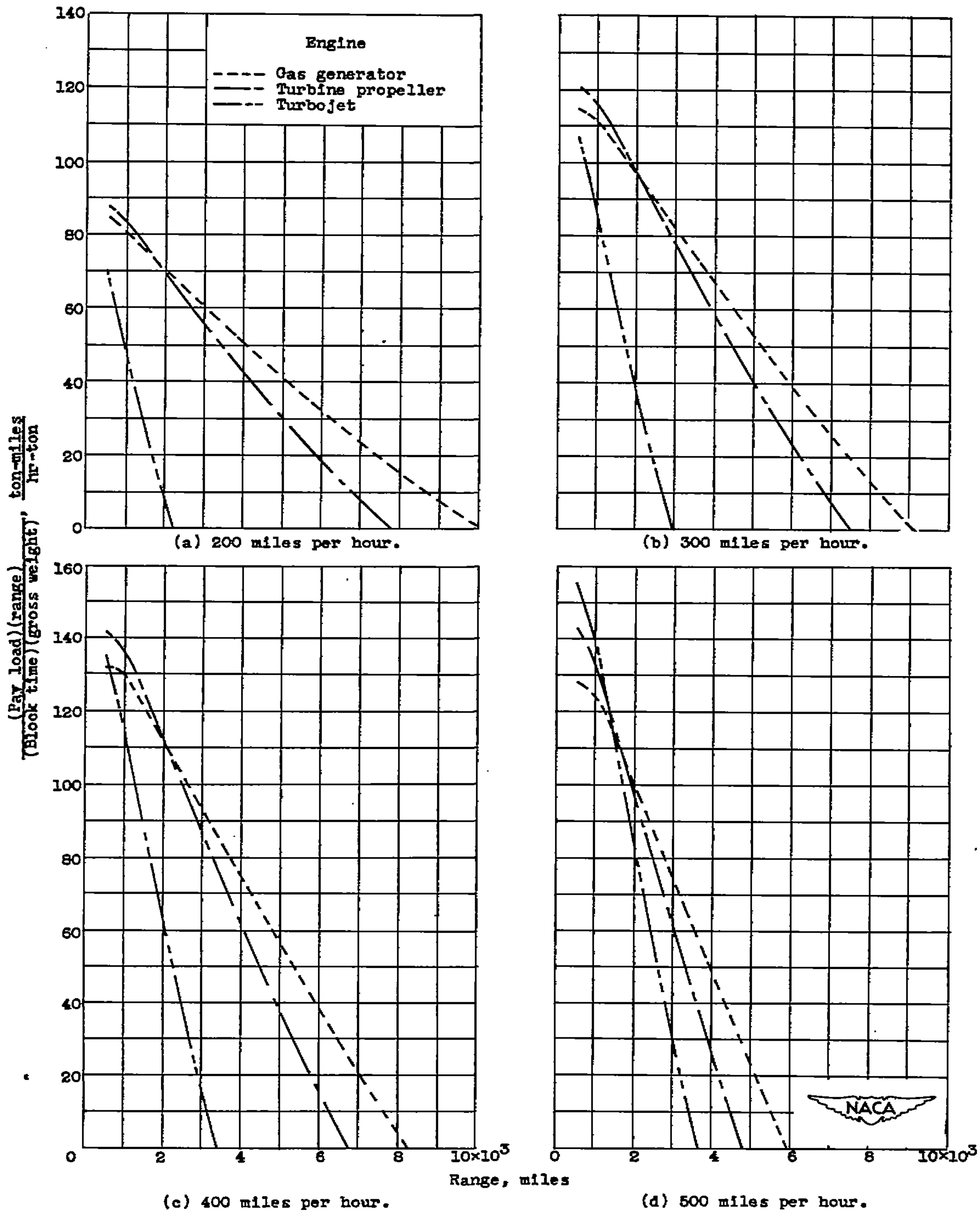
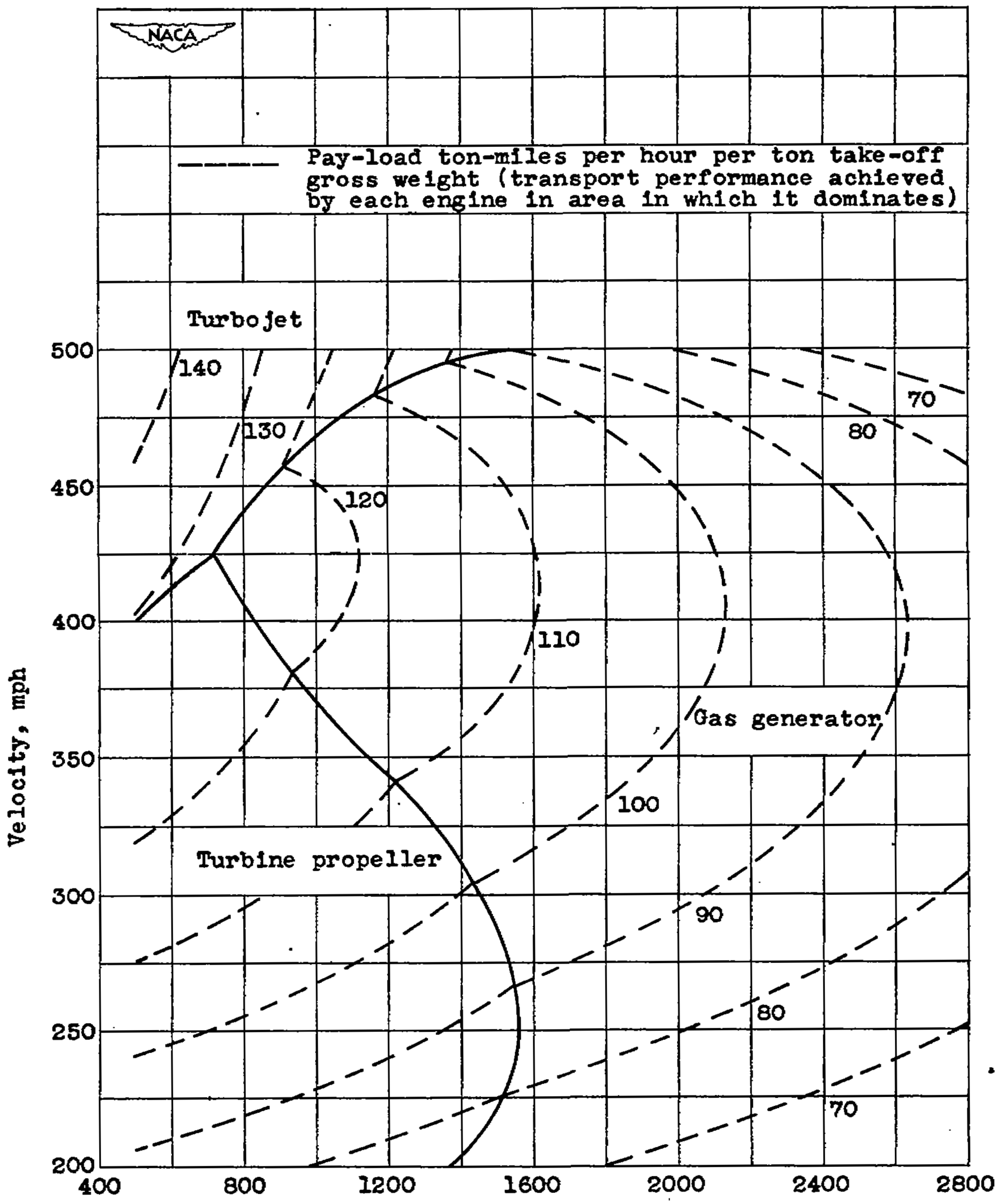


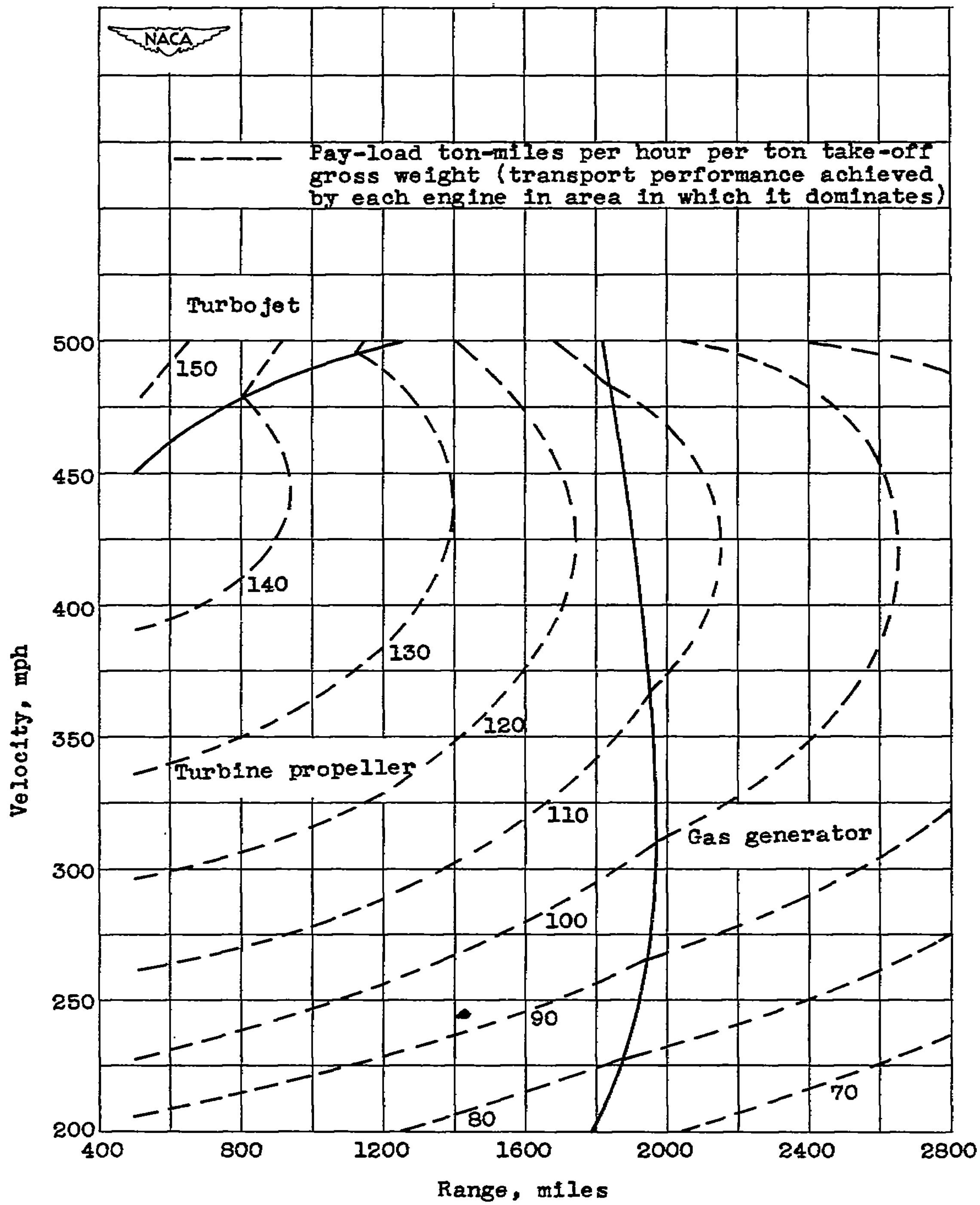
Figure 7. - Variation of ton-miles per hour per ton gross weight with range for several engines at four different velocities. Turbine-inlet total temperature, 1800° F.

[redacted]



(a) Turbine-inlet total temperature, 1400° F.

Figure 8. - Range-velocity spectrum.



(b) Turbine-inlet total temperature, 1800° F.

Figure 8. - Concluded. Range-velocity spectrum.

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