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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE

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FLIGHT INVESTIGATION OF THE COOLING CHARACTERISTICS OF A

TWO-ROW RADIAL ENGINE INSTALLATION

I - COOLING CORRELATION

By E. Barton Bell, James E. Morgan
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SUMMARY

Flight tests have been conducted to determine the cooling characteristics of a two-row radial engine at altitude in a twin-engine airplane and to investigate the accuracy with which low-altitude cooling-correlation equations can be used for making cooling predictions at higher altitudes. The test engine was operated over a wide range of conditions in level flight at density altitudes of 5000 and 20,000 feet.

Satisfactory correlation of the cooling variables was obtained at both altitudes by the NACA cooling-correlation method. By use of correlation equations developed for specific altitudes, average engine temperatures could generally be predicted to within $\pm 5^{\circ}$ F. When temperatures at an altitude of 20,000 feet were predicted by correlation equations established at 5000 feet, they were usually within -10° F of the actual measured value. Slightly more accurate results were obtained when the equations were based on the density of cooling air at the rear of the engine than when they were based on average or front densities; however, the difference was small for the range of altitudes encountered.

A relatively simple procedure is followed in this report, which may be used to evaluate the cooling performance of any conventional air-cooled power-plant installation and to determine the optimum conditions of operation for satisfactory cooling and maximum range.

INTRODUCTION

The problems involved in the correlation of important engine and cooling variables and of predicting cooling performance in flight have been the subject of many investigations. These investigations have been based to a great extent, however, on results of cooling tests made on single-cylinder test units, torque stands, and in wind tunnels.

Several important factors have prevented making consistently accurate flight cooling predictions from sea-level cooling tests. The effects of the greatly different atmospheric conditions existing at altitude and the accompanying compressibility phenomena are difficult to account for accurately because of the lack of experimental data. The distribution of cooling air may be greatly different in flight than in torque-stand tests and the effects of varying exhaust pressure further complicate the problem. The distribution of charge air is also likely to be different because of different throttle settings and changes in carburetor-entrance conditions owing to intake-scoop configuration.

Because considerable difficulty has been experienced in making flight cooling predictions from the results of ground-level test installations, a further logical step was to determine the accuracy with which a cooling correlation established in flight at a low altitude could be used for cooling predictions at higher altitudes. Flight tests to determine the cooling characteristics of a two-row radial engine were therefore conducted at the NACA Cleveland laboratory. In the present report, cooling-correlation equations are established for two altitudes by the method developed in reference 1 and the accuracy of using the low-altitude correlation for higher altitude predictions is investigated. The effect of using front, rear, and average cooling-air densities on the accuracy of the equations is also included. The equations that are developed are employed to evaluate the cooling performance and limitations of the engine installation.

In the second report to be written on this investigation, an analysis will be made of the factors affecting the cooling-air pressure distribution within the engine cowling. A study of the engine temperature distribution will be presented in the third report and some of the results of the second report will be used in an investigation of the factors controlling temperature distribution.

The tests were conducted with a twin-engine airplane in level flight at density altitudes of 5000 and 20,000 feet over as wide a range of power and cooling-air pressure drop as flight conditions would permit.

SYMBOLS

The following symbols are used throughout the report:

Cooling-Correlation Symbols

A	constant proportional to engine friction
B	constant proportional to blower power
bhp	brake horsepower
C	constant proportional to engine displacement
c_p	specific heat of air at constant pressure, 0.24 Btu per pound per °F
D	impeller diameter (0.917 for test engine), feet
g	acceleration due to gravity, 32.2 feet per second per second
ihp	indicated horsepower
isac	indicated specific air consumption, pounds per indicated horsepower per second
J	mechanical equivalent of heat, 778 foot-pounds per Btu
K,m,n	constants derived from cooling data
N	engine speed, rpm
P_{alt}	absolute pressure in exhaust manifold at altitude, inches mercury
P_{sl}	absolute pressure in exhaust manifold at sea level, inches mercury
T_a	cooling-air temperature (stagnation), °F
T_b	cylinder-barrel temperature, °F
T_c	charge-air temperature ahead of carburetor, °F
T_g	mean effective gas temperature, °F

T_{g_0}	reference mean effective gas temperature when charge-air temperature in manifold is 0° F, $^\circ$F
T_h	cylinder-head temperature, $^\circ$F
T_m	charge-air temperature in manifold calculated on dry-air basis, $^\circ$F
U	impeller tip speed, feet per second
W_c	weight of charge-air flow, pounds per second
ΔP	cooling-air pressure drop, inches water
ΔT	blower temperature rise across supercharger on dry-air basis, $^\circ$F
ΔT_g	change in mean effective gas temperature due to T_m, $^\circ$F
ρ	cooling-air density, slugs per cubic foot
ρ_f	front cooling-air density, slugs per cubic foot
ρ_o	NACA standard sea-level density, 0.002378 slugs per cubic foot
σ	ratio of cooling-air density to standard density, ρ/ρ_o
σ_f	σ based on front cooling-air density, ρ_f/ρ_o

Airplane-Performance Symbols

a	velocity of sound in air $(a = 33.42\sqrt{T + 459.4})$, miles per hour
C_D	coefficient of drag
C_L	coefficient of lift
F_c	compressibility factor
M	Mach number
q	dynamic pressure, inches water
q_c	impact pressure in compressible flow $(q_c = F_c q)$, inches water

S	wing area (602 for test airplane), square feet
T	ambient air temperature, °F
V	true airspeed, miles per hour (or ft/sec as specified)
W	airplane gross weight, pounds
η	propulsive efficiency

Pressure-Tube Symbols

H	total pressure
P	static pressure

Subscripts

ae	average engine
b	barrel
fr	front row
h	head
i	intake side of cylinder
rr	rear row
t	top of cylinder
1,2,3, etc.	axial location of pressure tubes in direction of cooling-air flow

AIRPLANE AND ENGINE

A twin-engine airplane (fig. 1) powered by two-row radial engines was used for these tests. This type of airplane afforded sufficient space for a large amount of instrumentation and allowed a wide range of operating conditions for the test engine; the service ceiling of the airplane was approximately 25,000 feet. The engines were enclosed with short-nose low-inlet-velocity cowlings having cowl flaps around the lower half of the periphery with the exception of the space at the bottom occupied by the oil-cooler duct. Charge air was admitted through twin intake ducts at the top of the cowling. The weight of the airplane during the flight tests was approximately 30,000 pounds.

The engine was of the 18-cylinder two-row radial air-cooled type with a normal rating of 1500 brake horsepower at an engine speed of 2400 rpm and a take-off rating of 1850 brake horsepower at an engine speed of 2600 rpm. The engine was equipped with a gear-driven, single-stage, two-speed supercharger having a low-blower gear ratio of 7.6:1 and a high-blower gear ratio of 9.45:1. A torquemeter having a gear ratio of 2:1 and an injection carburetor that was standard for the engine were used.

The four-bladed propeller was 13.5 feet in diameter, of the constant-speed type, and equipped with cuffs.

The fuel used throughout the tests conformed to specification AN-F-28.

INSTRUMENTATION

The engine-instrument installation was made on the right engine of the airplane. In addition to the standard flight and engine measurements, provisions were made to determine engine torque, fuel flow, weight of engine charge air, fuel-air ratio of individual cylinders, cylinder temperatures, cooling-air temperature and pressure drop, and cowl-flap angle. The engine fuel-air ratio was obtained, in most cases, by averaging the fuel-air ratio of the individual cylinders, as determined by Orsat analysis of the exhaust-gas samples. When exhaust-gas samples were not obtained, the engine fuel-air ratio was calculated from the fuel flow, which was measured by a fuel flowmeter, and the charge-air flow, which was determined by the method discussed in appendix A. Calculated values of fuel-air ratio agreed on the average with the available measured values to within ±3.5 percent. Continuous records were taken indicating airspeed, altitude, engine speed, torque, and manifold pressure. The power developed by the left engine was assumed to be equal to that of the right engine because all operating conditions were set the same on both engines.

Temperatures. - The locations for measuring cylinder temperatures are shown in figure 2. The temperatures used for correlation were obtained by thermocouples located on the rear of the head between the two top circumferential fins (T_{13}) and at the rear middle of the barrel halfway up the finning (T_6). In addition to these two thermocouples, cylinder temperatures were measured on the rear spark-plug gasket (T_{12}) and at the conventional location at the rear of the cylinder flange (T_{14}). Thermocouples T_{13} and T_6 were embedded and peened one-sixteenth inch below the surface of the cylinder and T_{14} was spot-welded to the surface of the cylinder flange.

The temperature of cooling air in front of the engine was taken as the stagnation air temperature computed from values obtained from a calibrated shielded thermocouple mounted below the fuselage. The temperature of the cooling air as it left the cylinders was obtained by thermocouples located on rakes behind each cylinder; one thermocouple was behind each head and one behind each barrel, as shown in figure 3(b). The temperature of the engine charge air was measured by two thermocouples in each of the two intake ducts leading to the carburetor. All temperatures were measured by iron-constantan thermocouples and recording potentiometers.

Pressures. - The pressure drop that was used for correlation was obtained by total-pressure tubes in front of the engine and static tubes behind the engine. The pressure in front of the engine was determined from the average of the total-pressure tubes located at the baffle entrance on the top and intake side of the front-row cylinders. The pressure at the rear of the engine was determined from the average of static-pressure tubes mounted on rakes behind the rear-row cylinders. The locations of these tubes are shown in figure 3 and described in table I. Instrumentation of the test engine included other pressure tubes (also shown in fig. 3) in order to compare pressure drops measured by the method used herein with pressure drops measured by several other methods. (See table II.) All pressures were obtained by a liquid manometer board that was photographed in flight or by recording manometers, which are accurate to within ±0.1 inch of water.

FLIGHT PROGRAM

Four series of tests were run for the purpose of obtaining data for cooling correlation at NACA density altitudes of 5000 and 20,000 feet and they were followed by other miscellaneous flights. Generally, four to six test points were obtained during each flight. The conditions for the four series of tests are listed in the following table:

Series	Brake horse-power	Density altitude (ft) (a)	Engine speed (rpm)	Pressure drop (in. water)	Carburetor setting
1	800	5,000	2400	Varied	Automatic rich
	1000	5,000	2400	---do---	Do.
	1250	5,000	2400	---do---	Do.
	1500	5,000	2400	---do---	Do.
	800	20,000	2400	---do---	Do.
	900	20,000	2400	---do---	Do.
	1000	20,000	2400	---do---	Do.
	1050	20,000	2400	---do---	Do.
	1085	20,000	2400	---do---	Do.
2	Varied	5,000	2400	[b]5.0	Automatic rich
3	800	5,000	2000	[b]4.0	Varied
	800	5,000	2400	[b]4.0	Do.
	1000	5,000	2400	[b]7.5	Do.
4	1000	5,000	Varied	[b]4.5	Automatic rich

[a]All flights at 5000-foot density altitude were made in low blower; all flights at 20,000-foot density altitude were made in high blower.
[b]Average value.

The pressure drop was varied by use of cowl flaps and by varying the airspeed. The airspeed was varied at constant power by use of the landing flaps and by lowering the landing gear.

DISCUSSION OF RESULTS

The method used for correlating the variables affecting cooling was developed in reference 1. A general form of the equation is

$$\frac{T_h - T_a}{T_g - T_h} = K \frac{{W_c}^n}{(\sigma \Delta P)^m}$$

The term $\sigma \Delta P$ as it appears in the correlation equation is used as an indication of the weight flow of cooling air passing over the cylinders. If the actual weight flow were used instead of pressure drop, one equation should suffice for all altitudes, assuming a negligible change in the over-all heat-transfer coefficient. If an equation based on $\sigma \Delta P$ is to be accurate for all altitudes, however, corrections for variations of cooling-air density across the cylinders must be applied to the pressure drop, as discussed in references 2 and 3. As a possible means of minimizing the necessity of making complex corrections for compressibility, cooling-air density measurements obtained at several general locations were used in the correlation equation to determine which density would result in the most accurate equation over a range of altitudes.

Correlation equations were obtained based on front, rear, and average cooling-air densities. The rear density was calculated from values of temperature and pressure measured behind the engine; the average density was obtained by averaging the front (stagnation) and rear densities. Curves of mean effective gas temperature T_{g_0} plotted against fuel-air ratio are shown in figure 4 and construction curves required to determine the correlation equations for front density are shown in figures 5 to 7; similar construction curves were required for the rear and average density equations but are not included. Details of the calculations are presented in appendix A. The coefficients and exponents of the equations are presented in the following table:

Altitude (ft)		5000			20,000		
Density		Front	Average	Rear	Front	Average	Rear
Heads	n	0.57	0.57	0.57	[a]0.57	[a]0.57	[a]0.57
	m	.34	.34	.34	.32	.31	.30
	K	.46	.44	.43	.46	.44	.42
Barrels	n	0.41	0.41	0.41	[a]0.41	[a]0.41	[a]0.41
	m	.37	.37	.37	.35	.33	.32
	K	.79	.76	.73	.81	.76	.72

[a]Assumed values. (See appendix A.)

From the table it may be seen that the exponent m decreases somewhat with an increase in altitude for all three densities. This decrease is largely attributed to changes in the cooling-air-flow pattern and to the effect of compressibility on the relation between cooling-air weight flow and pressure drop. The table further shows that at an altitude of 5000 feet, the exponent m is not noticeably affected by the various densities but that a change is reflected in the coefficient of the equation because of the use of the different densities. At an altitude of 20,000 feet, however, the values of the exponent m obtained when using front, rear, and average densities are slightly different. The variation, although not large, may be explained by the fact that the magnitude of rear density changes with a change in pressure drop because of the resulting difference in cooling-air temperature and pressure at the rear of the engine. This change in rear density is in contrast to a comparatively constant front density.

In order to evaluate the equations based on the three densities and to determine which of these densities will give the most accurate equation over the range of altitudes, average engine temperatures have been calculated using the 20,000-foot data in the 5000-foot equations. A summary of the results is given in the following table:

	Density	Difference between actual and calculated temperature when 20,000-foot data are used in the 5000-foot equations, °F	
		Average	Maximum
Head	Front	10	18
	Rear	7	14
	Average	9	17
Barrel	Front	8	15
	Rear	5	10
	Average	7	13

The computations show that all temperatures calculated for an altitude of 20,000 feet by means of the 5000-foot equation are less than the actual values. Furthermore, the computations based on rear density are slightly more accurate than those based on front density. Rear density, however, is difficult to calculate without knowledge of the heat rejected from the engine to the cooling air. On the other hand, the front density may be readily calculated for any anticipated flight conditions. The maximum error between the actual and calculated temperature when using the front density is only 5° F greater than when using the rear density, which indicates that unless more precise results are desired, front density can be satisfactorily used for altitudes up to 20,000 feet.

For altitudes above 20,000 feet, a Pratt & Whitney Aircraft report indicates that the use of rear density will result in the most accurate indications of weight flow; for such applications, a low-altitude correlation based on rear density would likely result in a marked improvement in accuracy over a similar correlation based on the front density.

When the equations are used for making predictions at altitudes at which they were determined, any of the three densities may be used with approximately the same degree of accuracy. For the present tests, temperatures calculated by this means were, on the average, within ±5° F of the actual measured temperatures.

In order to permit comparison of the correlation equations presented herein with those determined in other cooling investigations that are based on other methods of measuring temperature and pressure drop, figures 8 and 9 and table II are presented. Figure 8 shows the relation between average head temperatures T_{13} and average and maximum rear-spark-plug-gasket temperatures T_{12} and figure 9 shows the relation between average middle-barrel temperatures T_6 and average and maximum cylinder-flange temperatures T_{14}. Table II gives a basis for comparing the pressure drop measured by the method used herein and the pressure drop measured by three other conventional methods.

APPLICATION OF THE COOLING CORRELATION

In order to demonstrate the use of the correlation equations for evaluating airplane cooling performance, suitable curves have been calculated for a range of altitudes from sea level to 20,000 feet using the equations based on front density; pressure-drop calculations between sea level and 8000 feet were made by use of the cooling equation established at an altitude of 5000 feet and those calculations between 14,000 and 20,000 feet were made by use of the 20,000-foot equation; these ranges of altitude were arbitrarily chosen. Mean values of the constants of the two equations were used to obtain an equation for the intermediate altitudes. The 5000-foot equation could have been used throughout the range of altitudes with small error but inasmuch as the 20,000-foot equation was available, it was also used. Two curves of airplance performance (figs. 10 and 11) that facilitate determination of the value of cooling-air pressure drop for a given operating condition were also used in making these cooling-performance computations. The first of these airplane performance curves (fig. 10) shows the relation between dynamic pressure q and engine power during level flight for three cowl-flap settings and three gross weights. This relation was computed using the following aerodynamic equations in conjunction with a curve of ${C_L}^2$ plotted against C_D that was determined in flight:

$$q = \frac{1}{2}\rho\ V^2 \tag{1}$$

$$W = C_L\ (\rho/2)\ SV^2 \tag{2}$$

$$\eta\ \text{bhp} = C_D\ (\rho/2)\ \frac{SV^3}{550} \tag{3}$$

and

$$\sqrt{\sigma}\,\text{bhp} = \frac{C_D S q^{3/2}}{\eta\, 550} \frac{\sqrt{2}}{\sqrt{\rho_o}} \qquad (4)$$

when V is in feet per second. The procedure for determining such a curve in flight is described in reference 4. A value for q was assumed and equation (2) was solved for the value of C_L corresponding to the specified weight condition. The curve was then used to determine the value of C_D corresponding to the known value of C_L and the specified cowl-flap setting. Equation (4) was then solved for $\sqrt{\sigma}$ bhp using a value of 0.85 for the propulsive efficiency η inasmuch as estimates indicated that it remained practically constant for the wide range of conditions encountered with the test airplane. Computations made in this manner were used to establish the curves shown in figure 10. From this figure, knowing the density ratio, brake horsepower, gross weight of the airplane, and cowl-flap setting, the dynamic pressure q may be found and then converted to impact pressure q_c ($q_c = F_c q$). The second curve used to compute pressure drop is that of $\Delta P/q_c$ plotted against cowl-flap opening (fig. 11); this curve represents the average of a number of test points. As can be seen from figure 11, for this installation opening the cowl flaps fully produces about 60 percent more pressure drop than that obtained with closed cowl flaps at the same indicated airspeed. Because of the decreased airspeed when the cowl flaps are opened at constant power, however, the effective increase in pressure drop is only about 40 percent.

Temperature predictions based on pressure drops estimated by use of figures 10 and 11 showed an average error of $\pm 7^\circ$ F for 105 runs and thus are nearly as accurate as those based on measured pressure drops. An example of the procedure used for making temperature predictions is shown in appendix B.

A comparison of the cooling-air pressure drop available with the cooling-air pressure drop required to limit maximum cylinder-head temperatures to specified values is presented in figures 12 and 13. This comparison is shown for four operating conditions at maximum gross weight (36,000 lb) and a light gross weight (26,000 lb) in NACA standard atmospheric conditions and Army summer air. The fuel-air ratio specified for each operating condition is the approximate value that would be metered by a carburetor with standard setting.

The abrupt upward displacement of the curves of required pressure drop, when the supercharger is shifted from low- to high-blower ratio, is due to the additional cooling load imposed by the increased charge consumption and the rise in mixture temperature with the increased supercharger compression. The decrease in pressure drop available that results from increased gross weight is small at high power but more pronounced at low power as may be seen by comparing figure 12(a) (military power) with figure 13(b) (50 percent rated power).

Although the specific cooling performance of this particular installation may not be of general interest, a discussion of the curves serves to illustrate their application. The curves show, in general, that the engine in this installation will cool satisfactorily with standard carburetor setting under all power conditions in NACA standard atmospheric conditions. In Army summer air, which is representative of more severe cooling conditions, however, it may be seen from figure 13(a) that enrichment would be required to limit the maximum cylinder temperatures to specified values for high-blower operation at 70 percent rated power. Figure 13(b) indicates satisfactory cooling for 50 percent rated power with open cowl flaps for 26,000 pounds gross weight but shows that the temperature limits would be slightly exceeded for 36,000 pounds gross weight in Army summer air. A leaner carburetor setting at this power, if it allows satisfactory operation, would help to cool the engine because the specified fuel-air ratio is approximately that at which the cylinder temperatures peak and either richer or leaner operation would decrease the temperatures.

As a means of demonstrating the combined effects of cowl flaps and fuel-air ratio on specific range when they are varied to limit cylinder temperature to the maximum allowable value, figure 14 is presented; curves for two altitudes are included. Figure 14 shows that a sizable increase in the specific range of the airplane may be realized by operating as lean as feasible and providing adequate cooling by means of cowl flaps rather than cooling with excess fuel.

For the specific conditions of figure 14 at an altitude of 8000 feet and an air temperature of 70° F (corresponding to Army summer air), a 14-percent increase in specific range is possible by operating at a fuel-air ratio of 0.063 and three-fourths open (30°) cowl flaps rather than at a fuel-air ratio of 0.08 and one-fourth open (10°) cowl flaps, when the same cylinder-head temperatures prevail in both cases. The fuel-air ratio of 0.08 is approximately that which would be metered by the standard carburetor. The decrease in true airspeed due to the increased cowl-flap opening would be about 10 miles per hour for these conditions. The gaps in the curves for air temperatures of 46° and 56° F represent a range of fuel-air ratio at which the specified head temperature would be exceeded;

whereas the end points of all the curves indicate the limits of fuel-air ratio beyond which the head temperature would be less than the specified value.

These cooling-performance calculations have been made only for the cylinder head but they could be made equally as well for the barrel. For this particular installation, cooling of the barrel was less critical than of the head for the operating conditions encountered and thus the evaluation of cooling performance has been based on head-temperature limits. The performance curves indicate no serious cooling problem present for this installation in level flight when operating in the range of conditions that were assumed.

The procedure followed herein for predicting cooling performance is applicable to any airplane equipped with air-cooled engines. The use of this procedure allows an evaluation of the cooling characteristics of the power-plant installation over a wide range of operating conditions from data obtained in the comparatively few flights required to establish the cooling-correlation equation and airplane performance characteristics.

SUMMARY OF RESULTS

From flight cooling tests of a two-row radial engine in a twin-engine airplane, the following results were obtained that are applicable to level flight:

1. Satisfactory correlation of the cooling variables has been obtained in flight tests at density altitudes of 5000 and 20,000 feet.

2. Average engine temperatures calculated for a specific altitude by use of the correlation equations developed for that altitude were, on the average, within ±5° F of the actual measured temperature and were always within ±16° F.

3. When average engine temperatures at an altitude of 20,000 feet were predicted by cooling-correlation equations established at an altitude of 5000 feet, they were usually within -10° F of the actual measured value. Slightly more accurate results were obtained when the equations were based on the density of cooling air at the rear of the engine than when they were based on average or front densities; however, the difference was small for the range of altitudes encountered.

4. For specified airplane operating conditions, estimates of cooling-air pressure drop were made with which it was possible to compute cylinder temperatures comparable to the accuracy of calculations based on measured pressure drop.

5. The relatively simple procedure followed herein was used to evaluate the cooling performance of this typical air-cooled power-plant installation and to determine optimum conditions of operation for satisfactory cooling and maximum range.

Aircraft Engine Research Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio, November 30, 1945.

APPENDIX A

DETAILS OF CORRELATION CALCULATIONS

In reference 1 a method was established for correlating the important variables affecting engine cooling. This method equates the quantity of heat transferred from the products of combustion to the cylinder walls with the quantity of heat rejected from the cylinders to the cooling air. One form of the equation may be written

$$\frac{T_h - T_a}{T_g - T_h} = K \frac{{W_c}^n}{(\sigma \Delta P)^m}$$

The methods of obtaining the values of the variables in the equations are discussed in the following sections.

Head temperature, cooling-air temperature, and pressure drop. - The value of head temperature T_h used was the average temperature obtained by the 18 thermocouples T_{13}; T_a was taken as stagnation air temperature and ΔP was the average of the pressure drops measured by tubes in the baffle entrances of the front-row cylinders and at the rear of the rear-row cylinders as described in the section on instrumentation.

Mean effective gas temperature. - The mean effective gas temperature T_g is a function of fuel-air ratio, inlet manifold temperature, and exhaust pressure for a given engine with a fixed spark timing (references 1 and 5). An equation expressing this relation is

$$T_g = T_{g_0} + \Delta T_g \tag{1}$$

The change in mean effective gas temperature ΔT_g is dependent on the temperature of the charge in the manifold T_m. Inasmuch as the temperature in the manifold is difficult to measure by a thermocouple owing to the partial vaporization of fuel, it was approximated by summing up the charge-air inlet temperature T_c and the computed temperature rise across the supercharger ΔT. The value of T_m on a dry-air basis is expressed as

$$T_m = T_c + \Delta T \tag{2}$$

From reference 6, an approximate expression for blower temperature rise ΔT on a dry-air basis (without fuel) is derived as

$$\Delta T = \frac{U^2}{Jc_p g} \tag{3}$$

Substituting for ΔT in equation (2) gives

$$T_m = T_c + \frac{U^2}{Jc_p g} \tag{4}$$

From single-cylinder tests (reference 5), it has been found that a change of 1° F in the temperature of the charge resulted in a change of about 0.8° F in the mean effective gas temperature for the heads. This effect was assumed to hold true for multicylinder flight tests because of satisfactory results in wind-tunnel tests on multicylinder engines (reference 7). Accordingly

$$T_g = T_{g_0} + 0.8\ T_m \tag{5}$$

When equation (4) is combined with equation (5), the following expression may be written

$$T_g = T_{g_0} + 0.8\left(T_c + \frac{U^2}{Jc_p g}\right) \tag{6}$$

For the engine tested the following equations were computed for the heads:

Low blower

$$T_g = T_{g_0} + 0.8\left[T_c + 22.1\left(\frac{N}{1000}\right)^2\right] \tag{7}$$

High blower

$$T_g = T_{g_0} + 0.8\left[T_c + 34.3\left(\frac{N}{1000}\right)^2\right] \tag{8}$$

For the barrels, the change in mean effective gas temperature was taken as 0.5° F when the charge temperature was changed 1° F. Accordingly, the equations for the barrels are:

Low blower

$$T_g = T_{g_0} + 0.5\left[T_c + 22.1\left(\frac{N}{1000}\right)^2\right] \tag{9}$$

High blower

$$T_g = T_{g_0} + 0.5\left[T_c + 34.3\left(\frac{N}{1000}\right)^2\right] \tag{10}$$

Equations (7) to (10) were used to obtain curves of T_{g_0} against fuel-air ratio (fig. 4) for a density altitude of 5000 feet as discussed in a later section on calculation procedure. Because the mean effective gas temperature is affected by a change in exhaust pressure, a separate T_{g_0} curve was calculated for 20,000 feet (fig. 4). This curve was obtained by reducing the values of the 5000-foot curve by a fixed percentage determined from reference 6, which shows the drop in T_g with exhaust pressure.

Charge-air weight flow. - For any given fuel-air ratio, the charge-air flow to an engine is approximately proportional to the indicated horsepower. Thus, a single curve should be obtained when charge-air flow per unit time per indicated horsepower, or indicated specific air consumption, is plotted against fuel-air ratio. Once such a curve has been established, it is necessary to know only the fuel-air ratio and indicated horsepower in order to determine charge-air flow. Reference 7 presents an equation that expresses indicated horsepower in terms of readily measured variables. The general form of the equation is

$$\text{ihp} = \text{bhp} + \left[A + B(W_c)\right]\left(\frac{N}{1000}\right)^2 - C\,(P_{sl} - P_{alt})\,\frac{N}{1000} \qquad (11)$$

The specific forms of equation (11) for the test engine are:

Low blower

$$\text{ihp} = \text{bhp} + \left[27 + 8.64\ W_c\right]\left(\frac{N}{1000}\right)^2 - 1.74(P_{sl} - P_{alt})\,\frac{N}{1000} \qquad (12)$$

High blower

$$\text{ihp} = \text{bhp} + \left[27 + 13.36\ W_c\right]\left(\frac{N}{1000}\right)^2 - 1.74(P_{sl} - P_{alt})\,\frac{N}{1000} \qquad (13)$$

In order to determine the curve of indicated specific air consumption plotted against fuel-air ratio from the flight data, it was first necessary to make a carburetor calibration of compensated metering pressure against weight of charge-air flow. This calibration was made in a test cell with the carburetor intake ducts assembled in place to simulate the flight installation. Values of W_c obtained from the metering pressure read in flight were then used in conjunction with equations (12) and (13) to establish the indicated specific air consumption curve shown in figure 15. This curve was assumed to be valid for all conditions of power and altitude. For the correlation equations, the weight of charge-air flow was then calculated from this curve and from equations (12) and (13).

Calculation procedure. - In flight tests it is difficult to obtain and hold constant the exact desired conditions; more steps are therefore required to reduce flight cooling data than to reduce test-stand or wind-tunnel data where desired conditions can be maintained within close limits. Because it is impossible to maintain constant cooling-air pressure drop, weight of charge-air flow, and fuel-air ratio in these tests, it was necessary to apply corrections for the variation of these factors.

The procedure used in obtaining the final cooling-correlation equations at an altitude of 5000 feet was as follows: The data taken in flight series 1 and 2 were used in conjunction with a previously established curve of T_g plotted against fuel-air ratio from reference 8 to obtain the first approximate exponents n and m. It was necessary to use this T_g curve established in other tests because the runs in flight series 1 and 2 could not be made at a constant fuel-air ratio of 0.08, which would have been desirable. By substitution of the approximate exponents n and m in the general form of the cooling equation, the values of K were computed for the individual data points. When these values of K were used, the values of $(T_h - T_a)/(T_g - T_h)$ were corrected for small variations in charge-air flow W_c and a new and more nearly correct exponent m was obtained. In the same fashion, the equations were corrected for small variations in $\sigma\Delta P$ and a new exponent n was obtained. By the use of these corrected values of n and m, new values of K were calculated for several runs made at a fuel-air ratio of approximately 0.08. When these calculations of K were made, the value of T_g, based on a value of T_{g_0} of 1086° F, was obtained from equation (7) or (8). The average of these new values of K was used with the data from flight series 3 to plot a curve of T_{g_0} against fuel-air ratio (fig. 4). Final values of the exponents n and m were obtained by using this corrected T_g curve. The final correlation curves for all runs were then plotted in the form $(T_h - T_a)/(T_g - T_h)$ against $\left(W_c^{n/m}\right)/\sigma\Delta P$ using the final corrected exponents n and m and the T_g curve that had been established.

Inasmuch as the range of charge-air flow at an altitude of 20,000 feet was very limited, it was impossible to determine the exponent n from data obtained at that altitude. The assumption was therefore made that altitude does not affect the heat-transfer process from the combustion gases to the cylinder walls and the values of the exponent n for an altitude of 20,000 feet were assumed to be the same as the value determined for 5000 feet. Final values of pressure-drop exponent and the correlation equations for an altitude of 20,000 feet were determined in the same manner as at 5000 feet.

A similar procedure was used for determining the correlation curves for the cylinder barrels.

APPENDIX B

SAMPLE PREDICTION OF CYLINDER TEMPERATURE

In order to calculate the maximum spark-plug-gasket temperature the following conditions in level flight were assumed:

Brake horsepower	1700
Engine speed, rpm	2600
Fuel-air ratio	0.10
Supercharger	low blower
Cowl flap	closed
Pressure altitude, feet	5000
Free-air temperature, $^{\circ}F$	81
Free-air density ratio	0.798
Airplane gross weight, pounds	36,000

Cooling-air pressure drop. - In order to obtain $\sigma \Delta P$ for use in the correlation equation, the dynamic pressure q must be found and converted to the impact pressure q_c by means of the compressibility factor F_c, which is determined from Mach number M.

The dynamic pressure q may be obtained from figure 10. For the given conditions

$$\sqrt{\sigma}\ \text{bhp} = \sqrt{0.798} \times 1700 = 1519$$

therefore, from figure 10

$$q = 32.3 \text{ inches water}$$

$$F_c = 1 + \frac{M^2}{4} + \frac{M^4}{40} + \frac{M^6}{1600} + \ldots$$

$$M = V/A$$

$$V = 45.08\sqrt{q/\sigma} = 287 \text{ miles per hour}$$

$$a = 33.42\sqrt{T + 459.4} = 777 \text{ miles per hour}$$

$$M = \frac{287}{777} = 0.37$$

$$F_c = 1 + \frac{(0.37)^2}{4} + \frac{(0.37)^4}{40} + \ldots$$

$$F_c = 1.034$$

$$q_c = F_c q = 1.034 \times 32.3 = 33.5 \text{ inches water}$$

From figure 11, for closed cowl flaps $\Delta P/q_c = 0.28$. Therefore

$$\Delta P = 0.28 \times 33.5 = 9.38 \text{ inches water}$$

Front cooling-air density ratio σ_f, computed from stagnation conditions, may be obtained by use of the equation

$$\sigma_f = \sigma_{\text{free air}} \left(1 + 0.2M^2\right)^{2.5} = 0.854$$

Therefore

$$\sigma_f \Delta P = 0.854 \times 9.38 = 8.0 \text{ inches water}$$

Cooling-air temperature. - The cooling-air temperature will be taken as the ambient air temperature plus the full adiabatic rise (stagnation conditions)

$$T_a = T + 1.79 \left(\frac{V}{100}\right)^2$$

$$T_a = 81 + 1.79 \left(\frac{287}{100}\right)^2 = 96^\circ \text{ F}$$

Mean effective gas temperature. - In the absence of carburetor heat or charge cooling, T_c is assumed equal to T_a. When the value of T_a is substituted for T_c and the value of T_{g_0} is used that corresponds to a fuel-air ratio of 0.10 (fig. 4) in equation (7) of appendix A

$$T_g = 900 + 0.8 \left[96 + 22.1 \ (2.6)^2\right] = 1096^\circ \text{ F.}$$

Charge-air flow. - In order to determine W_c, a value of 0.00172 is obtained for isac from figure 15 for the given fuel-air ratio of 0.10. When ihp is expressed as W_c/isac and substituted in equation (12) (appendix A)

$$W_c = 0.00172 \left[1700 + (27 + 8.64\ W_c)\left(\frac{2600}{1000}\right)^2 - 1.74\ (29.92 - 24.89)\ \frac{2600}{1000}\right]$$

or

$$W_c = 3.56 \text{ pounds per second}$$

When the values that have been obtained for $\sigma\Delta P$, T_a, T_g, and W_c are substituted in the general form of the cooling equation and the values of exponents and coefficients that were obtained at an altitude of 5000 feet and based on front density are used, the equation may be solved for T_h.

$$\frac{T_h - 96}{1096 - T_h} = 0.456 \frac{(3.56)^{0.57}}{(8.0)^{0.34}}$$

or

$$T_h = 413^\circ \text{ F}$$

By use of figure 8 the maximum rear-spark-plug-gasket temperature corresponding to an average temperature T_{13} of 413° F is approximately 466° F.

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TABLE I - PRESSURE-TUBE LOCATIONS

Pressure tube (a)	Type of tube	Pressure-measurement location	Circumferential position on cylinder	Axial location in direction of cooling-air flow (in.)	Radial location from cylinder-flange base (in.)
H_{h2i}	Total head	Between fin and baffle at head baffle entrance	Intake side	3/16 downstream of entrance baffle curl	$7\frac{13}{16}$
H_{h2t}	Total head	Between fin and baffle at head baffle entrance	Center of cylinder	3/16 downstream of entrance baffle curl	$12\frac{7}{8}$
H_{b2i}	Total head	Between fin and baffle at barrel baffle entrance	Intake side	3/16 downstream of entrance baffle curl	$3\frac{3}{16}$
P_{h4}	Closed-end static	Rear of head on rake	Center of cylinder	7/8 downstream of barrel fins	7
P_{b4}	Closed-end static	Rear of barrel on rake	Center of cylinder	7/8 downstream of barrel fins	$3\frac{7}{8}$

[a]A sketch of the pressure-tube installation and an explanation of the symbols are presented in figure 3.

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TABLE II - COMPARISON OF PRESSURE DROP MEASURED BY FOUR METHODS

Pressure-drop designation (1)	Method	$\Delta P/q_c$		$\Delta P/\Delta P_1$	
		Cowl flaps full open	Cowl flaps closed	Cowl flaps full open	Cowl flaps closed
	Heads				
[2]ΔP_1	$\left(\frac{H_{h2i} + H_{h2t}}{2}\right)_{fr} - (P_{h4})_{rr}$	0.45	0.28	1.00	1.00
ΔP_2	$\left(\frac{H_{h2i} + H_{h2t}}{2}\right)_{fr} - (P_{h7})_{rr}$	.45	.29	1.00	1.04
ΔP_3	$\left(\frac{H_{h2i} + H_{h2t}}{2}\right)_{fr} - (P_{h5})_{rr}$	.50	.33	1.11	1.18
ΔP_4	[3]$(H_{h1}) - (P_{h4})_{ae}$	.37	.24	.82	.86
	Barrels				
[2]ΔP_1	$(H_{b2i})_{fr} - (P_{b4})_{rr}$	0.40	0.24	1.00	1.00
ΔP_2	$(H_{b2i})_{fr} - (P_{b7})_{rr}$	.43	.27	1.07	1.12
ΔP_3	$(H_{b2i})_{fr} - (P_{b6})_{rr}$	.42	.26	1.05	1.08
ΔP_4	[3]$(H_{b1}) - (P_{b4})_{ae}$	.30	.19	.75	.79

[1]A sketch of the pressure-tube installation and an explanation of the symbols are presented in figure 3.
[2]Method used for correlation.
[3]In front of cylinders 2, 6, 11, and 16 only.

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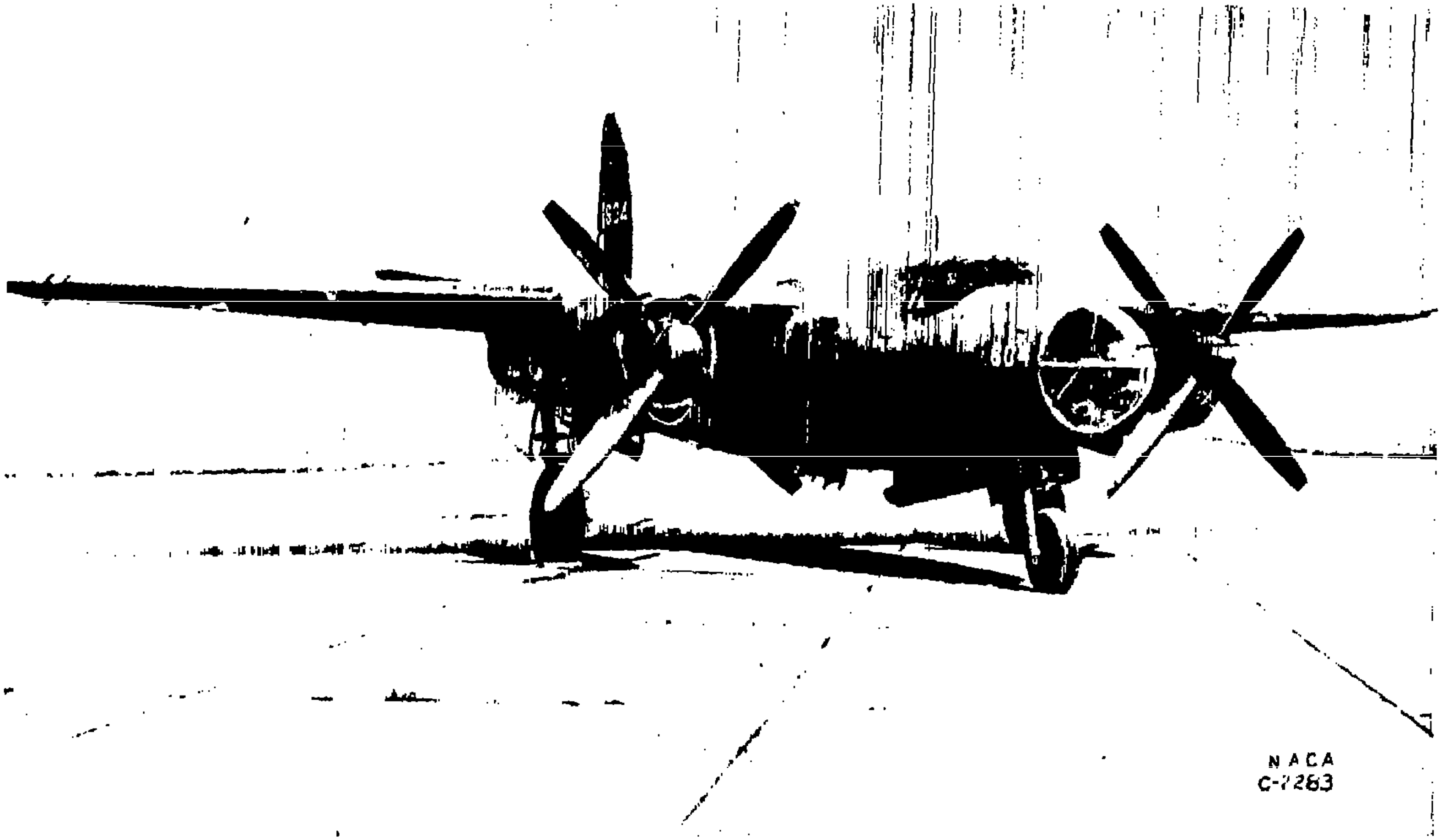

Figure 1. - Three-quarter front view of test airplane.

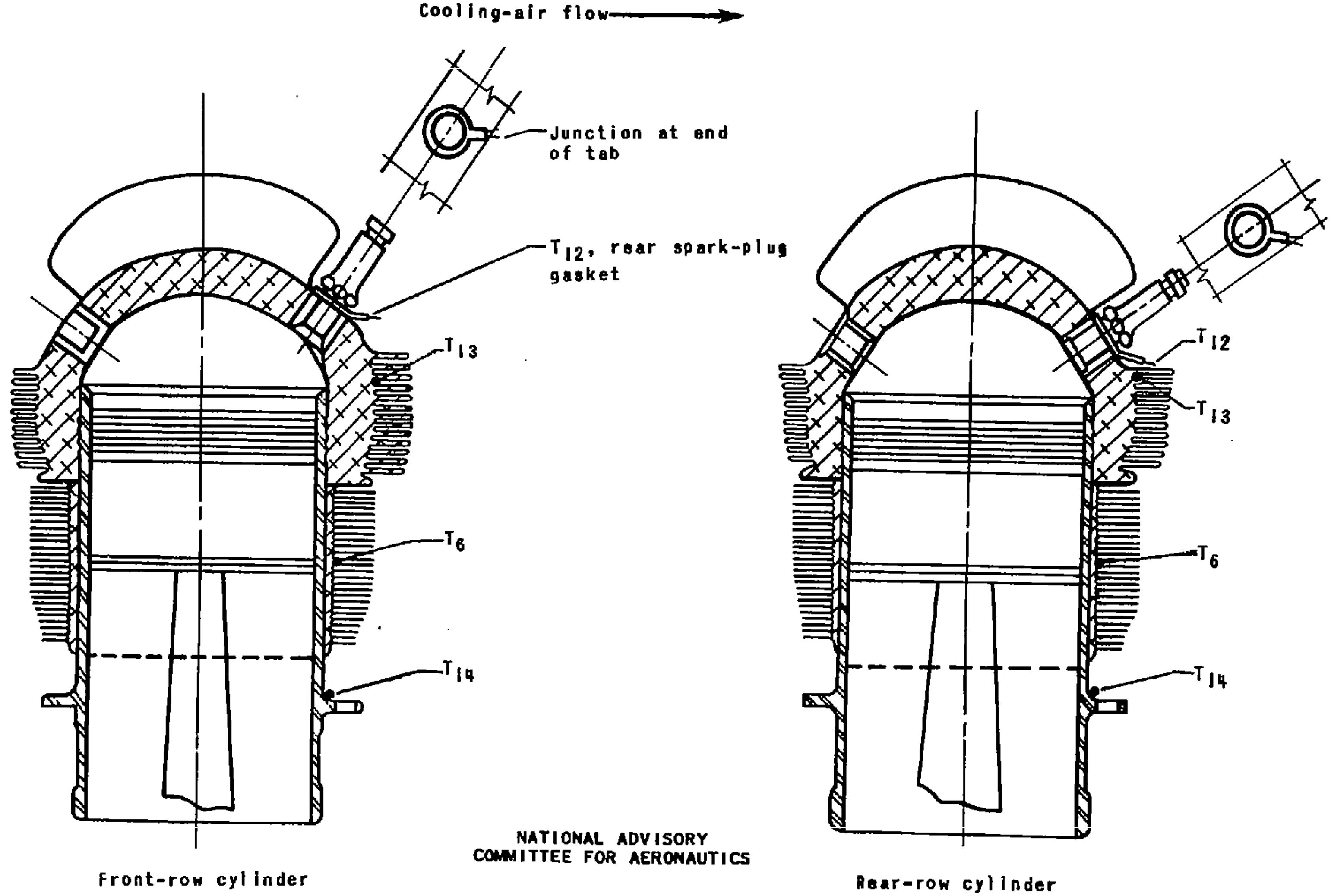

Figure 2. - Side sectional view of test-engine cylinders showing thermocouple locations.

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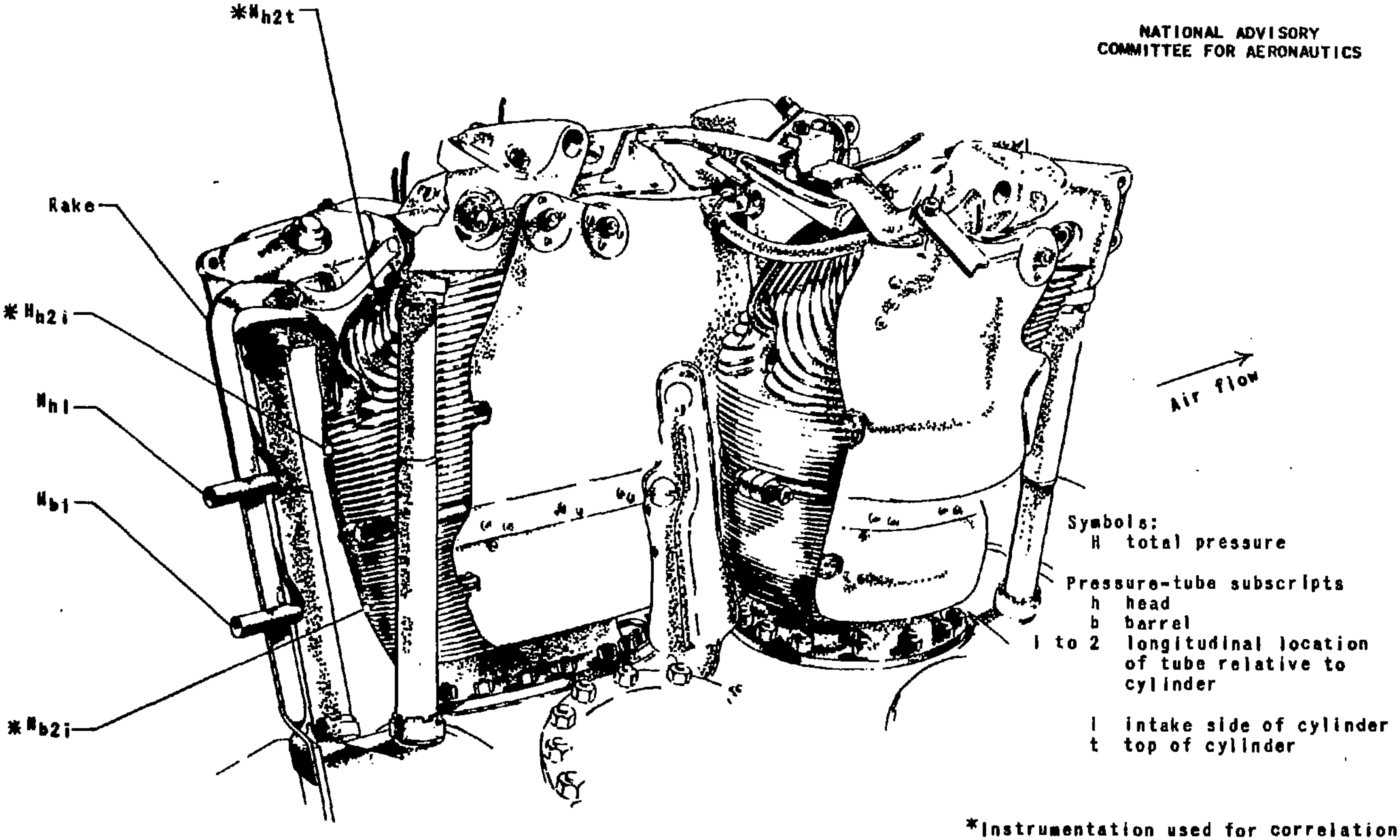

(a) Front three-quarter view.

Figure 3. - Front- and rear-row cylinders showing pressure-tube and cooling-air thermocouple installation.

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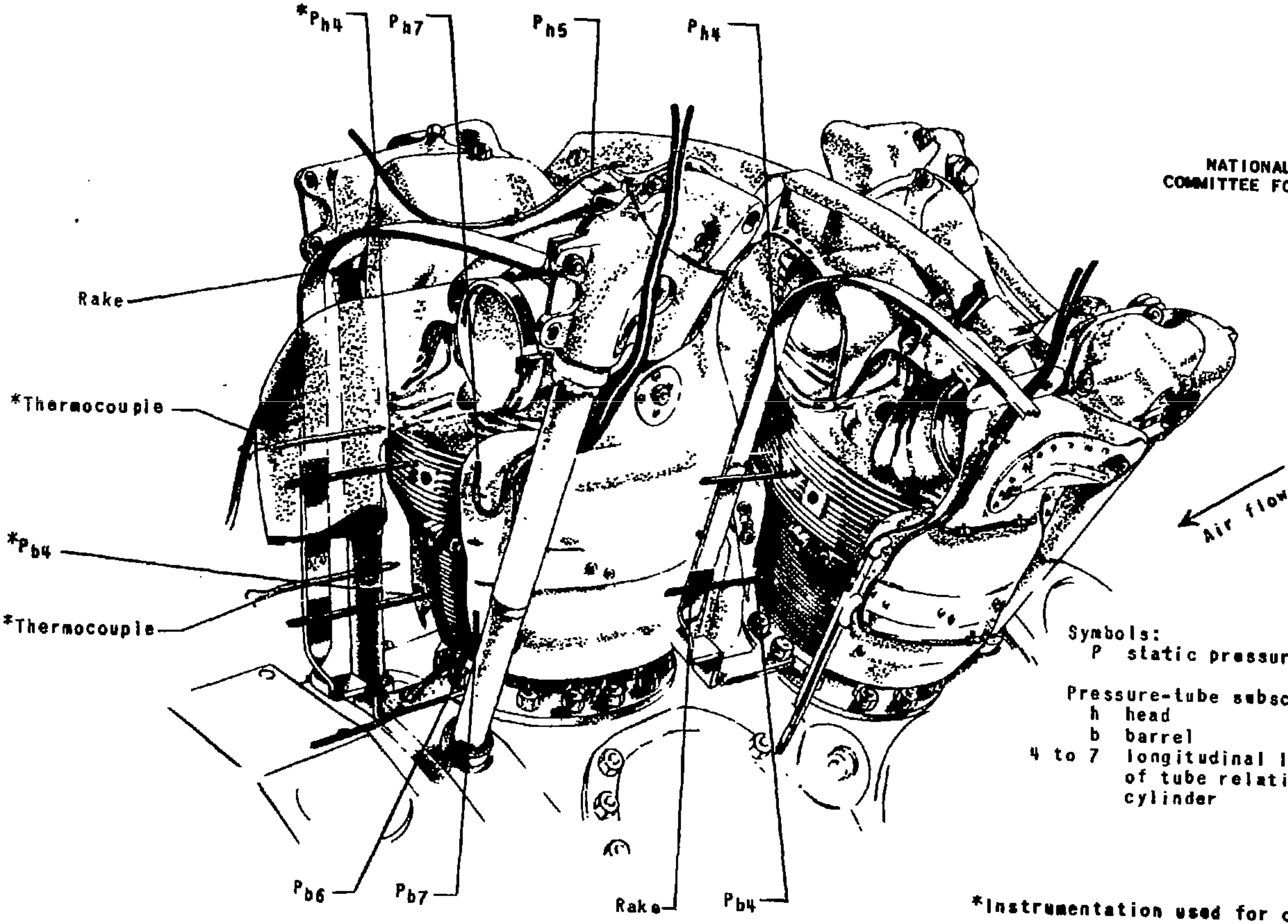


Symbols:
P static pressure

Pressure-tube subscripts:
h head
b barrel
4 to 7 longitudinal location of tube relative to cylinder

*Instrumentation used for correlation

(b) Rear three-quarter view.

Figure 3. - Concluded.

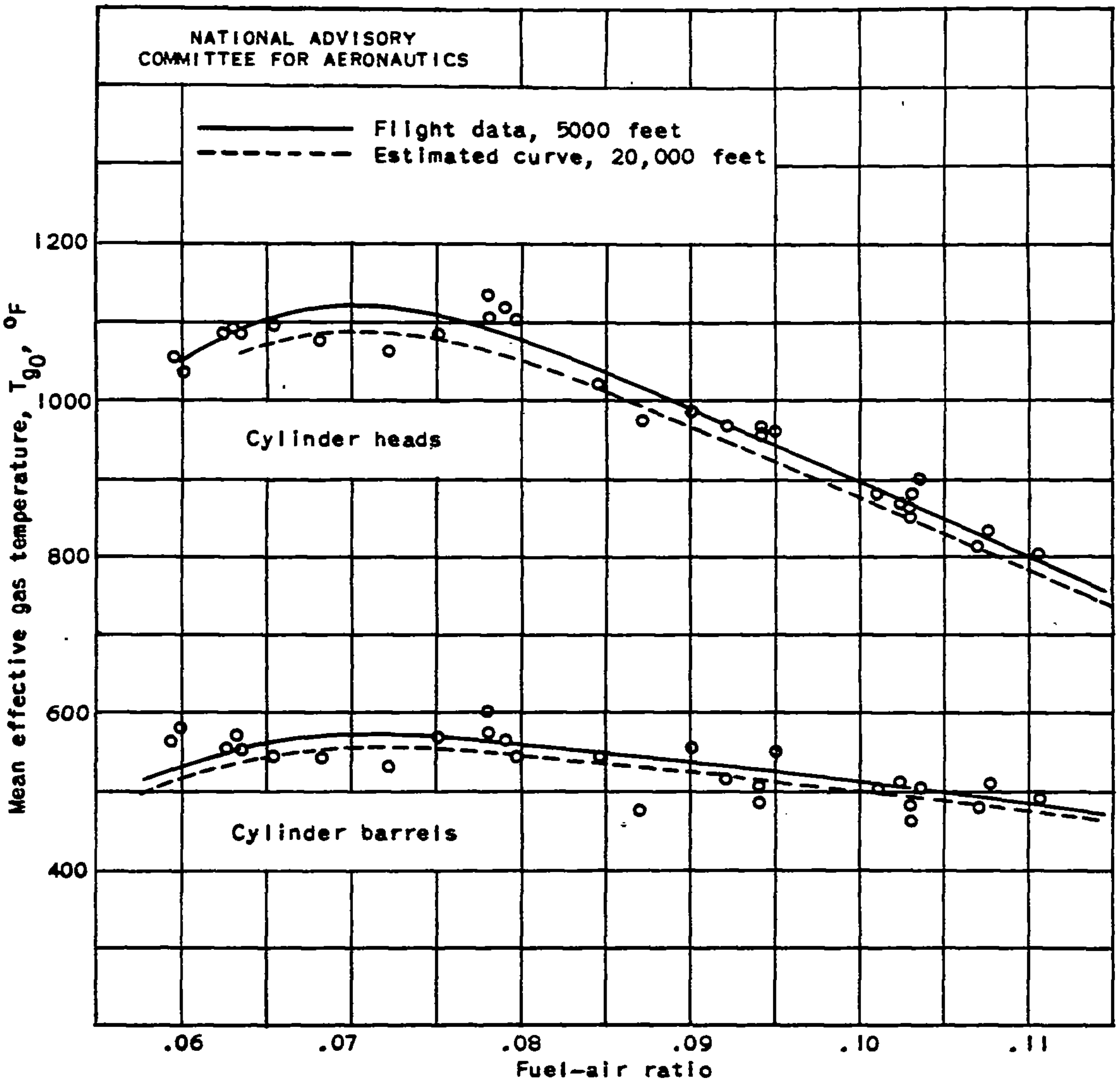


Figure 4. - Variation of mean effective gas temperature with fuel-air ratio for cylinder heads and barrels at density altitudes of 5000 and 20,000 feet.

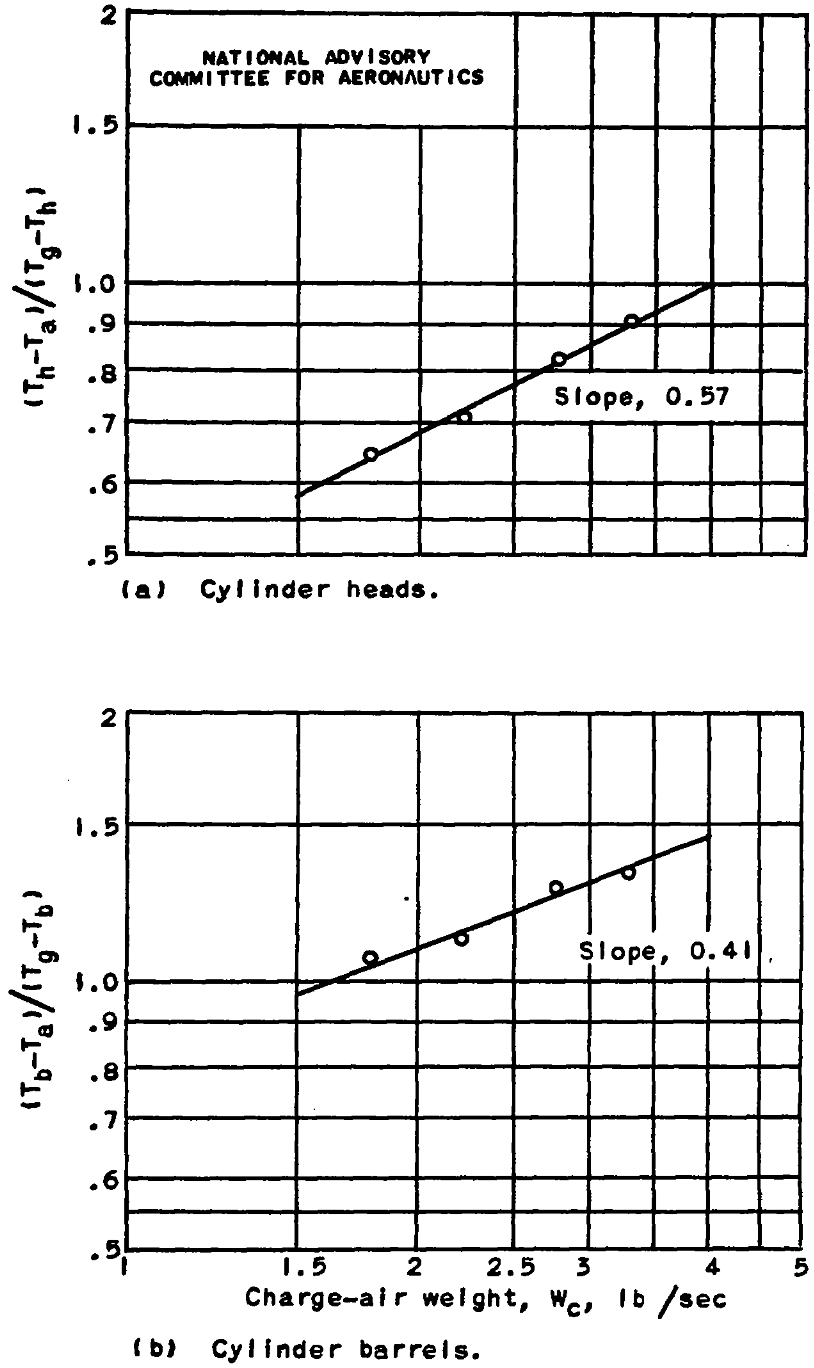
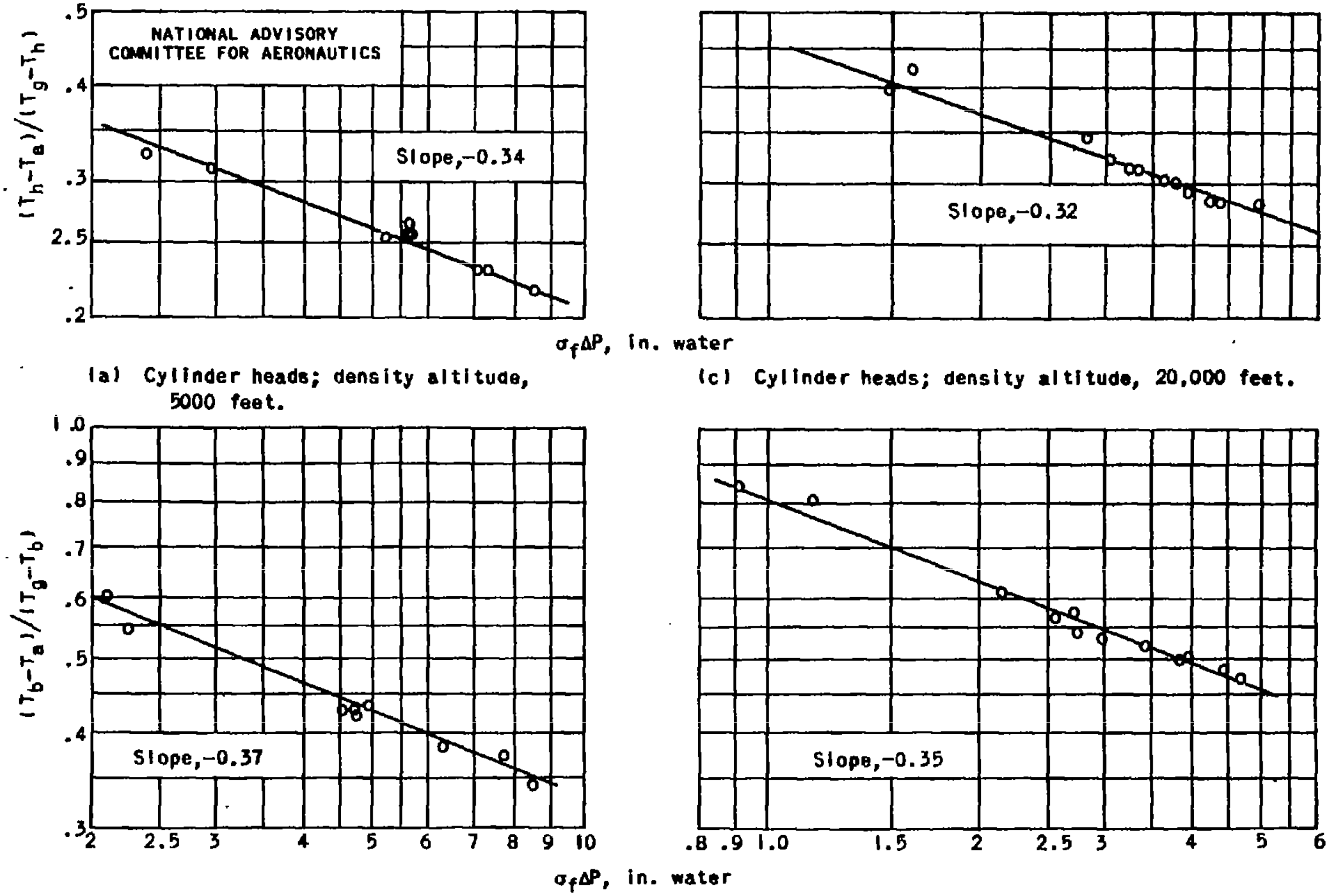


Figure 5. - Variation of $(T_h-T_a)/(T_g-T_h)$ and $(T_b-T_a)/(T_g-T_b)$ with charge-air weight W_c for cylinder heads and barrels at density altitude of 5000 feet.



(a) Cylinder heads; density altitude, 5000 feet.

(c) Cylinder heads; density altitude, 20,000 feet.

(b) Cylinder barrels; density altitude, 5000 feet.

(d) Cylinder barrels; density altitude, 20,000 feet.

Figure 6. – Variation of $(T_h-T_a)/(T_g-T_h)$ and $(T_b-T_a)/(T_g-T_b)$ with cooling-air pressure drop $\sigma_f\Delta P$ for cylinder heads and barrels.

	Pressure drop (in. water) Density altitude (ft)		Engine power (bhp) Density altitude (ft)		Engine speed (rpm) Density altitude (ft)	
	5000	20,000	5000	20,000	5000	20,000
○	Varied	Varied	800	800	2400	2400
□	Varied	Varied	1000	900	2400	2400
◇	Varied	Varied	1250	1000	2400	2400
△	Varied	Varied	1500	1050	2400	2400
▽	Constant	Varied	800	1085	Varied	2400
◁	Constant	------	Varied	----	2400	----

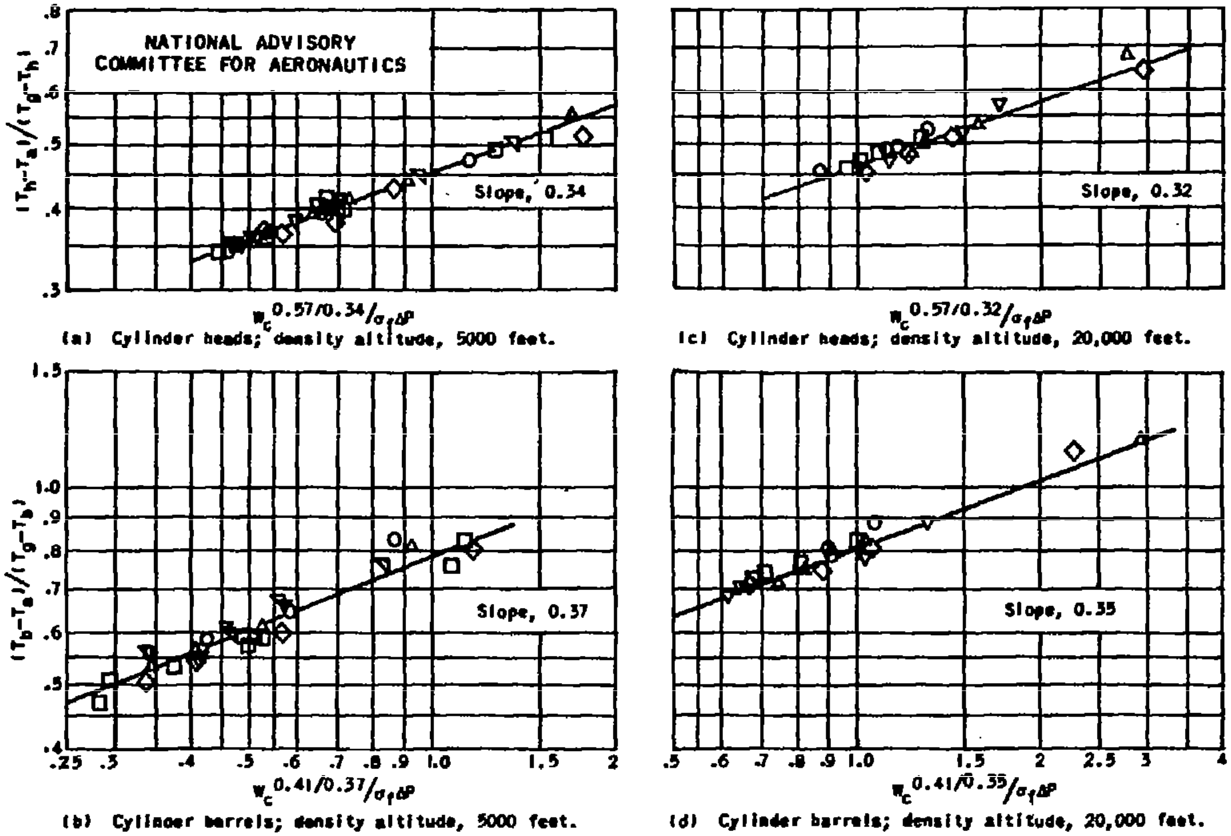


Figure 7. - Cooling-correlation curves for cylinder heads and barrels.

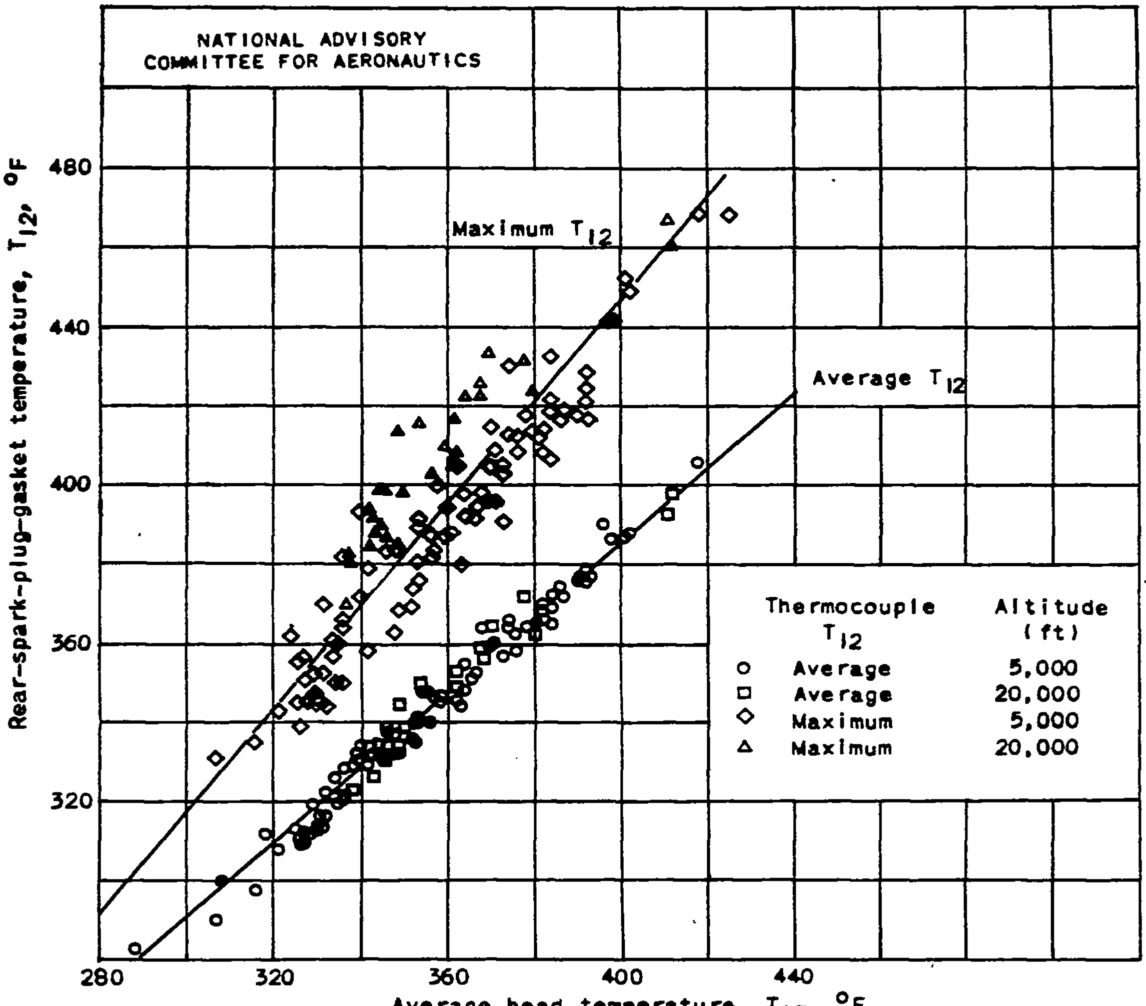


Figure 8. – Variation of maximum and average rear-spark-plug-gasket temperatures with average head temperature T_{13}.

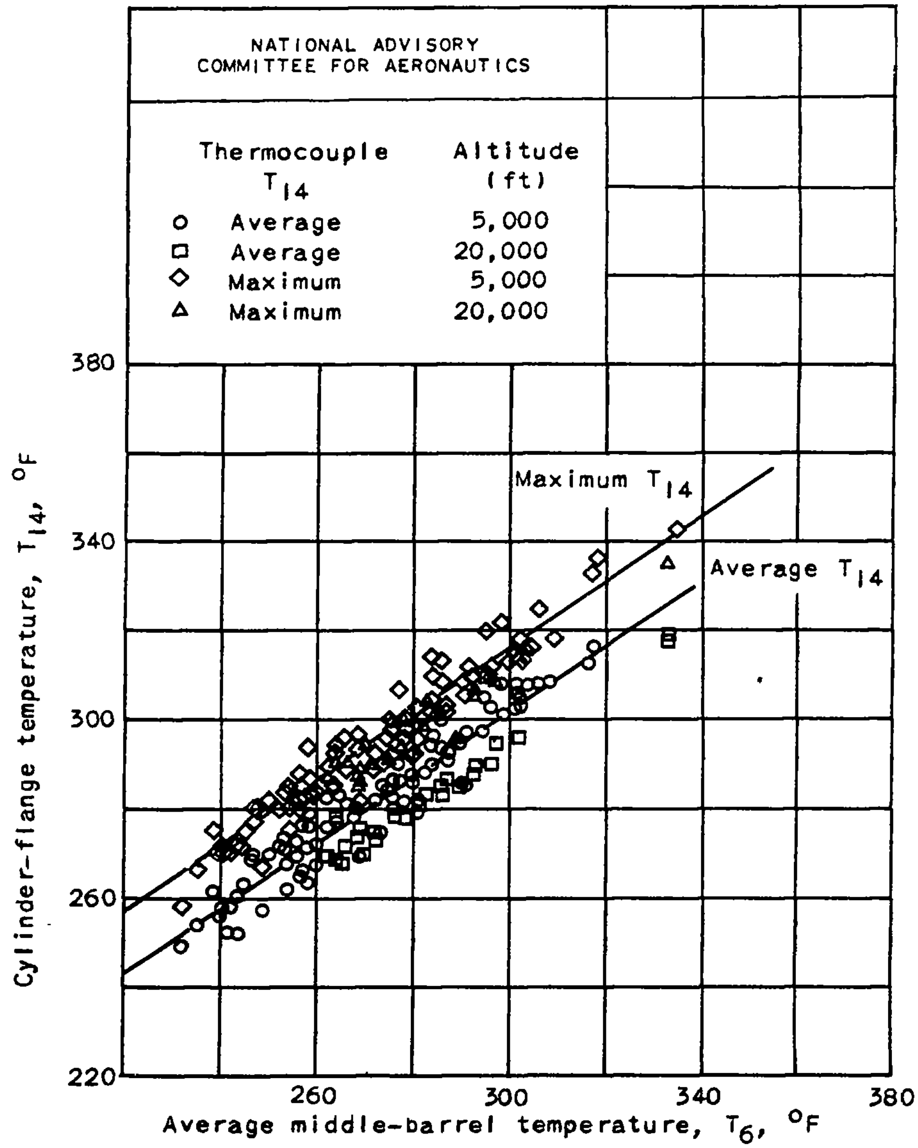


Figure 9. – Variation of maximum and average cylinder-flange temperatures T_{14} with average middle-barrel temperature T_6.

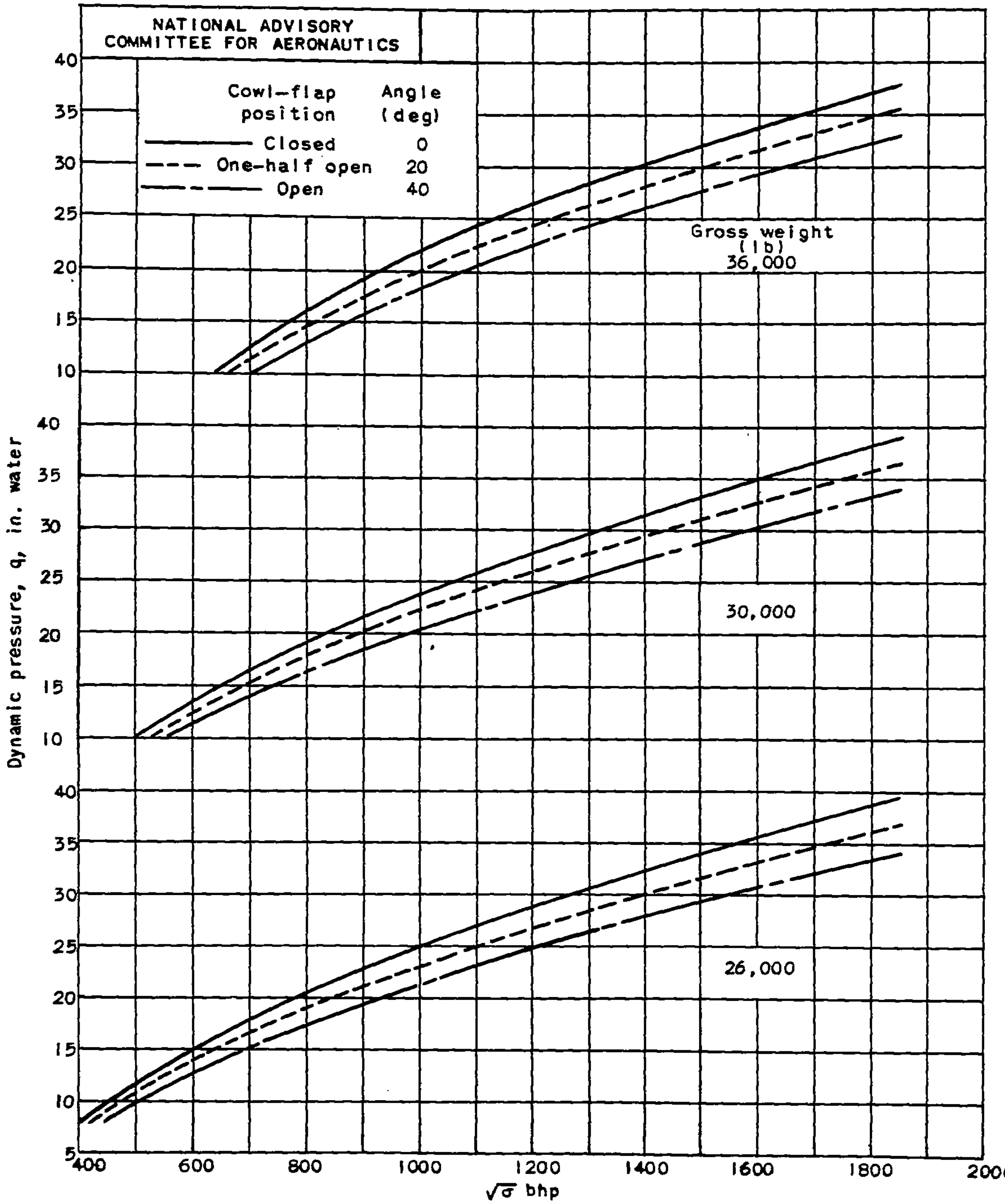


Figure 10. - Computed variation of dynamic pressure q with $\sqrt{\sigma}$ bhp. Propulsive efficiency, 0.85; level flight.

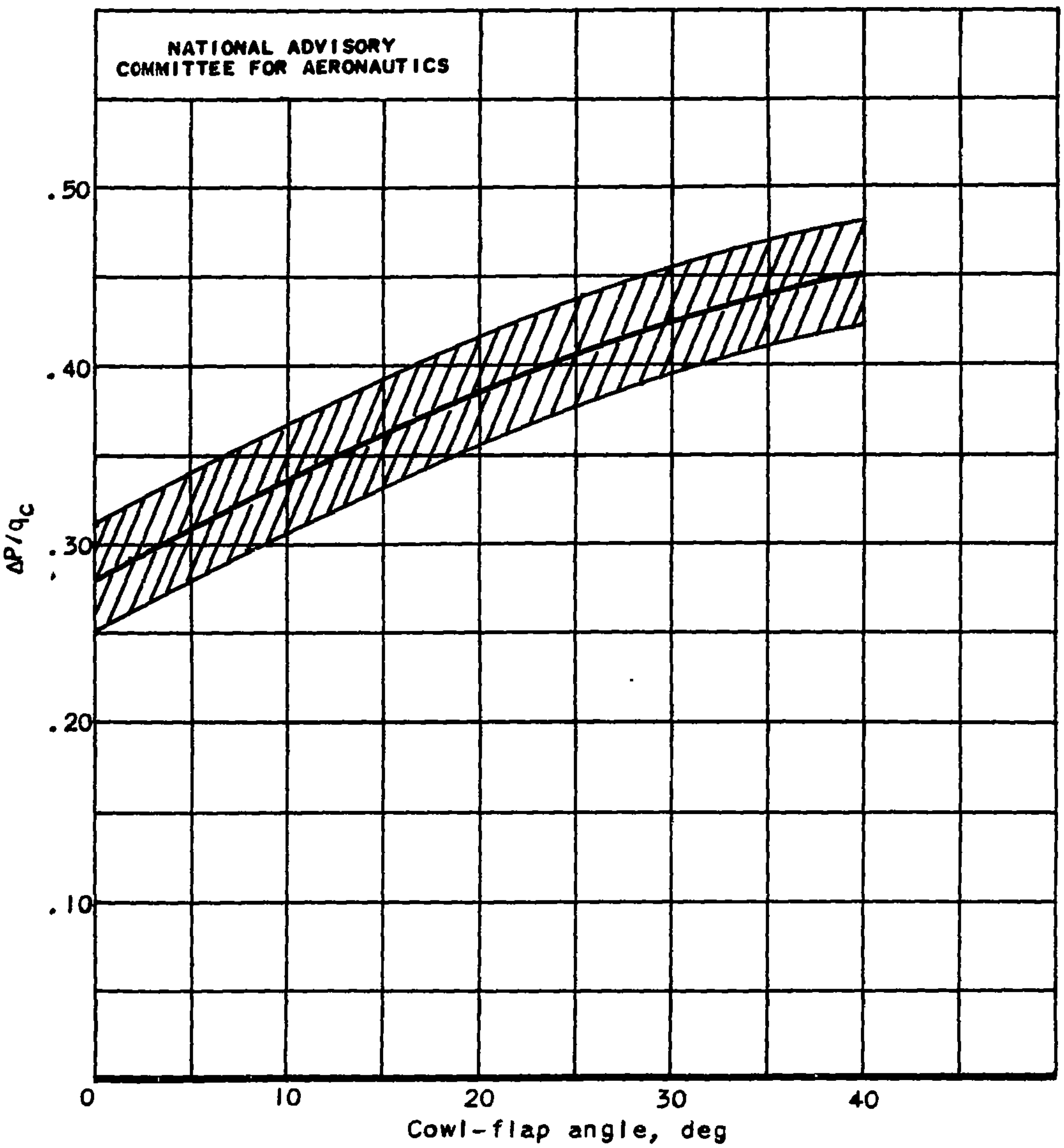
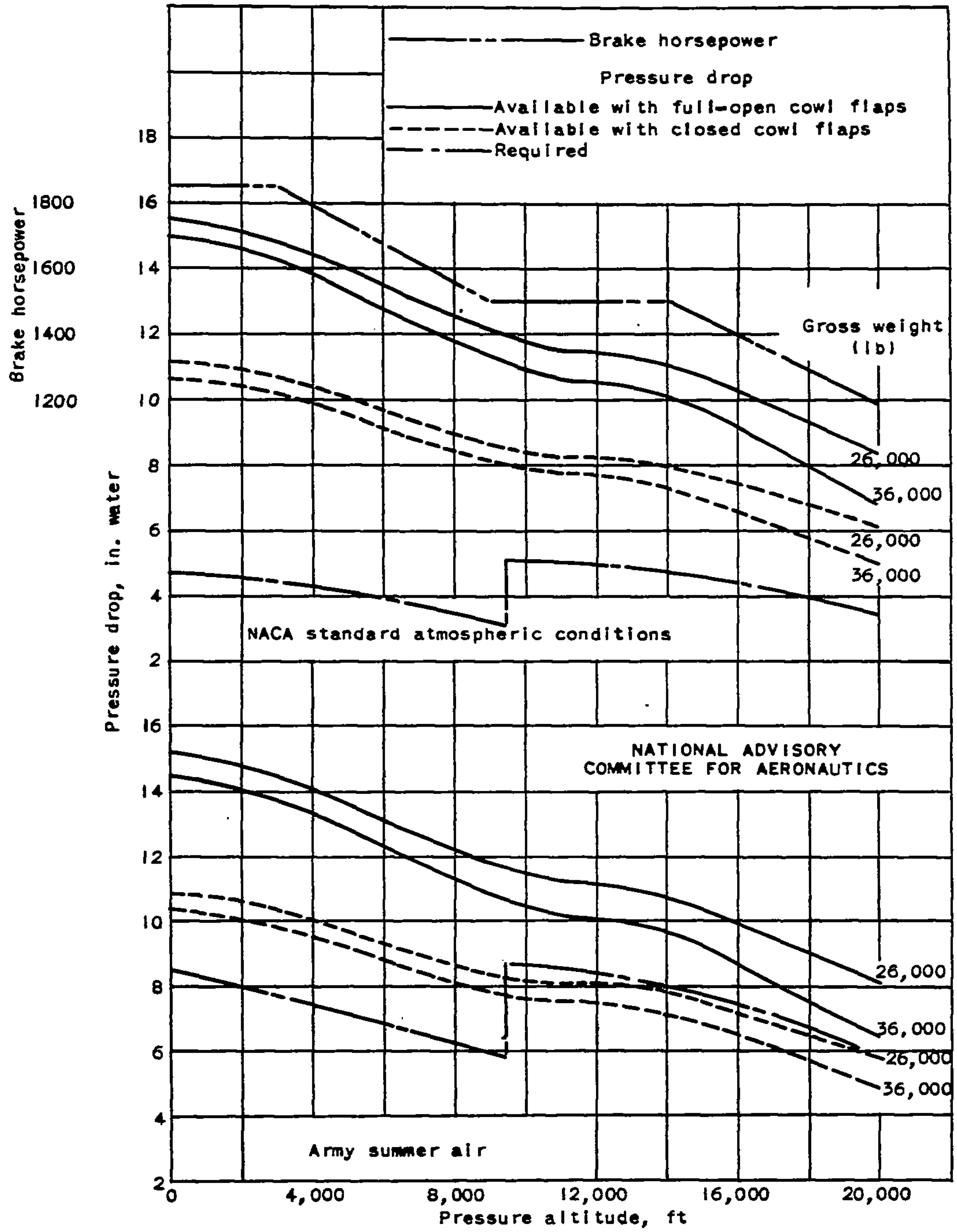
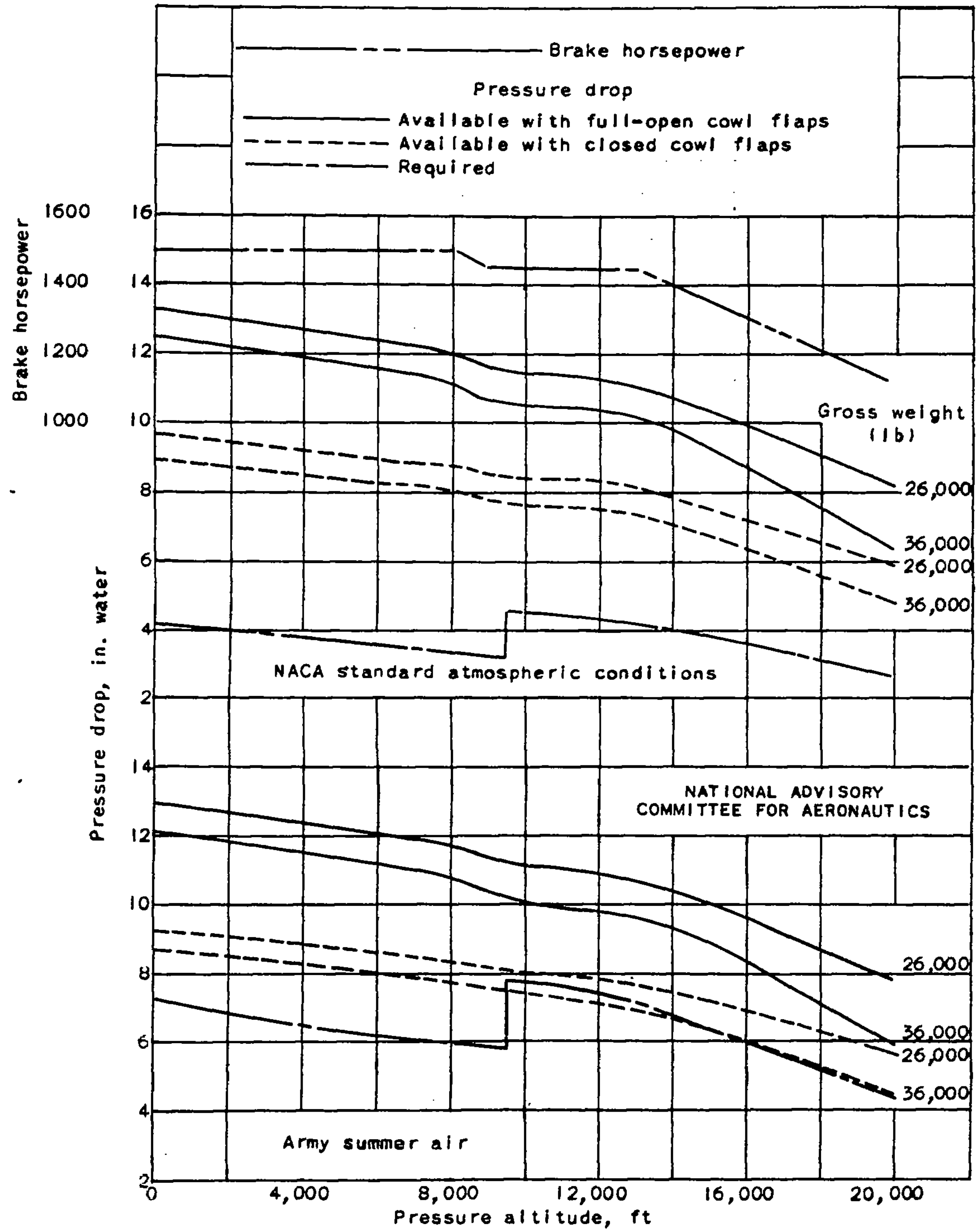


Figure 11. - Variation of $\Delta P/q_c$ with cowl-flap angle for cylinder heads.



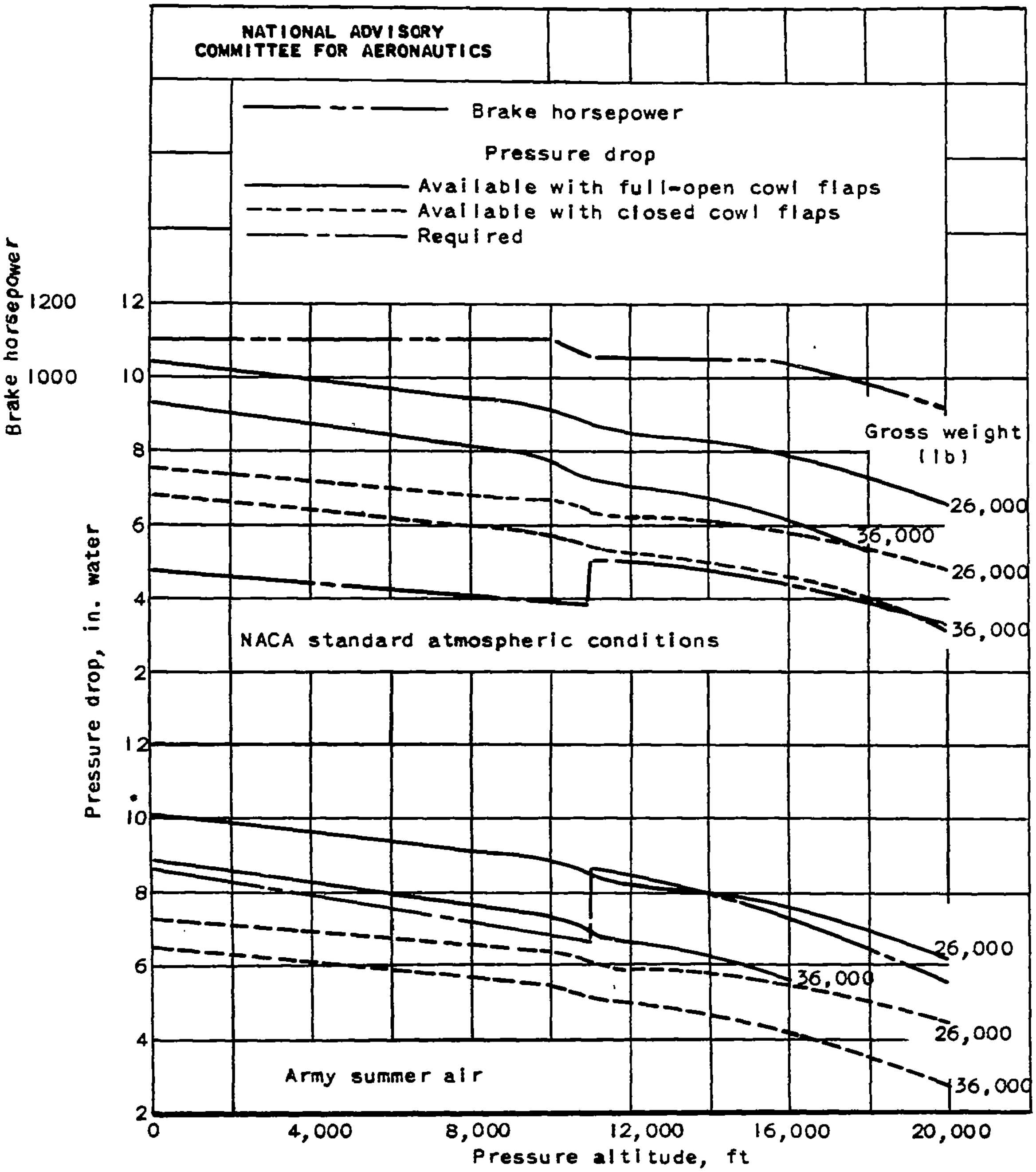
(a) Military power operation; fuel-air ratio, 0.100; engine speed, 2600 rpm.

Figure 12. – Comparison of pressure drop available for cooling with pressure drop required to limit maximum spark-plug-gasket temperature to 500° F. Supercharger shifted to high blower at 9500 feet; level flight.



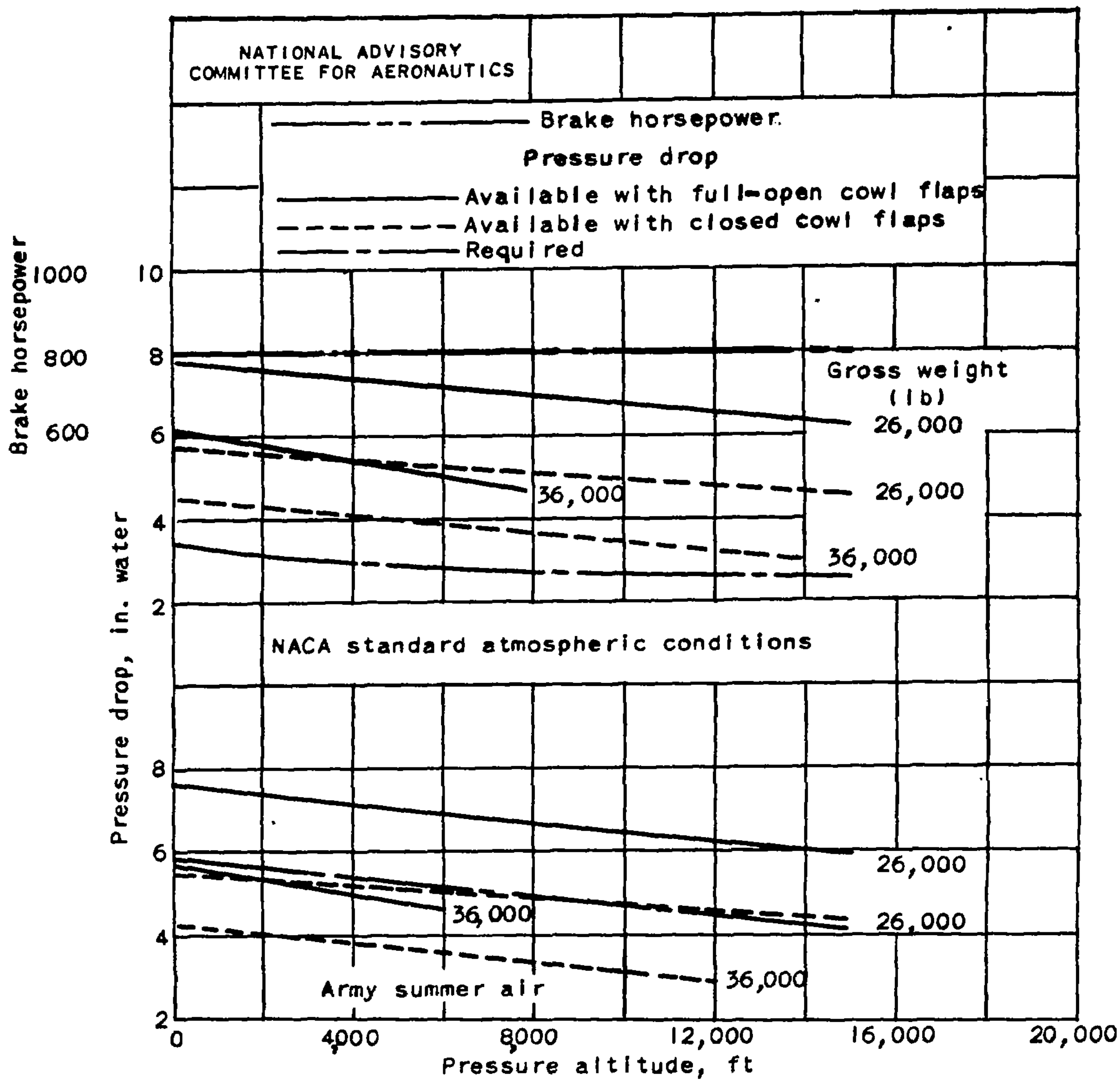
(b) 100 percent rated power to critical altitude; fuel-air ratio, 0.095; engine speed, 2400 rpm.

Figure 12. - Concluded.



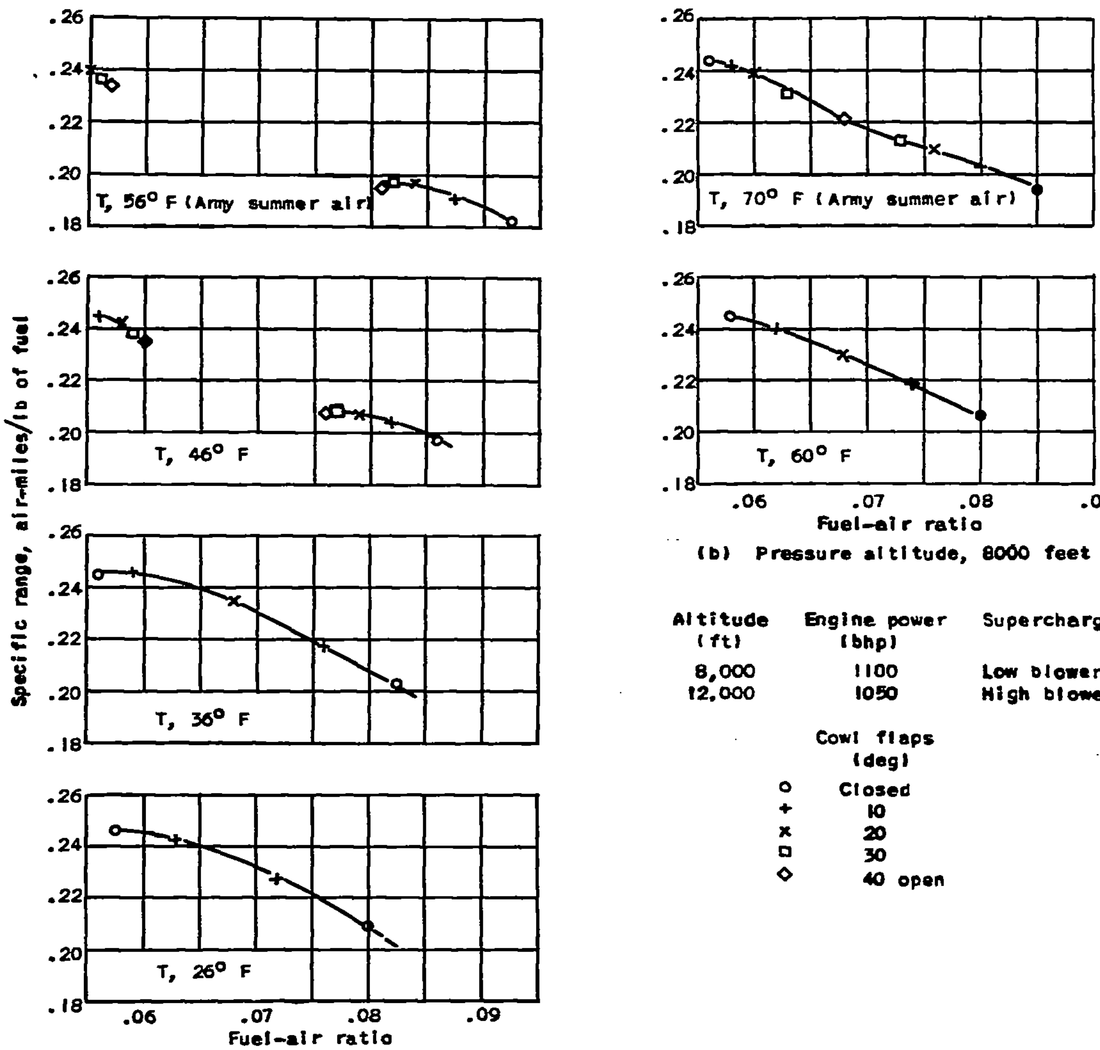
(a) 70 percent rated power to critical altitude; fuel-air ratio, 0.08; engine speed, 2100 rpm; supercharger shifted to high blower at 11,000 feet.

Figure 13. - Comparison of pressure drop available for cooling with pressure drop required to limit maximum spark-plug-gasket temperature to 450° F. Level flight.



(b) 50 percent rated power; fuel-air ratio 0.068; engine speed, 1800 rpm; supercharger, low blower.

Figure 13. - Concluded.



(a) Pressure altitude, 12,000 feet.

(b) Pressure altitude, 8000 feet

Altitude (ft)	Engine power (bhp)	Supercharger
8,000	1100	Low blower
12,000	1050	High blower

	Cowl flaps (deg)
o	Closed
+	10
x	20
□	30
◇	40 open

Figure 14. - Computed variation of specific range with fuel-air ratio when cowl flaps are varied to maintain maximum spark-plug-gasket temperature of 450° F. Gross weight, 30,000 pounds; engine speed, 2100 rpm.

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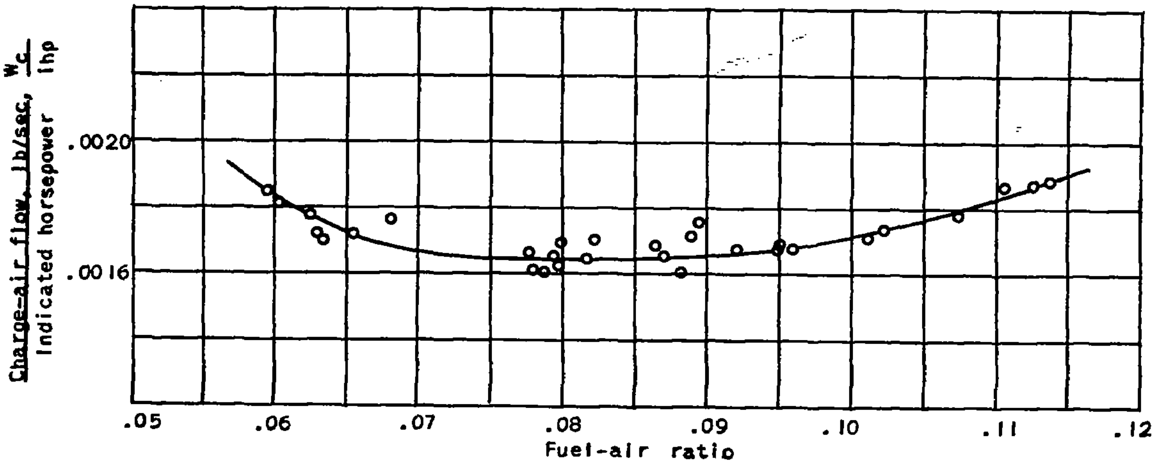

Figure 15. - Variation of indicated specific air consumption with fuel-air ratio. Density altitude, 5000 feet.

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Morgan, James E.
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ABSTRACT

Tests were conducted on twin-engine airplane in level flight at altitudes of 5000 and 20,000 feet over wide range of operating conditions. Accuracy with which low-altitude cooling-correlation equations can be used for making cooling predictions were investigated. By the use of correlation equations developed for specific altitudes, average engine temperatures can be predicted to within ±5°F. Method for correlating the variables affecting cooling is discussed.

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