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TURBOSUPERCHARGER-ROTOR TEMPERATURES IN FLIGHT

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SUMMARY

Temperatures of a turbosupercharger rotor were measured in flight for a variety of conditions by thermocouples, the leads of which were brought away from the turbine by means of rotating slip rings and stationary brushes.

A consistent and almost linear relation was shown between turbine temperature at the outer edge of the rim and the effective exhaust-gas temperature at the surfaces of the blades on three successive flights at cruising power. Similar, but not so well defined, variations were obtained at two other power conditions. During flights at numerous power conditions, turbine temperatures varied widely and no acceptable method of correlation could be determined.

It was estimated that the maximum temperature at the rim of the rotor at rated altitude would be approximately 1175° F for this particular installation.

INTRODUCTION

The high rotational speeds at which a turbosupercharger operates impose large stresses on the turbine, whereas the extreme temperatures encountered reduce the ability of the turbine to withstand those stresses. The most critical conditions exist along the blades because they are continually immersed in exhaust gases at temperatures higher than those at which present commercial materials can maintain the strength required in current applications. Sufficient cooling must be provided to keep the blade temperatures within safe limits. As trends toward greater speeds and higher exhaust temperatures progress, the problem of providing adequate cooling becomes more critical.

An investigation was undertaken by the NACA to evaluate the effects of the various factors involved upon turbine temperatures. The proposed investigation included the measurement of turbine temperatures in flight under a variety of operating conditions, the determination of the effects of individual factors upon turbine temperatures by means of a series of ground tests in an altitude chamber, and an analysis and correlation of the data obtained during these two phases. The turbine temperatures measured in flight are reported herein.

For the flight program the Army Air Corps recommended an airplane on the basis of satisfactory turbosupercharger performance over an extended period of service. A means of measuring turbine temperatures was devised, bench-tested, and installed. The ensuing flights were made at the NACA Langley Field laboratory during 1942 but the information as originally released received very limited distribution.

Maintenance problems not associated with the experimental apparatus led to the termination of the flight investigation before completion of the program. Data at the rated altitude of the turbosupercharger are therefore meager.

TEST INSTALLATION

Test equipment. - Turbine temperatures were measured on a commercial turbosupercharger installed in a single-engine pursuit airplane. The turbosupercharger was located on the under side of the fuselage directly beneath the engine; the rotor and a portion of the nozzle box extended beyond the cowling into the air stream.

The cooling cap originally furnished with the turbosupercharger was a conventional convection-type cap, which directed a stream of air against the rim of the turbine. It was necessary to redesign the cap to provide room for the turbine-temperature measuring apparatus. Passage area through the redesigned cap was adjusted to give cooling-air flows equal to those of the original cap with equal pressure drops across the two. Figure 1 shows the redesigned cap in position.

A bleeder was installed in the air duct leading from the turbosupercharger compressor outlet to the carburetor to make possible a variation in turbosupercharger load independent of engine conditions.

Instrumentation. - Chromel-alumel thermocouples were used to measure the temperatures at four points on the outer surface of the turbine disk. Three of these points were located on the rim of the disk at $4\frac{23}{32}$-, $4\frac{15}{32}$-, and $4\frac{5}{32}$-inch radii; the fourth point was at a radius of $2\frac{1}{2}$ inches.

The thermocouples were of 28-gage oxide-coated wires that were individually welded to the surface of the turbine at the hot junction. Each pair of leads was encased in 0.050-inch outside-diameter stainless-steel tubing into which a ceramic cement was forced under pressure to serve as an insulator. These tubes were clamped to the outer surface of the turbine.

Thermocouple leads were brought away from the turbine through rotating chromel and alumel slip rings and stationary brushes of the same materials. The slip rings had an outside diameter of $1\frac{1}{16}$ inches and the brushes were $\frac{1}{8}$-inch-diameter buttons. The brushes were mounted on $2\frac{1}{2}$-inch spring-steel arms and were designed to contact the sides rather than the rims of the rings. Contact was made only during the periods in which turbine temperatures were taken. Figure 2 shows the thermocouples and slip rings installed on the turbine. The brush assembly is also shown. Thermal electromotive forces from these turbine thermocouples were taken by a null method with a small potentiometer mounted in the cockpit of the airplane for this purpose.

Some idea of the magnitude and distribution of temperatures in the gases between the cooling cap and the turbine rotor was obtained from 12 chromel-alumel thermocouples installed in that space. Each pair of 28-gage thermocouple leads was encased in an 0.050-inch outside-diameter stainless-steel tube with a filler of ceramic cement. These thermocouple tubes were clamped to the outside surface of the cap and, at the desired locations (fig. 3), were allowed to project through drilled holes approximately one-eighth inch into the space between the cap and the turbine rotor.

The velocity head in the cooling-cap inlet was measured by two static taps and a total-head tube located in the circular inlet section $1\frac{3}{4}$ inches from the entrance. A sea-level calibration of weight flow against velocity head served as the basis from which weight flow at altitude was calculated.

Exhaust-gas temperatures were obtained 4 inches upstream of the nozzle-box inlet by means of a quadruple-shielded chromel-alumel thermocouple.

A straight section $2\frac{1}{2}$ inches in diameter and 40 inches in length was incorporated in the bleeder duct to insure suitable flow conditions at a measuring plane. A static tap, a total-head tube inserted to one-third the duct diameter, and a thermocouple were used to obtain data from which bleeder air weight flow could be calculated. A butterfly valve controlled from the cockpit permitted regulation of the air flow in flight.

The temperature rise through the compressor was obtained from unshielded chromel-alumel thermocouples in the inlet and outlet ducts. This value was used to calculate the approximate power required by the compressor.

Electromotive forces from all of the thermocouples except those on the turbine disk were automatically recorded at least once a minute. In addition, automatic pressure recorders were used to obtain a continuous record of the following variables:

Indicated air speed
Velocity head in cooling-cap inlet
Manifold pressure
Velocity head in bleeder duct
Bleeder-duct static pressure

On all pressure and temperature records, a timing device marked 1-second intervals in order that an accurate time relation could be established. All automatic apparatus was electrically driven and controlled from a single switch in the cockpit. This switch was momentarily tripped each time a turbine temperature was taken and the resulting traces on the films in the recorders were used to establish the time relation between turbine temperature and other data.

Values for the following variables were read from indicating instruments in the cockpit before and after each run:

Engine speed
Altitude
Turbine speed
Free-air temperature
Nozzle-box pressure
Fuel consumption
Total pressure - compressor-inlet duct
Static pressure - compressor-outlet duct

TESTS

Test procedure. - Seventeen test flights were made. During the first nine flights, when the desired altitude was reached, the airplane was leveled off, a designated power condition was established, and 5 minutes was allowed for temperatures to approach equilibrium. During this period, such readings as were not automatically recorded were noted by the pilot. At the end of the period, the electromotive forces of the four turbine thermocouples were taken in rotation a number of times. Indicating instruments were read and recorded again at the end of the run. The automatic recorders were allowed to operate for 1 minute before and during the period in which turbine-temperature data were taken. As each turbine-temperature reading was made, the recorders were momentarily turned off; the resultant break in the film record was used to establish the time relation among the various data.

As military power could not be continuously maintained for more than 5 minutes, the time allowed for turbine temperatures to approach equilibrium was reduced to $2\frac{1}{2}$ minutes during runs at this power condition. As much turbine-temperature data as possible were obtained in the remaining time.

Because of the frequent wide variations in successive temperature readings at a given point on the turbine during the first nine flights, more frequent readings were obtained from the thermocouple at the greatest radius even though it meant getting little or no data from the other thermocouples. Accordingly, on flight 10 and thereafter data were taken on this basis.

The bleeder was installed in the duct between the compressor and the carburetor only for flight 15.

Test conditions. - Turbine temperatures were observed at an altitude of approximately 15,000 feet at various power conditions over the entire operating range of the engine. Runs were made at cruising, rated, and military powers and at intermediate power conditions. At cruising power where operating procedure permitted leaning out the fuel-air mixture, runs were made with various mixture-control settings from full rich to automatic lean; thus, a wide range of exhaust temperatures was obtained. On a single flight at rated power, three runs were made at mixture strengths covering as wide a range as feasible. Runs at three different mixture strengths were made at cruising power while a constant additional load was imposed on the turbosupercharger by bleeding air from the induction system between the turbosupercharger and the carburetor.

All data were obtained in level flight because it proved impossible to maintain a given set of conditions in climb while taking turbine-temperature data. Table I gives the conditions under which each run was made.

CALCULATIONS

Cooling-air flow. - A sea-level calibration of weight flow against velocity head in the cooling-cap inlet was used as the basis for calculating cooling-air weight flows at altitude. For equal velocity heads, it follows that

$$\frac{W_{c1}}{W_{c0}}=\sqrt{\frac{P_1T_0}{P_0T_1}}$$

where

W_c air flow through cooling cap, (lb/sec)

P total pressure in cooling-cap inlet, (lb/sq in.)

T total temperature in cooling-cap inlet, °R

Subscripts 0 and 1 denote sea-level and altitude conditions, respectively.

Substituting the values prevailing during the sea-level calibrations for P_0 and T_0 yields

$$W_{c1} = 0.51\ W_{c0}\sqrt{\frac{P_1}{T_1}}$$

Power required by compressor. - The compressor power requirements were calculated from the measured temperature rise across the compressor and the charge-air weight flow as determined from a calibration by G. L. Sanwald at the Naval Air Material center (Philadelphia) in 1940 by the following equation:

$$hp = \frac{\gamma}{\gamma-1}\ \frac{RW_e\ \Delta T_c}{550}$$

where

γ ratio of specific heats

R gas constant for normal air, (ft-lb)/(lb mass) ($^{\circ}$F)

W_e engine charge air, (lb/sec)

ΔT_c temperature rise through compressor, $^{\circ}$F

Effective exhaust-gas temperature at blade surfaces. - In the calculation of the gas temperature at the blade surfaces, the nozzle-box temperature was taken as the value measured just ahead of the inlet, the temperature drop through the nozzles was taken as 85 percent of the adiabatic drop encountered in expanding from nozzle-box pressure to the pressure of the surrounding atmosphere, and a recovery coefficient of 85 percent was assumed at the blade surfaces. It then followed that

$$T_e = T_n - 0.85\, T_n \left[1 - \left(\frac{P_1}{P_n}\right)^{\frac{\gamma-1}{\gamma}}\right] + \frac{0.85v^2}{2Jgc_p}$$

where

T_e effective gas temperature at blade surfaces, $^{\circ}$R

T_n nozzle-box inlet temperature, $^{\circ}$R

P_n nozzle-box pressure, (lb/sq in.)

v gas velocity relative to turbine blades, (ft/sec)

J mechanical equivalent of heat, (ft-lb)/Btu

g acceleration of gravity, (ft/sec^2)

c_p specific heat of exhaust gases, Btu/(lb) ($^{\circ}$F)

The laboratory development work on the turbine-thermocouple apparatus indicated that the errors introduced into the temperature measurements by the sliding contacts between brushes and slip rings were within $\pm 25^{\circ}$ F at the rated turbine speed of 21,300 rpm.

After a change in the flight conditions affecting turbine temperature, it was not feasible to allow sufficient time for the disk temperatures to reach equilibrium. Subsequent ground tests in an altitude chamber (reference 1) indicated that it would require at least 1 hour to reach equilibrium. During the ground tests, however, values within 10° F of equilibrium were reached at the inner edge of the rim ($4\frac{5}{32}$-in. radius) and middle of the rim ($4\frac{15}{32}$-in. radius) in approximately 30 and 10 minutes, respectively. Flight tests showed that 5 minutes were sufficient for the temperature at the outer edge of the rim to reach an apparently steady value. (See fig. 4.) These considerations would indicate that the 5-minute period used in flight was sufficient to allow the outer edge and the middle of the rim to reach temperatures near enough equilibrium to be representative of the conditions under which they were measured. Data from the thermocouple at the inner edge of the rim are less reliable and data at the $2\frac{1}{2}$-inch radius are of little value.

The greater number of temperature readings taken at a given point on the turbine during each run of flights 10 to 16 assured more nearly representative average temperatures. These data are therefore more reliable than the data of other flights.

The exhaust-gas temperatures measured at the nozzle-box inlet were subject to several errors. Radiation and conduction of heat away from the tip of the thermocouple sheath resulted in a hot-junction temperature below that of the surrounding gases. According to the manufacturer's tests, the quadruple-shielded thermocouple should indicate temperatures from 0° to 20° F below actual gas temperatures. No attempt was made to check these values. Recording and reading film records introduced another possible error of approximately ±2 percent or ±30° F. These errors are naturally reflected in the calculated effective gas temperatures at the turbine-blade surfaces. In addition, the errors involved in determining the pressure drops across the nozzles add another ±10° F.

The accuracies to which other values were determined are given in the following table:

Cooling-air flow, percent	±10
Engine speed, rpm	±30
Turbine speed, rpm	±250
Altitude, feet	±250
Manifold pressure, in. Hg	±0.3
Nozzle-box pressure, in. Hg	±0.3
Bleeder-duct static pressure, in. Hg	±0.1
Bleeder-duct velocity head, in. water	±0.1
Compressor-inlet pressure, in. Hg	±0.25
Compressor-outlet pressure, in. Hg	±0.4
Air temperatures, $^{\circ}$F	±6

RESULTS AND DISCUSSION

The effect of variations in exhaust-gas temperature upon turbine temperature at the outer edge of the rim ($4\frac{23}{32}$-in. radius) is shown for five flights at an altitude of approximately 15,000 feet in figure 5. At cruising power flights 8, 10, and 12 were made under the same conditions with no air flow through the bleeder duct and show a consistent, almost linear, relation between these temperatures. Average turbine temperatures during these flights fell within ±25° F of a mean curve when plotted against exhaust-gas temperatures at the blade surfaces. When the same engine power was maintained and the load on the turbosupercharger was increased approximately 50 percent (flight 15) by bleeding air from the high-pressure side of the compressor, the turbine temperatures varied with exhaust temperatures at the blade surfaces at nearly the same rate as at cruising power.

At higher engine powers, operating considerations made it impracticable to vary appreciably the fuel-air ratio and hence the exhaust-gas temperature. During flight 13 at 115-percent rated engine power, a variation of 60° F was as wide a change in exhaust-gas temperature as was feasible to obtain. In this interval the turbine temperature changed more rapidly with exhaust-gas temperature than at cruising power. Because of the narrow temperature range involved, however, this increased rate of change has little significance.

The power demand upon the turbosupercharger was increased during flight 15 by bleeding air from the compressor-outlet duct.

The turbine temperatures at the outer edge of the rim averaged 140° F above those observed at comparable exhaust-gas temperatures during flight 12 (fig. 5). Inasmuch as engine power conditions were the same on both flights, airspeed, and hence air flow through the cooling cap, were almost the same. Turbine speed during flight 15 was 8 percent higher; thus the effectiveness of the cooling air was increased due to its higher velocity relative to the turbine disk. Aside from the differences due to this increased effectiveness of cooling air, the differences in turbine temperatures were due to factors resulting from the increased turbine load; namely, the increase in mass flow of exhaust gases through the turbine and the higher relative velocity of these gases with respect to the blades.

During flight 13 at 115-percent rated engine power, the effect of the increased mass flow through the turbine was partly offset by the greater mass flow of cooling air resulting from the higher air speed. The turbosupercharger load was 25 percent above that of flight 12 and the cooling-air flow increased 15 percent. Increases of 45° to 75° F over the turbine temperatures of flight 12 were observed at comparable exhaust temperatures.

Turbine temperatures at the edge of the rim ($4\frac{23}{32}$-in. radius) for a wide variety of conditions have been plotted against the effective exhaust-gas temperatures at the blade surfaces in figure 6. Table I presents the conditions for each run. The rated-power and the cruising-power runs previously discussed have been replotted in figure 6 for comparison.

At an altitude of approximately 25,000 feet, three successful runs were made, which gave turbine temperatures of 1074°, 1069°, and 1059° F at the outer edge of the rim for rated, cruising, and an intermediate power, respectively. (See fig. 4(d), 4(e), and 4(f) and table I, flight 16.) Temperatures at the middle of the rim were obtained at that altitude on flights 14 and 16. (See table I.)

Turbine temperatures were obtained for altitudes of approximately 10,000 and 20,000 feet at approximately rated power during flights 1 and 2. (See table I.)

The temperature distribution in the gases between the cooling cap and the outer surface of the turbine is shown for four sets of conditions in figure 7. The average temperature in this region is shown in figure 8 for all flights after flight 6.

Exhaustive attempts have been made to correlate the effects of the various factors upon turbine temperatures but the data are inadequate. Corrections based upon the altitude-chamber tests of reference 1 were ineffective.

The wide temperature variations observed during flight 5 (fig. 6) may have been due to excessive temperatures of the gases between the cooling cap and the outer surface of the turbine, possibly as a result of afterburning. Because this extreme temperature variation did not occur again, no conclusions as to its cause can be drawn. A marked resemblance exists between the plot of turbine temperature against exhaust-gas temperature in figure 5 and the plot of gas temperature between the cooling cap and the rotor against exhaust-gas temperature curves in figure 8 insofar as the normal cruising-power flights are concerned. The differences in rotor temperatures during flights 10 and 12 and the differences between average gas temperatures between the cooling cap and the rotor are of the same order of magnitude. This similarity may offer an explanation of the differences in turbine temperatures during these two flights. The variations observed in the cooling-air flow and the power required by the compressor do not account for these differences.

The maximum turbine temperatures occurring in normal operation would be expected at rated altitude (25,000 ft) under cruising conditions with the leanest mixture permissible. No turbine-temperature data were obtained during the single run attempted under these conditions. A turbine temperature of 1069° F was obtained, however, during a rich-mixture run at cruising power at an altitude of 25,000 feet. If the variation of turbine temperature with exhaust-gas temperature is of the same order as observed at an altitude of 15,000 feet, the turbine temperature at $4\frac{23}{32}$-inch radius under lean-mixture cruising conditions at 25,000 feet would be approximately 1175° F.

Aircraft Engine Research Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio, July 22, 1946.

REFERENCE

1. Hartwig, Frederick J., Jr.: Comparative Effectiveness of a Convection-Type and a Radiation-Type Cooling Cap on a Turbo-supercharger. NACA TN No. 1082, 1946.

TABLE I - FLIGHT CONDITIONS

Flight	Altitude (ft)	Engine speed (rpm)	Manifold pressure (in. Hg)	Turbine temperature at $4\frac{23}{32}$ inch radius (°F)	Turbine temperature at $4\frac{15}{32}$ inch radius (°F)	Turbine temperature at $4\frac{5}{32}$ inch radius (°F)	Turbine temperature at $2\frac{1}{2}$ inch radius (°F)	Exhaust temperature at nozzle-box inlet (°F)	Exhaust temperature leaving nozzles (°F)	Effective exhaust temperature at blade surfaces (°F)	Airspeed (mph)	Air flow through cooling cap (lb/sec)	Free air temperature (°F)	Nozzle-box pressure (in. Hg)	Estimated compressor power (hp)	Turbine speed (rpm)	Supercharger-inlet pressure total (in. Hg)	Supercharger-outlet pressure static (in. Hg)	Temperature rise through compressor (°F)	Average gas temperature between cap and rotor (°F)
[a]1	9,200																			
	9,600	2580	31.0	---	936	785	---	1635	1544	1567	222	0.0715	21	25.2	---	9,600	22.6	26.0	---	---
	14,500																			
	13,500	2530	30.5	---	920	835	---	1650	1514	1550	238	.0665	8	23.5	---	13,000	19.0	25.5	---	---
2	19,800	2600	30.2	926	867	800	405	1640	1423	1482	268.5	0.0660	-10	22.5	58.5	17,000	15.0	26.5	115	---
[a]3	14,800																			
	14,400	2200	24.3	910	853	753	---	1473	1392	1415	233	0.0612	37	20.5	14.5	9,500	18.5	22.0	44	---
	14,800	2320	27.7	931	832	731	---	1535	1414	1447	253	.0660	37	22.5	25.6	12,500	18.4	25.2	61	---
	14,800																			
	14,600	2520	29.7	899	806	724	---	1605	1467	1505	259	.0695	37	23.0	34.2	13,750±750	18.6	26.0	71	---
4	14,800	2200	25.6	973	852	813	481	1577	1480	1505	224	0.0621	15	21.2	20.2	11,500	18.0	23.5	55.5	---
	14,700	2580	31.8	994	853	794	511	1624	1481	1522	251.5	.0722	15	24.5	45.0	15,000	18.5	28.0	85.5	---
	14,700	2900	37.2	977	---	888	523	1697	1482	1557	270	.0786	15	28.0	---	17,000	18.5	31.0	---	---
5	14,600	2250	23.6	949	885	893	430	1552	1478	1497	220	0.0629	10	20.0	13.7	9,500	18.0	21.5	47	---
	14,600	2440	26.0	1048	901	925	447	1585	1476	1505	237	.0671	10	21.5	21.0	11,000	18.0	23.2	57.5	---
	14,600	2415	28.9	1101	877	946	433	1605	1478	1511	247	.0705	10	23.0	30.5	13,500±500	18.2	25.8	70	---
	14,700	2600	31.6	1041	898	920	433	1640	1477	1520	258	.0733	10	24.5	43.3	14,750±250	18.4	28.0	87	---
6	14,600	2300	24.0	---	949	---	520	1549	1463	1490	229	0.0629	16	20.5	12.2	8,900	18.0	19.5	35	---
	14,600	2430	27.9	---	936	---	511	1608	1490	1527	241	.0701	16	22.0	24.9	10,500	18.0	22.5	56	---
	14,600	2570	31.5	---	980	---	562	1652	1505	1552	253	.0704	16	23.5	36.4	11,600	18.0	24.5	71	---
7	14,600	2180	26.0	917	840	800	424	1428	1331	1357	236	0.0690	30	21.5	21.6	11,400	17.6	23.5	59	402
	14,600	2480	28.6	920	848	727	466	1493	1364	1397	253	.0762	30	23.0	34.2	13,300	18.0	25.3	74	407
	14,600	2600	32.0	984	882	953	472	1550	1394	1435	262	.0798	30	24.4	46.3	14,500	18.5	27.0	89	425
8	14,600	2275	26.4	995	894	---	523	1655	1542	1572	239	0.0700	37	21.8	29.1	12,200	17.8	24.0	73	459
	14,600	2275	26.9	950	871	---	494	1647	1531	1562	243	.0709	37	22.0	31.9	12,300	17.7	24.5	80	447
	14,600	2275	27.1	1085	980	---	495	1763	1643	1675	242	.0703	37	22.0	31.0	12,400	18.0	24.6	78	508
9	14,600	2400	26.2	945	---	---	---	1610	1501	1530	244.5	0.0719	30	21.7	23.7	12,000	17.7	23.5	60	460
	14,600	2600	30.5	960	---	---	---	1670	1518	1558	260	.0783	30	23.8	44.9	14,000	18.0	26.3	89	483
	14,600	2900	37.6	930	---	---	---	1745	1522	1581	283	.0850	30	28.0	77.8	17,500	18.3	31.1	124	486
10	14,600	2280	26.6	872	740	736	---	1470	1359	1388	225	0.0658	37	22.1	28.0	12,400	17.5	24.2	70	382
	14,600	2280	26.9	950	---	---	---	1625	1508	1539	228	.0656	37	---	---	---	---	---	70	443
	14,600	2280	27.1	1035	850	837	---	1733	1614	1646	228	.0664	36	22.1	27.3	12,300	18.0	24.3	70	497
11	14,600	2280	26.7	840	---	---	---	1455	1345	1373	237	0.0688	36	22.2	28.9	12,450	17.6	24.4	73	345
	14,600	---	---	---	---	---	---	---	---	---	240	---	36	22.3	---	12,600	17.7	24.9	---	---
12	14,600	2280	26.6	890	658	---	---	1464	1356	1384	237	0.0685	36	22.0	23.8	12,300	17.5	24.2	62	416
	14,600	2280	26.9	955	727	---	---	1578	1466	1495	234.5	.0685	36	---	25.4	---	---	---	65	457
	14,600	2280	27.1	1055	792	---	---	1702	1585	1616	237	.0685	36	22.0	26.1	12,400	17.6	24.6	66	504
13	14,600	2580	33.3	1006	---	---	---	1624	1451	1494	266	0.0784	37	25.2	58.4	16,000	17.6	28.5	107	440
	14,600	2580	33.1	985	---	---	---	1602	1429	1472	264	.0784	37	---	57.4	---	---	---	106	431
	14,600	2580	33.4	1060	---	---	---	1662	1482	1528	270	.0787	37	25.6	53.2	16,000	17.8	29.1	112	437
14	25,100	2260	26.1	---	909	---	---	1474	1236	1324	258	0.0508	0	21	58.6	19,400	10.5	24.0	149	454
	24,900	2320	26.7	---	1005	---	---	1730	1472	1564	254	.0508	0	21	60.3	19,300	10.8	24.0	148	567
15	14,600	2280	26.3	1030	---	---	---	1440	1299	1341	230.5	0.0652	42	24.2	40.3	13,400	17.0	23.4	77	402
	14,600	2280	26.8	1160	---	---	---	1680	1530	1575	234	.0672	42	24.2	42.4	---	---	23.9	79	517
	14,600	2280	26.5	1055	---	---	---	1570	1423	1468	232	.0665	42	24.2	39.7	13,350	17.4	23.8	75	446
[a]16	24,000																			
	24,500	2600	31.0	1074	951	---	---	1635	1337	1422	277	0.0610	2	23.8	80.3	20,000	12.32	26.5	159	527
	24,500	2600	28.0	1059	---	---	---	1647	1354	1444	267	.0575	2	23.8	---	19,000	12.32	23.4	140	534
	24,500	2280	27.0	1069	940	---	---	1647	1396	1469	270	.0577	2	21.2	57.2	18,200	12.26	24.3	134	521
[b]17	24,000	2600	31	---	---	---	---	1660	1375	---	220	0.0485	---	---	---	---	---	---	---	526

[a]Altitude varied within limits shown.

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[b]The values shown for flight 17 were obtained during climb at an altitude of approximately 24,000 feet.

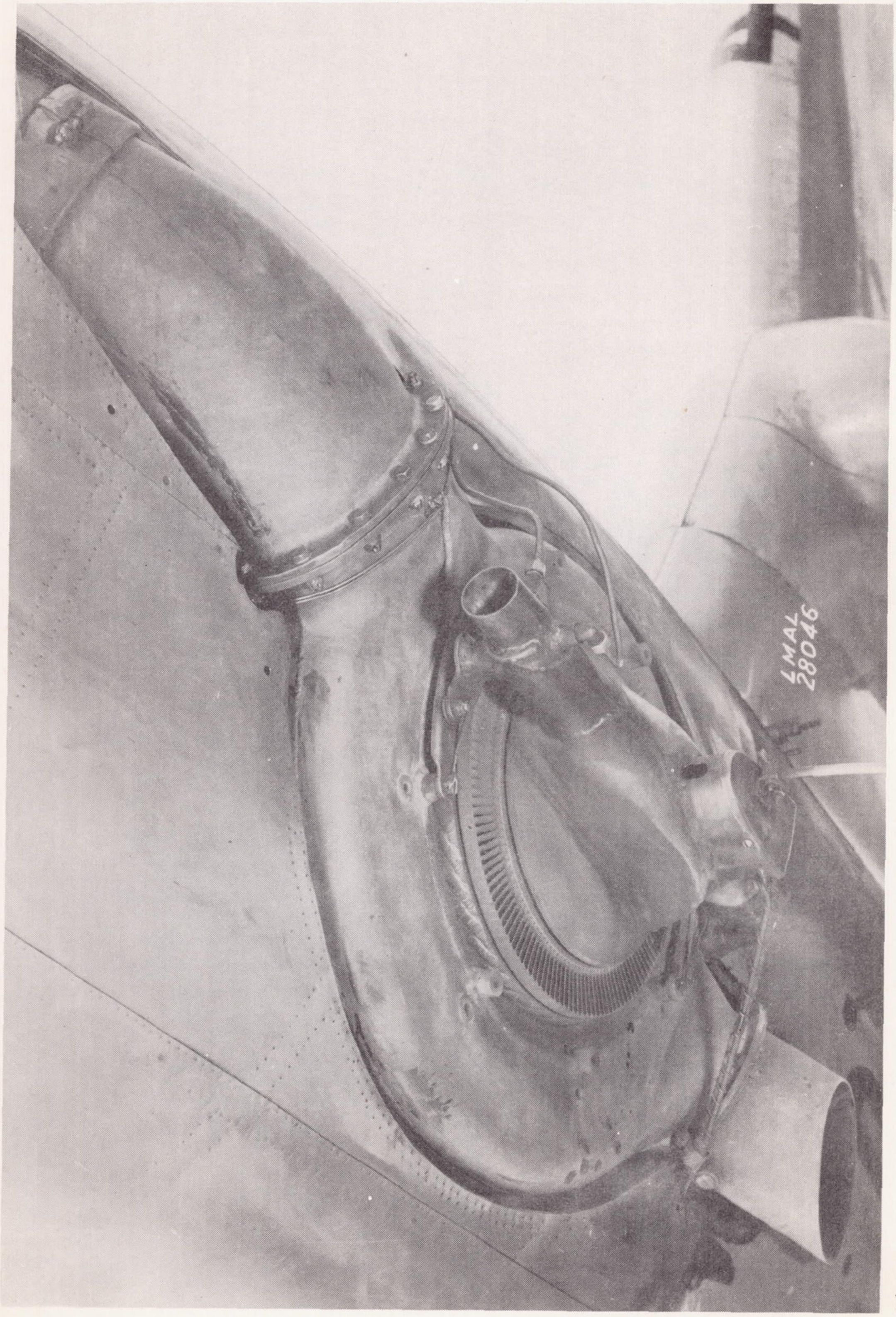


Figure 1. - Turbosupercharger with cooling cap in position ready for flight.

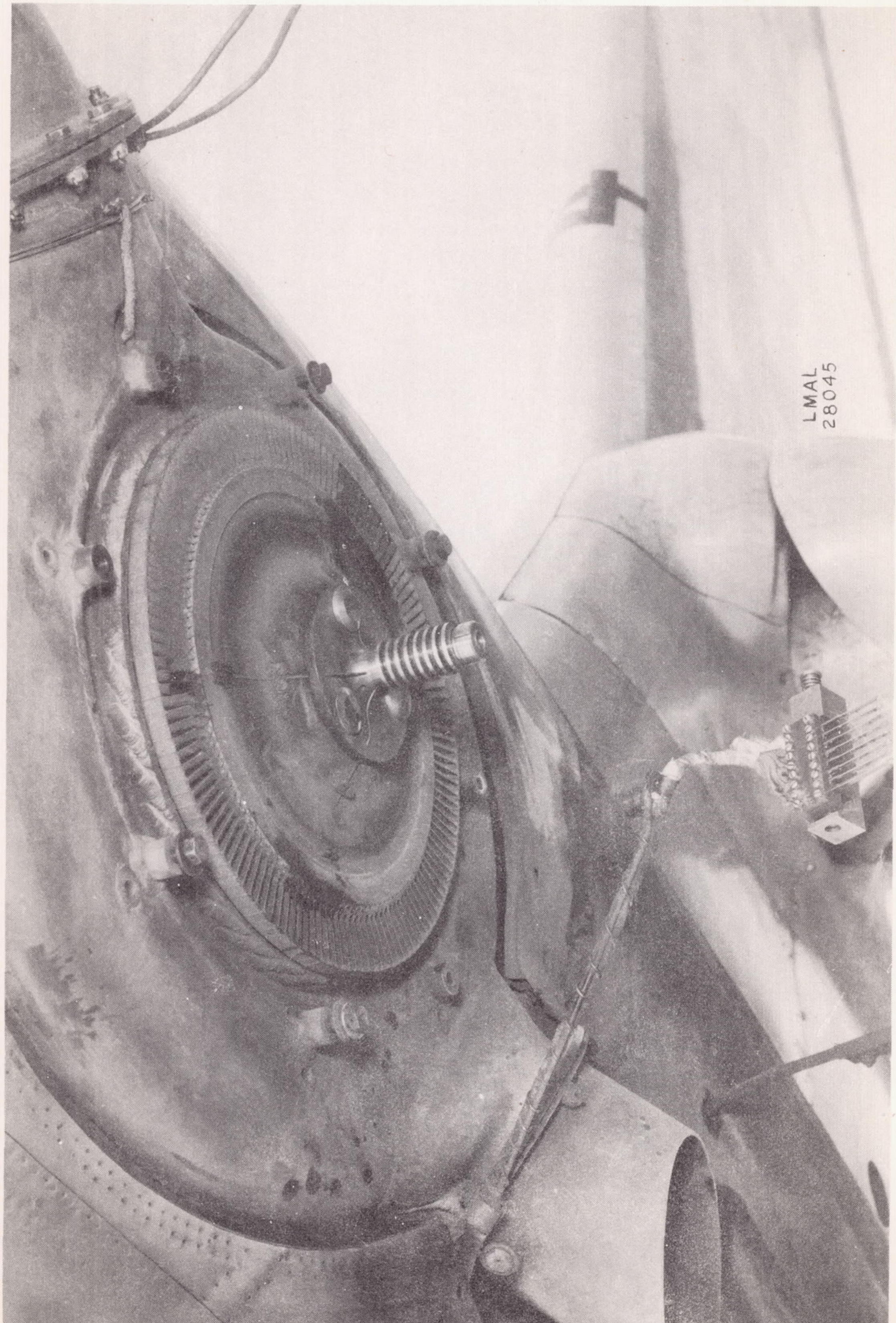


Figure 2. - Turbine thermocouples, slip rings, and brush assembly.

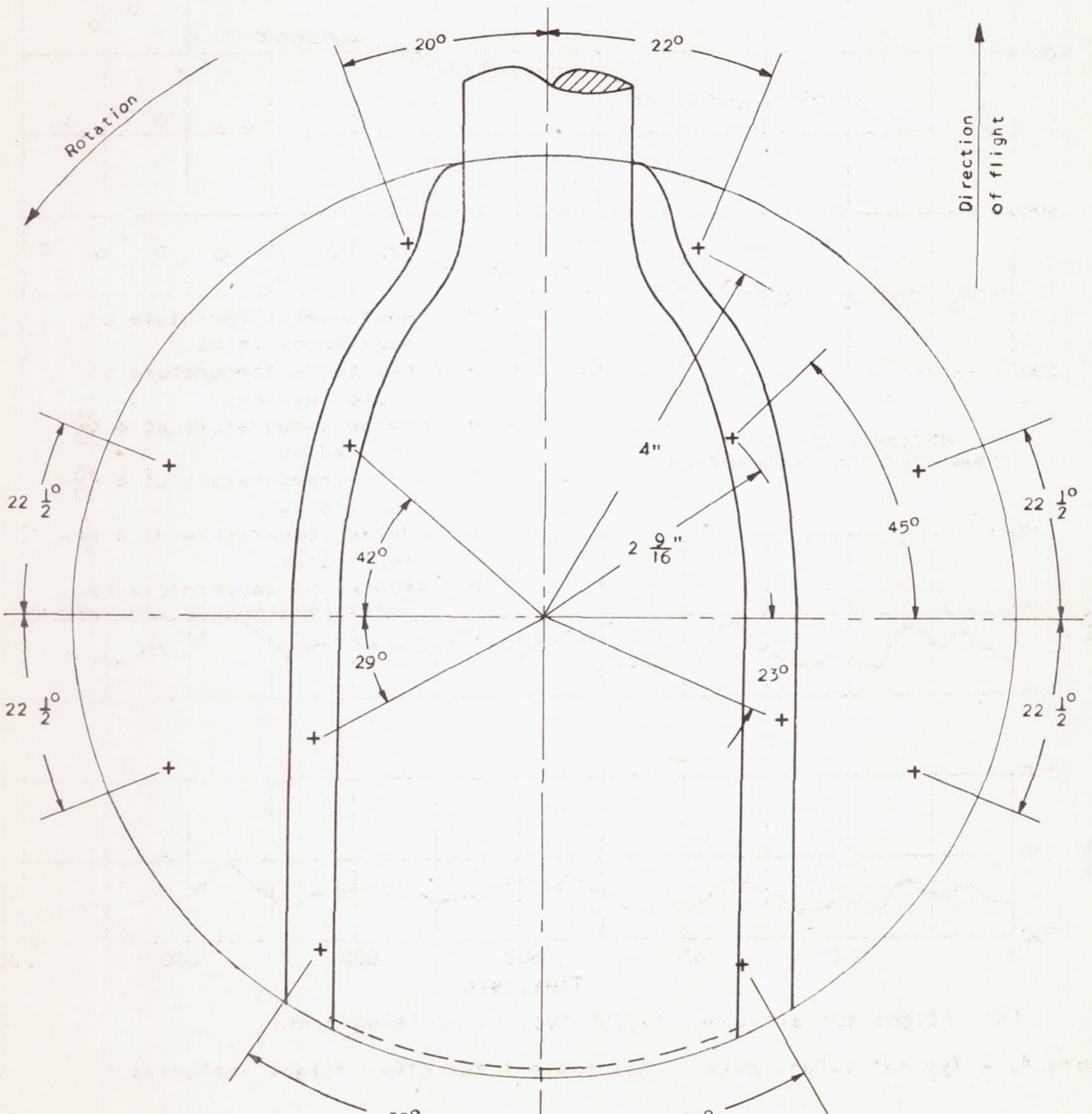
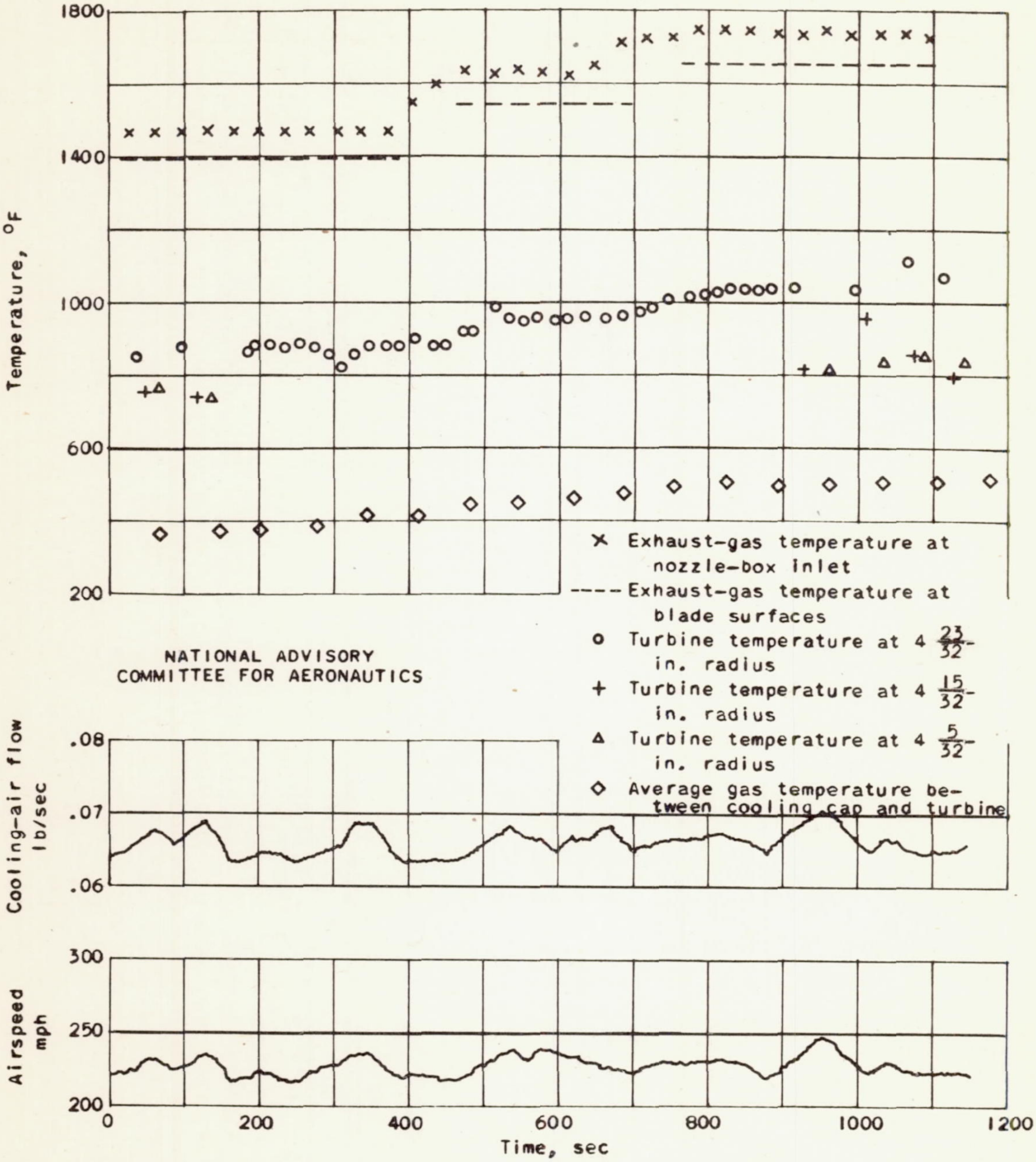
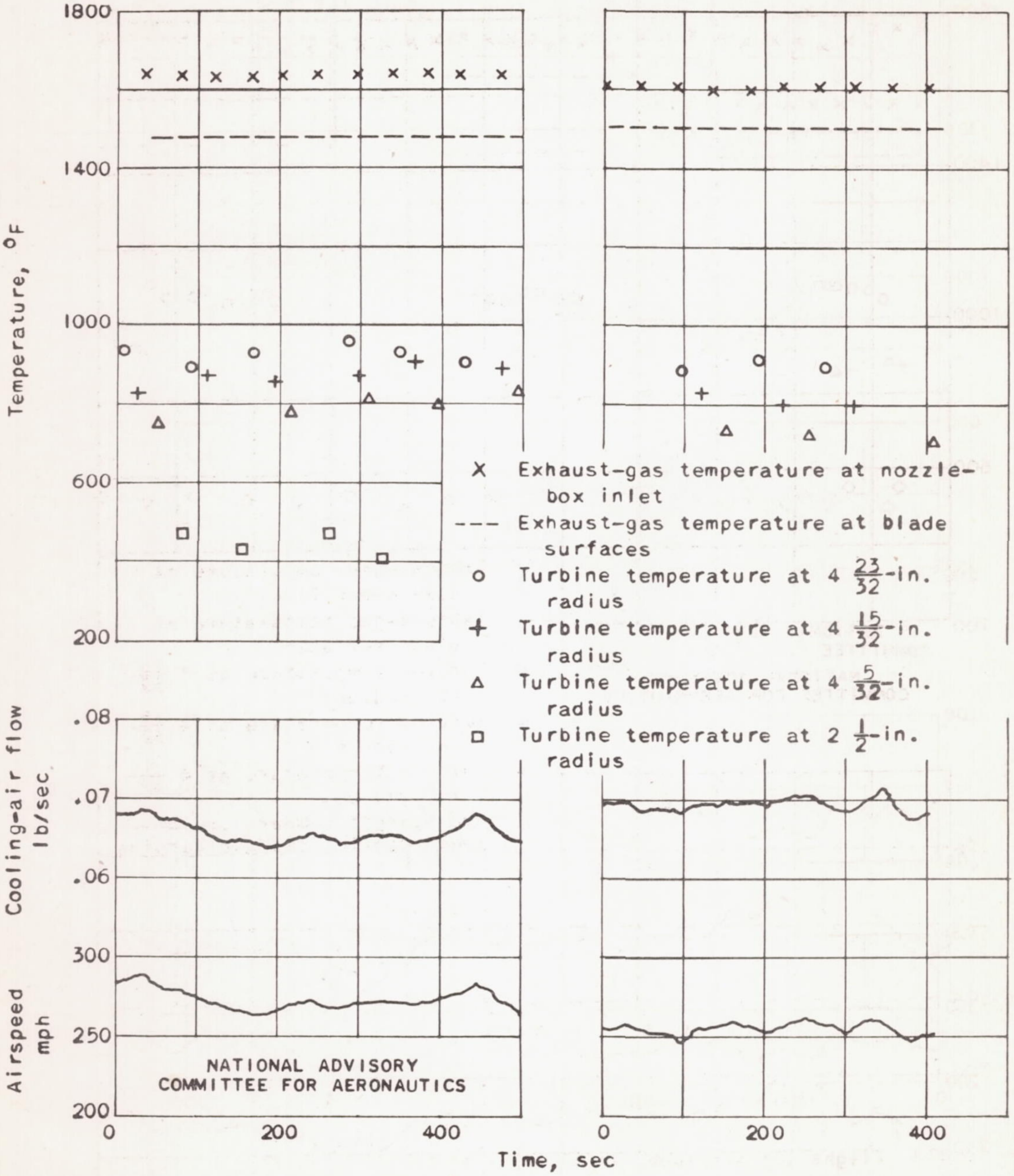


Figure 3. - Location of thermocouples between cooling cap and turbine rotor.



(a) Flight 10; altitude, 14,600 feet at cruising power.

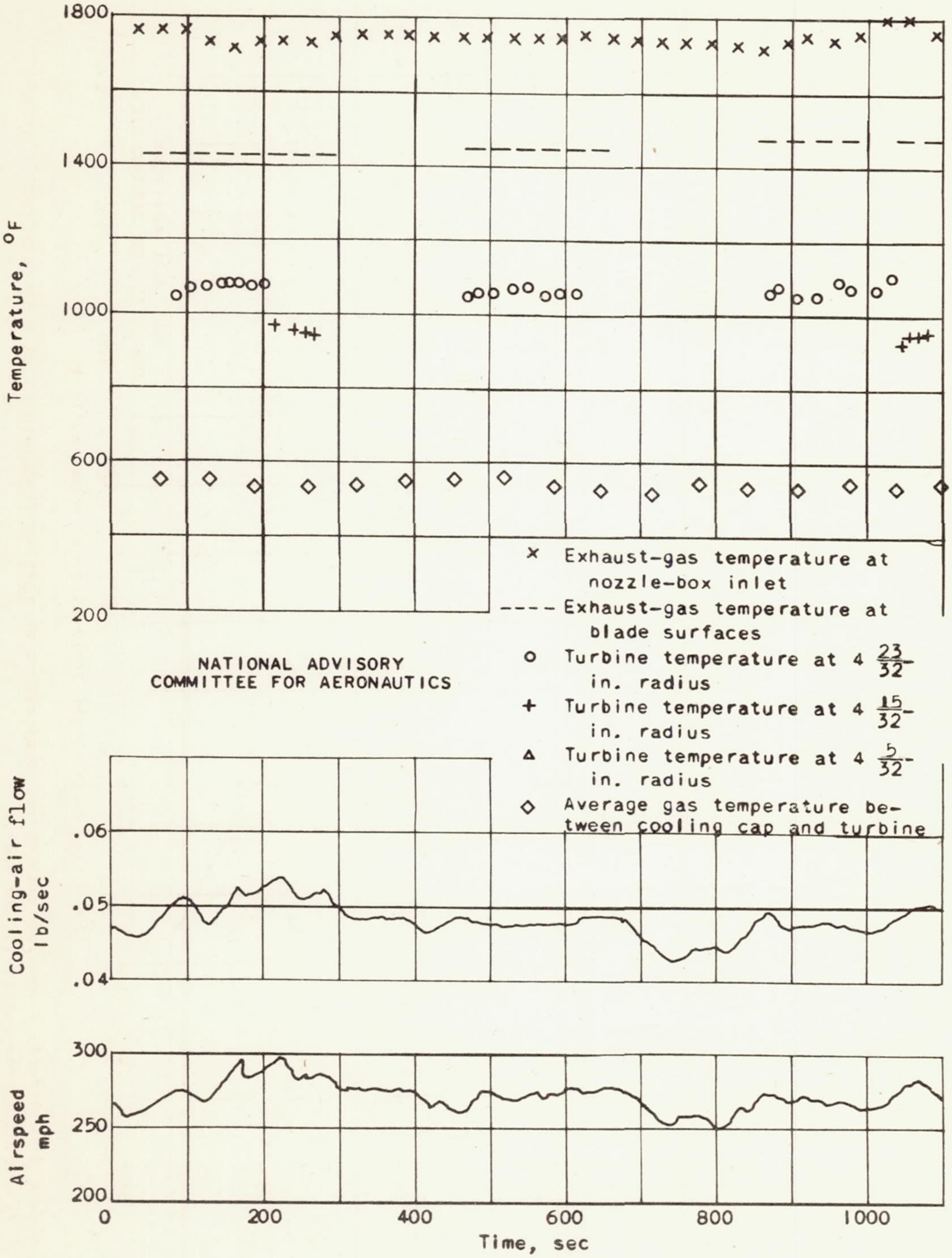
Figure 4. - Typical flight data. (See table I for other flight variables.)



(b) Flight 2; altitude, 19,800 feet at approximate rated power.

(c) Flight 3; altitude, 14,700 feet at approximated rated power.

Figure 4. - Continued. Typical flight data. (See table I for other flight variables.)



(d) Flight 16; altitude, 24,300 feet at rated power.

(e) Flight 16; altitude, 24,300 feet at intermediate power.

(f) Flight 16; altitude, 24,300 feet at cruising power.

Figure 4. – Concluded. Typical flight data. (See table I for other flight variables.)

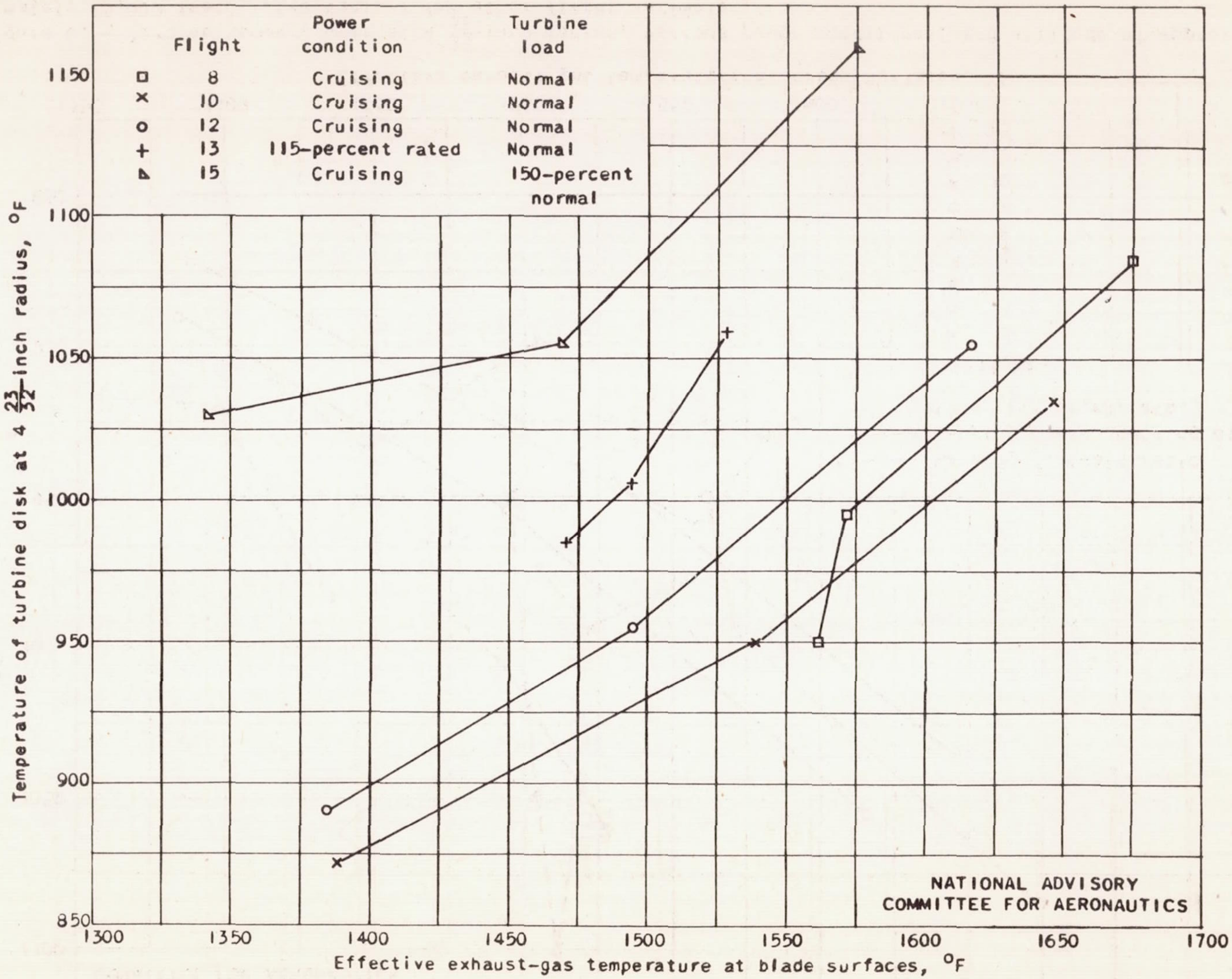


Figure 5. - Effect of varying exhaust-gas temperature on turbine temperatures at 4 $\frac{23}{32}$-inch radius at altitude of approximately 15,000 feet.

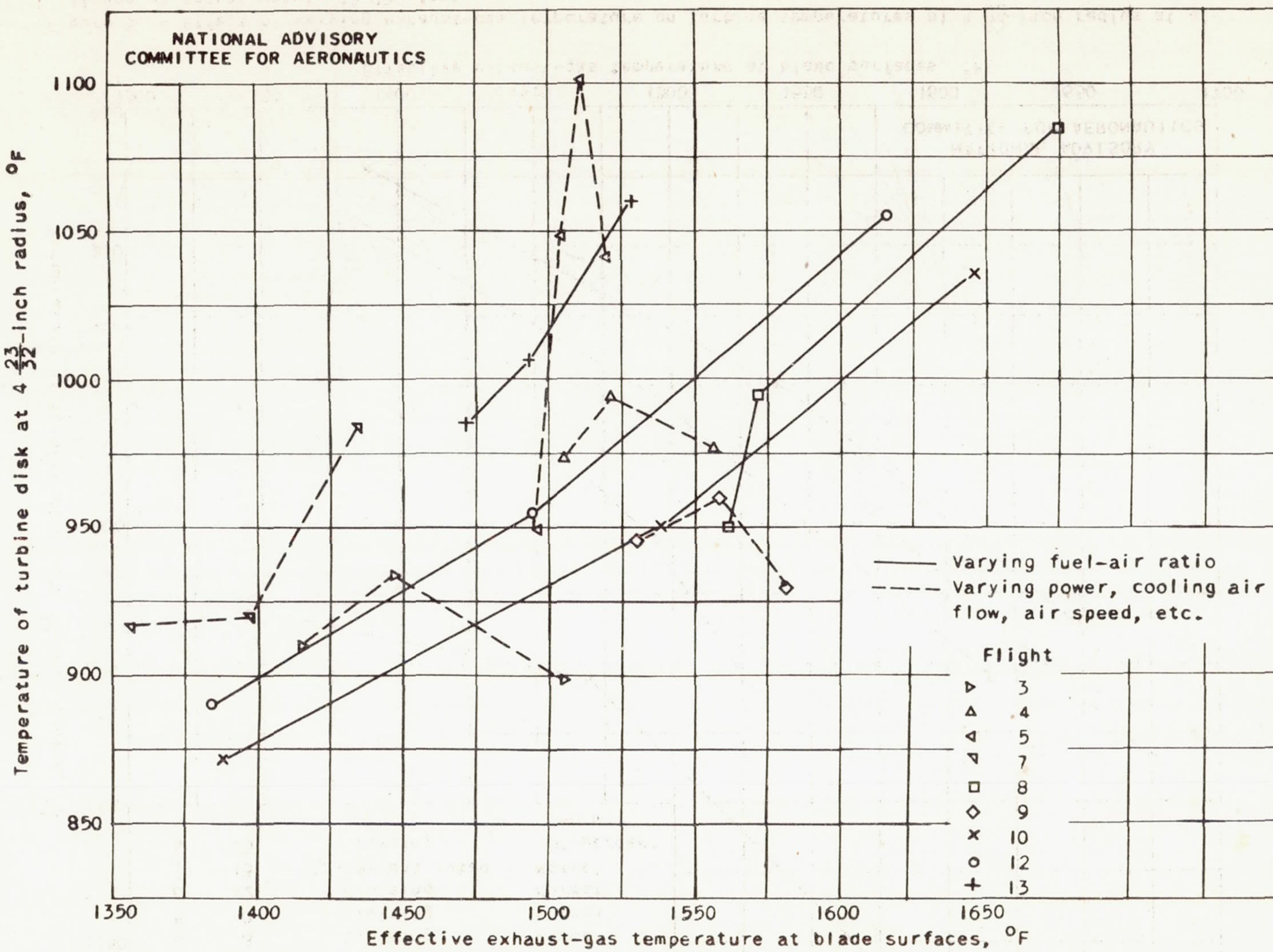
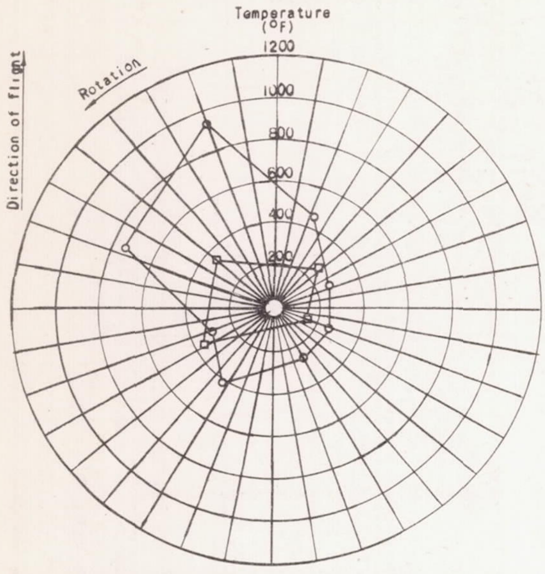
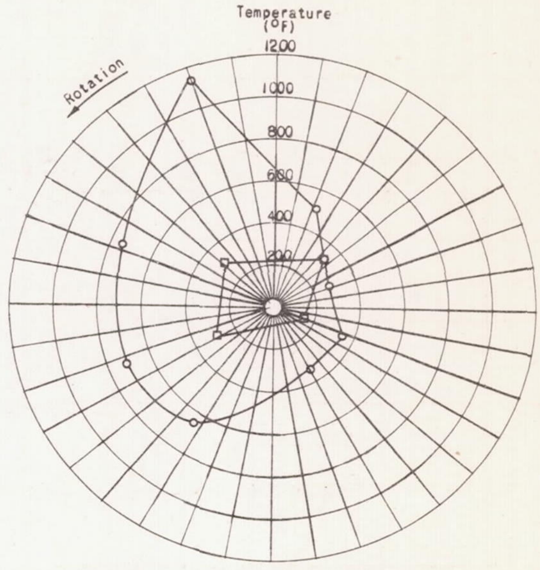


Figure 6. - Turbine temperatures at 4 $\frac{23}{32}$-inch radius, various power conditions, and altitude of approximately 15,000 feet. (See table I for other flight variables.)

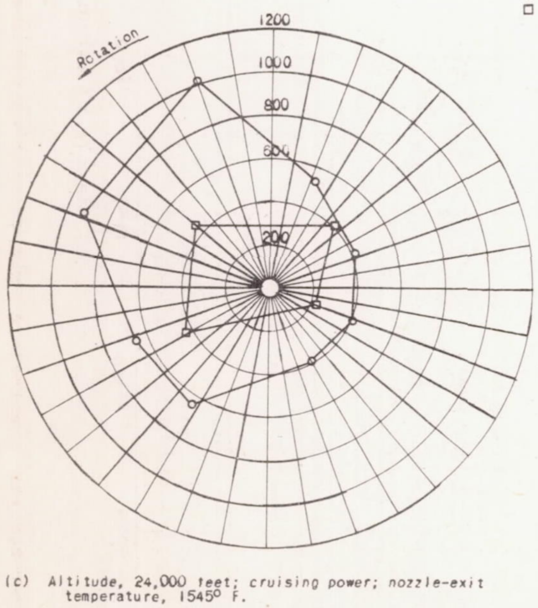


(a) Altitude, 14,600 feet; cruising power; nozzle-exit temperature, 1360° F.

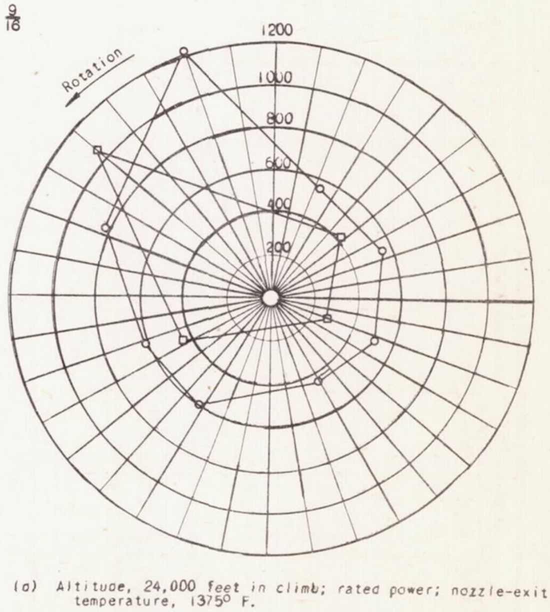


(b) Altitude, 14,600 feet; military power; nozzle-exit temperature, 1520° F.

Radius (in.)	
○	4
□	2 $\frac{9}{16}$



(c) Altitude, 24,000 feet; cruising power; nozzle-exit temperature, 1545° F.



(d) Altitude, 24,000 feet in climb; rated power; nozzle-exit temperature, 1375° F.

Figure 7. - Temperature distribution in the gases between cooling cap and turbine rotor.

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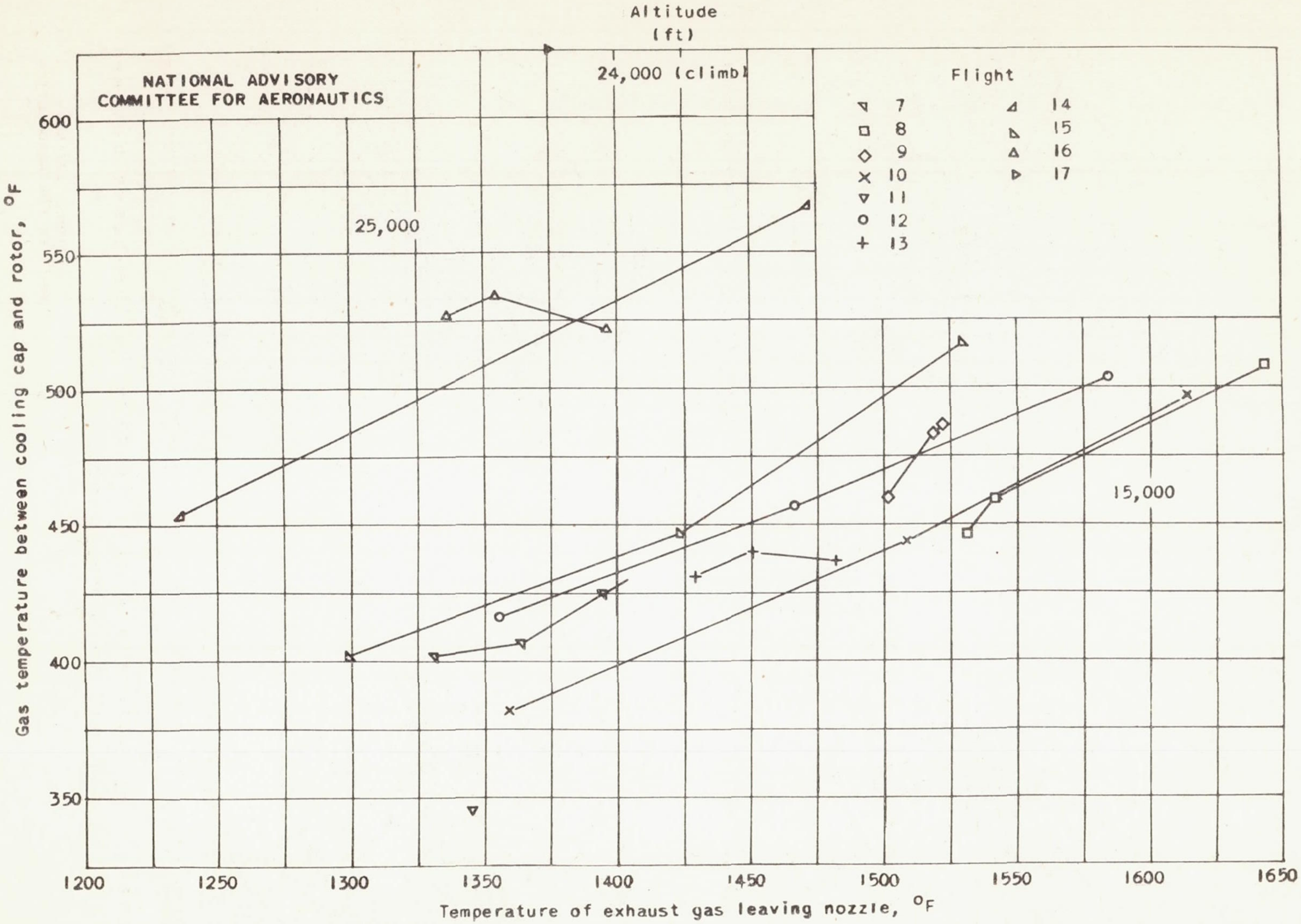

Figure 8. - Average gas temperature between cooling cap and turbine rotor.