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GAS-TURBINE-ENGINE PERFORMANCE WHEN HEAT FROM LIQUID-COOLED TURBINES IS REJECTED AHEAD OF, WITHIN, OR BEHIND MAIN COMPRESSOR

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RESEARCH MEMORANDUM

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SUMMARY

Liquid-cooled turbine-engine performance is substantially affected by the location where heat from the coolant is rejected. Methods, advantages, and disadvantages of locating rotating heat exchangers ahead of, within, and behind the main engine compressor are therefore discussed. For the best engine performance, heat rejection should occur at the compressor discharge. Although performance would be poorer, heat rejection at the compressor inlet would permit cooled-engine operation at very high flight speeds where compressor discharge temperatures are very high. This location would also permit a system with adequate coolant pumping characteristics that appears practical with respect to fabrication and operation.

From this analysis it appears that for turbojet engines: (1) Higher flight Mach numbers are possible using liquid-cooling with heat rejection in a heat exchanger ahead of the compressor than using air-cooling with unrefrigerated compressor bleed air. (2) When it is possible to use a liquid-cooling system in which heat is rejected at the compressor exit, the engine performance will be superior to that obtained with air-cooling. (3) With heat rejected from liquid-cooling at locations other than the compressor exit, air-cooled engine performance will probably be superior until a flight Mach number is reached at which some device is required for cooling the air after it is bled from the compressor. For turboprop engines it appears that cooling of small turbine blades may be more successful with liquids than with air, but the heat from the liquid coolant will probably have to be rejected at the compressor exit in order to ensure engine performance that is superior to that with air-cooling.

INTRODUCTION

In order to evaluate the relative merits of various types of turbine-cooling systems, it is necessary to have a knowledge of the effect of cooling on engine performance. The primary effects of liquid-cooling are the removal of energy from the gases by heat transfer to the turbine and rejection of this heat into the engine cycle. This report shows the effect on engine performance of rejecting heat from liquid-cooled turbines at three different locations within the engines.

Most work that has been conducted on liquid-cooled turbines in the past, such as that reported in references 1 and 2, considered water as the coolant, and part or all the heat from the water was rejected in a stationary heat exchanger. Water was a logical coolant because of its excellent heat-transfer characteristics. In addition, rather complete knowledge of its fluid properties made water ideal as a coolant for studies of forced and free convection in liquid-cooled turbines. Water as a turbine coolant, however, has one very serious disadvantage; the boiling point is so low that, unless the entire coolant system is under very high pressure, the turbine is overcooled and the heat-rejection rates or blade temperature gradients may be excessive. A further disadvantage occurs at high flight speeds, because the ram-air temperature exceeds the boiling temperature of water at normal pressures and heat rejection in an air heat exchanger may become impossible. The ram-air temperature reaches 212° F at a flight Mach number of about 1.2 at standard sea-level conditions and about 1.9 in the stratosphere.

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Pressurization of the entire coolant system that utilizes a stationary heat exchanger offers only partial relief. In order that the water reach a temperature that does not result in overcooling of the turbine, pressurization of approximately 3000 pounds per square inch is required. Seals for transferring coolant between rotating and stationary parts of the engine have not been developed to operate satisfactorily at such high pressures. A natural solution, then, would be to utilize a coolant such as a liquid metal, metal salt, or metal hydroxide that has a boiling temperature in excess of 1000° F at normal pressures. These coolants oxidize when in contact with air, so that seals between rotating and stationary parts would still be a problem. The task of making such seals absolutely airtight (air leakage should not be more than a few cc per year) is prohibitive.

A solution to the problems involved with excessive heat-rejection rates and flight at high speeds would be to utilize a rotating heat exchanger connected to the turbine rotor, so that seals between rotating and stationary engine parts would be eliminated. This arrangement, as suggested in reference 3, would permit use of water at supercritical pressures and temperatures and also the use of liquid metals, metal salts, or metal hydroxides as the coolant.

Air-cooling of turbines becomes more difficult as flight speed increases, because the temperature of the cooling air that is bled from the compressor becomes so high that very large amounts of cooling air may be required. At flight Mach numbers of the order of 2.5 or higher some method of refrigerating the cooling air would probably be necessary. An alternate cooling system could possibly use a liquid coolant with a coolant temperature of the order of 1000° F and reject the heat to air at approximately ram temperature. This coolant system should operate satisfactorily at flight Mach numbers up to at least 3. An advantage of liquid-cooling over air-cooling lies in the high heat capacity per unit volume of liquid coolant. This characteristic permits cooling of small turbine blades, such as can be encountered on turboprop engines or high-pressure-ratio turbojet engines, much easier than is possible with air-cooling. It appears, therefore, that further study of various types of liquid-cooled systems is warranted.

In the present study of liquid-cooling systems, rotating heat exchangers are considered at locations ahead of, behind, and within the main compressor. The purpose of this report is (1) to discuss the relative advantages and disadvantages of rotating heat exchangers at the three locations, (2) to show the effects of heat rejection on the performance of a liquid-cooled turbine engine when rejecting heat at the three different heat-exchanger locations, and (3) to compare the performance of air- and liquid-cooled engines. For comparison, the performance of a liquid-cooled engine with heat rejection into a sink outside the engine is also shown.

The results are presented from a thermodynamic study; actual design studies of the various systems have not been made. Some of the systems presented may not be practical for all applications. It is believed, however, that there is some merit in each system. Results are presented for afterburning and nonafterburning engines for a range of flight Mach numbers from 0.8 to 3.0, flight altitudes of sea level and 50,000 to 80,000 feet, sea-level static compressor pressure ratios from 4 to 12, and turbine-inlet temperatures of 2460° and 2800° R using heat-rejection rates from reference 4. One-spool turbojet engines with both one- and two-stage turbines and a particular multistage turboprop engine are considered.

LIQUID-COOLING HEAT-REJECTION METHODS

The use of rotating heat exchangers to eliminate seals in the coolant system between rotating and stationary parts almost dictates that the heat from the coolant be rejected to the compressor air. It appears that it may be feasible to reject this heat either ahead of, within, or behind the main compressor. Advantages and disadvantages of heat rejection at each location are discussed in this section. Consideration is given to rejecting the entire heat load from the turbine in each of the previously mentioned locations. Although in actual application it may be desirable to reject heat at two or more of the locations simultaneously, this condition is not considered herein. The effect of such operation can probably be inferred from the results presented.

In practically any liquid-cooling system there is a possibility that the coolant will freeze. This is particularly true of coolants suitable for operation at high temperatures. Water used at supercritical temperatures cannot tolerate any additives such as antifreeze. Additives alter the desirable heat-transfer characteristics and may make the water react with the metal surfaces at high temperatures. The liquid metals, metal salts, and metal hydroxides that have been considered for coolants have freezing points varying from about 15° to 600° F. With any of the proposed systems it is advantageous to have a coolant that contracts upon freezing. Some very promising coolants such as NaK (mixture of sodium and potassium) have this characteristic. When freezing occurs within a system, unbalance may occur in the rotating parts until the system is thawed. This problem could occur in any liquid-cooled system; it is not unique to the systems proposed herein. It may be possible by the proper design to utilize pressurizing devices within the system that would prohibit drainage to low parts of the system, so that unbalance with the coolant frozen could be eliminated. 4007

Rejection of heat from the first stages of a multistage turbine to the last stages was studied briefly. A few calculations indicated that gas temperatures at the last stages were too high and the surface area of the stages too small to permit heat rejection in this manner.

Heat Rejection within Main Compressor

The main engine axial-flow compressor offers a logical location for rejection of heat from the turbine because of the relatively large surface area that is available on the compressor blades. Heat from the turbine rotors could be rejected in the compressor rotor blades, and heat from the turbine stator could be rejected in the compressor stator blades. Such a system is illustrated schematically in figure 1(a).

Advantages. -

(1) Less alteration of the external appearance of the engine would be required than for either of the other two systems. The compressor could probably be of approximately the same geometry, although some increase in compressor length might be required because of added stages resulting from lower permissible blade loading, as will be discussed later.

(2) It is not expected that additional pressure losses within the engine would result from a properly designed compressor used as a heat exchanger. In other words, it is assumed that compressor efficiency would not be affected adversely. As stated in the previous advantage, however, the compressor may have to differ from a conventional compressor.

Disadvantages. -

(1) Heating of the boundary layer around the compressor blade may result in decreased boundary-layer stability, and the permissible blade loading would be smaller. As a result, a larger number of compressor stages may be required for a specified compressor pressure ratio.

(2) The stress levels in compressor blades, particularly the front stages, may be too high to permit increasing their temperature to 1000°
4007 or 1100° F by circulating high-temperature coolant through them. The use of lower coolant temperatures would defeat the purpose of the proposed system.

(3) The work required to compress air to a specified pressure is a direct function of the absolute air temperature. Therefore, rejection of heat to the compressor air results in increased compressor power requirements.

(4) Pumping the coolant in rotating heat exchangers depends on natural-convection forces within the coolant passages. These forces in turn are a function of coolant density change due to temperature changes and differences in radius of rotation between the heated and cooled portions of the circuit. For some designs where the compressor and turbine diameters are approximately equal, the natural-convection pumping may be inadequate, particularly in the rear compressor stages.

(5) Limited calculations indicate that for high heat-rejection rates the surface area within the compressor may be marginal, particularly in regard to stator cooling. This area, of course, is a function of the compressor design. Conservative designs with more compressor stages would be better with respect to area than advanced designs such as a transonic compressor.

(6) Under subfreezing conditions at the compressor inlet some difficulty may be encountered in thawing the coolant system. A possible thawing method might be hot-gas bleed to the compressor inlet. As mentioned previously, freezing could be encountered with any of the coolants suitable for operation at high temperatures.

Heat Rejection ahead of Main Compressor

Blade stress is proportional to the square of the blade tip speed. The stresses in the rotating heat exchanger could be very greatly reduced, therefore, by decreasing the heat-exchanger diameter. Thus, a small auxiliary compressor might be added ahead of the main compressor to serve as the heat exchanger (shown schematically in fig. 1(b)). This compressor would reject heat to only part of the main compressor air, and the pressure rise in the compressor would have to be only high enough to overcome frictional losses. The compressor blade design and compressor length could be tailored to provide adequate heat-transfer surface area for the heat-rejection rates required. Limited studies indicate that this auxiliary compressor could be considerably smaller in diameter and length than the main compressor. Also shown in figure 1(b) is a stationary heat exchanger at the compressor inlet for rejecting heat from the turbine stator blades.

Advantages. -

(1) The stress level in the heat exchanger is very much lower than in the compressor blades, so there would be less danger due to heating of the blades.

(2) The heat exchanger can be readily thawed by frictional heating if the exit guide vanes of the auxiliary compressor are closed.

(3) Coolant pumping characteristics of the rotating members are superior to those of the system in which heat is rejected in the main compressor blades because of the smaller diameter.

(4) The heat-transfer surface area can be controlled by the length of the heat exchanger, and heat-transfer coefficients can be controlled by choice of blade sizes within the compressor. Such a compressor might be made of many rows of small blades spaced close together without stator blades. There is also the possibility, however, that stator blades could be used, and they could serve as the stationary heat exchanger for rejecting heat from the turbine stator blades.

(5) The average air temperature to which the heat would be rejected would be lower than for any of the other systems considered, because there would be no heat of compression added. As a result, higher flight Mach numbers would be possible and heat-exchanger surface area could be smaller than for the other systems.

Disadvantages. -

(1) The system results in increased engine length and weight, and it may complicate the front compressor bearing arrangement.

(2) The discharge of heated air at the inner diameter of the main compressor inlet may have a deleterious effect on compressor performance due to flow distortion. The seriousness of this effect requires further investigation.

(3) Under some flight attitudes, flow distributions from the inlet diffuser may make it difficult to obtain an adequate quantity of air flow through the auxiliary-compressor heat exchanger.

(4) A small amount of added turbine work is required to rotate this heat exchanger and to provide sufficient pressure rise to overcome frictional losses.

(5) Rejection of heat ahead of the main compressor results in in-
4007 creases in compressor power requirements that are higher than those for heat rejection within the main compressor.

Heat Rejection behind Main Compressor

Heat rejection within or ahead of the main compressor results in increased turbine work requirements because of the heat addition to the compressor air. This objection could be overcome if the heat could be rejected at the compressor discharge in a manner similar to that illustrated in figure 1(c). Although the stresses in this type of rotating heat exchanger would probably be higher than for the small-diameter heat exchanger placed ahead of the main compressor, the stresses would be smaller than those of the turbine blades or the early stages of the compressor because of a higher hub-tip radius ratio (shorter blades). The essential idea of this type of system is to extend the compressor to serve as a heat exchanger, but the design would be one that favored heat transfer and would not necessarily result in a pressure rise across the unit.

Advantages. -

(1) This is a regenerative type of system where heat removed from the cycle is replaced at the most advantageous spot, just ahead of the primary burners. This arrangement results in the smallest possible performance loss.

(2) Under all flight conditions the temperature of the heat exchanger would be high enough to thaw coolants such as water or NaK.

(3) Heat-transfer surface area and heat-transfer coefficients can be controlled by the length and the design of the heat exchanger.

Disadvantages. -

(1) The heat is rejected to the compressor air after all the heat of compression has been added. This higher temperature would impose a lower flight Mach number limitation on this system than for either of the other systems discussed.

(2) The system results in increased engine length and weight. For some applications the required length of the heat exchanger may not be feasible.

(3) For some designs the coolant pumping forces due to natural convection may be inadequate. If the compressor and turbine are approximately the same diameter, inadequate natural-convection pumping is almost a certainty.

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Coolant Pumping

As mentioned in the discussion of heat rejection within the main compressor, the pumping of the coolant in the rotating system would depend on the natural-convection forces within the coolant. These forces can become very high with the proper design. The pumping forces are higher for water than for most other coolants; but, if there is an adequate change of radius between the heated and cooled portions of the coolant circuit, other coolants such as NaK should be satisfactory. If necessary, the natural-convection pumping could be augmented by other means. Mechanical pumps would probably cause some rather difficult engineering problems, but it appears that electromagnetic pumps could be used with coolants that have a high electrical conductivity, such as some liquid metals. Power for the pump could be supplied through slip rings.

For stator cooling the coolant could be pumped by either sealed mechanical pumps or electromagnetic pumps. The electromagnetic pumps would probably be more satisfactory if the coolant has a high enough electrical conductivity, because the possibility of coolant contamination due to air leaks would be eliminated.

ANALYTICAL PROCEDURES

Engine performance was calculated by use of the procedures and curves of reference 5 and the cooled-turbine heat-rejection rates presented in reference 4. These heat-rejection rates are based on the assigned values of compressor equivalent weight flow, turbine aerodynamic design, work split between turbine stages, and turbine blade temperature, solidity, and aspect ratio specified in reference 4. Other assigned values required for calculation of engine performance are listed in the following table:

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Variables and assigned constants	Turbojet		Turboprop
Number of turbine stages	1	2	3
Flight Mach number, M	0.8-3.0	0.8-3.0	0.8
Flight altitude, ft	50,000 to 80,000	Sea level, and 50,000-80,000	Sea level
Sea-level static compressor pressure ratio, p_2'/p_1'	4-6	6-12	12
Turbine-inlet temperature, T_3, $^{\circ}R$	2460,2800	2460,2800	2460
Compressor adiabatic efficiency	0.85	0.85	0.85
Turbine adiabatic efficiency	0.85	0.85	0.85
Primary-combustor efficiency	0.98	0.98	0.98
Afterburner efficiency	0.90	0.90	----
Afterburner temperature, $^{\circ}R$	3500	3500	----
Exhaust-nozzle efficiency	0.90	0.90	0.90
Turboprop gearbox efficiency	----	----	0.95
Turboprop propeller efficiency	----	----	0.80
Primary-combustor pressure ratio, p_3'/p_2'	0.95	0.95	0.95
Tailpipe and afterburner pressure ratio, p_5'/p_4'	0.90	0.90	----
Tailpipe pressure ratio (without afterburner), p_5'/p_4'	0.95	0.95	0.95
Heat-rejection rates, Q/w_T	Ref. 4	Ref. 4	Ref. 4

Symbols are defined in appendix A. The ram recovery p_1'/p_0' was assumed to vary with flight Mach number. Values used were 0.96, 0.87, and 0.65 at Mach numbers of 0.8, 2.0, and 3.0, respectively.

In this report engine performance is given on a relative basis. The standard is the performance calculated for no heat rejection. As an example, relative thrust values are equal to the ratio of the specific thrust with heat rejection to the specific thrust without heat rejection. For turboprop engines the relative equivalent horsepower is obtained from the sum of the shaft horsepower and the equivalent jet thrust horsepower (product of jet thrust and velocity divided by propeller efficiency with proper conversion units). The relative specific fuel consumption for turbojet engines is based on the fuel flow per pound of thrust; for turboprop engines it is based on the fuel flow per equivalent horsepower.

It is assumed that the entire heat load from the liquid-cooled turbines is rejected either ahead of the main compressor, within the main compressor, after the main compressor, or in a sink external to the engine. The changes necessary in the engine performance calculations because of heat rejection at the locations mentioned are discussed in this section.

Heat Rejection ahead of Main Compressor

With heat rejection ahead of the compressor, the compressor-inlet temperature is increased. The increase of air temperature due to this addition of heat can be closely approximated by

$$\Delta\theta_C \approx 0.00819 \left(\frac{Q}{w_T}\right)_{tot} \tag{1}$$

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where $(Q/w_T)_{tot}$ is the sum of the heat-rejection rates for the turbine stator and rotor blades. The constant 0.00819 is $(1+f)/518.7c_p$, where mean values of specific heat of 0.24 Btu per pound per $^{\circ}R$ and fuel-air ratio of 0.02 were used. Because of the small magnitude of this correction, a refinement for variations in specific heat due to temperature level and fuel-air ratio is not warranted.

The compressor specific work $\Delta H_C'$ and compressor-outlet temperature θ_2 are calculated from the following equations for a given compressor pressure ratio:

$$\Delta H_C' = \left(\frac{\Delta H_C'}{\theta_1}\right)_{no} (\theta_1 + \Delta\theta_C) \tag{2}$$

and

$$\theta_2 = \left(\frac{\theta_2}{\theta_1}\right)_{no} (\theta_1 + \Delta\theta_C) \tag{3}$$

where $(\Delta H_C'/\theta_1)_{no}$ and $(\theta_2/\theta_1)_{no}$ are obtained from reference 5 for no heat rejection.

The temperature of the gases is reduced by removal of heat by turbine cooling. This reduction of gas temperature is given by (ref. 5)

$$\Delta\theta_T \approx 0.00638 \left(\frac{Q}{w_T}\right)_{tot} \tag{4}$$

Assuming that the entire heat removal takes place at the turbine inlet,

$$\theta_3 = \theta_{3,no} - \Delta\theta_T \tag{5}$$

Actually, the heat removal from the gases occurs all the way through the turbine, but calculations show that the assumption that all the heat is removed at the turbine inlet has a small effect on calculated performance. The calculations are greatly simplified by this assumption. Once the temperature corrections are made for heat addition and heat removal, the procedures of reference 5 are followed to determine the engine performance when heat is rejected ahead of the main compressor.

Heat Rejection behind Main Compressor

When heat is added at the compressor outlet, only the compressor-outlet temperature is increased; that is,

$$\theta_2 = \left(\frac{\theta_2}{\theta_1}\right)_{no} \theta_1 + \Delta\theta_C \tag{6}$$

where $\Delta\theta_C$ is determined from equation (1). As before, the turbine-inlet temperature θ_3 must be corrected as in equation (5). Once again, the engine performance is determined in accordance with reference 5.

Heat Rejection within Main Compressor

When heat from the turbine stator and rotor blades is rejected to the compressor stator and rotor blades, respectively, both the compressor specific work $\Delta H_C'$ and outlet temperature are affected. If it is assumed that the compressor is divided into three sections of equal pressure ratio with one-third of the heat added at the entrance of each section, then, as shown in appendix B, the total compressor specific work for a constant compressor polytropic efficiency in all stages is

$$\Delta H_C' = \left(\frac{\Delta H_C'}{\theta_1}\right)_f \left(\theta_1 + \Delta\theta_C + \left(\frac{\theta_2}{\theta_1}\right)_f \left\{\left[1 + \left(\frac{\theta_2}{\theta_1}\right)_f\right]\left(\theta_1 + \frac{\Delta\theta_C}{3}\right) + \frac{\Delta\theta_C}{3}\right\}\right) \tag{7}$$

where

$$\left(\frac{\Delta H_C'}{\theta_1}\right)_f = \frac{\left(\frac{\Delta H_C'}{\theta_1}\right)_{no}}{1 + \left(\frac{\theta_2}{\theta_1}\right)_f\left[1 + \left(\frac{\theta_2}{\theta_1}\right)_f\right]}$$

and

$$\left(\frac{\theta_2}{\theta_1}\right)_f = \left(\frac{\theta_2}{\theta_1}\right)_{no}^{1/3}$$

The subscript f denotes the portion of the work or temperature ratio for the cube root of the total compressor pressure ratio with no heat rejection. The compressor-outlet temperature, as shown in appendix B, is given by 4007

$$\theta_2 = \left(\frac{\theta_2}{\theta_1}\right)_f \left\{\left(\frac{\theta_2}{\theta_1}\right)_f \left[\left(\frac{\theta_2}{\theta_1}\right)_f \left(\theta_1 + \frac{\Delta\theta_C}{3}\right) + \frac{\Delta\theta_C}{3}\right] + \frac{\Delta\theta_C}{3}\right\} \tag{8}$$

Once $\Delta H_C'$ and θ_2 are determined, the engine performance is obtained from reference 5.

Heat Rejection in Sink External to Engine

For comparison purposes a thermodynamic study was made for the case where rotating heat exchangers were not used and the coolant was transferred to a heat exchanger where the heat rejected would not enter into the engine cycle. For this case the relative engine thrust or power would be essentially the same as when the heat is rejected behind the main compressor. The relative specific fuel consumption, however, would increase by the same rate that the relative thrust or power decreased. This effect occurs because the fuel-flow rate for a constant turbine-inlet temperature would be constant regardless of the magnitude of the heat-rejection rate. As a result the relative specific fuel consumption would be the reciprocal of the relative thrust or relative equivalent horsepower. (The possible thrust obtainable from heated air that could be discharged from a heat exchanger in flight is not considered.)

RESULTS AND DISCUSSION

Turbojet-Engine Performance

In general, the turbine equivalent work $\Delta H_T'/\theta_3$ is increased by turbine cooling, because the gas temperature is reduced (see eq. (5)). In addition, when heat is rejected ahead of and within the compressor, the compressor specific work $\Delta H_C'$ is increased in accordance with equations (2) and (7). As $\Delta H_C'$ goes up, $\Delta H_T'/\theta_3$ rises. The resulting increases in $\Delta H_T'/\theta_3$ due to heat rejection for a given turbine-inlet temperature and adiabatic efficiency reduce the turbine pressure ratio p_4'/p_3'. A reduction in p_4'/p_3' results in a decrease in the net specific thrust $F_n/w_C\sqrt{\theta_1}$.

The effect of heat rejection on specific fuel consumption is generally smaller than the effect on thrust, except for heat rejection external to the engine, as previously explained. Since the compressor-outlet temperature is increased because of heat rejection within the engine, less
4007 fuel per pound of compressor-inlet air is burned for a cooled engine than for an uncooled engine for a constant turbine-inlet temperature. Thus, the thrust specific fuel consumption is affected to a smaller degree than the specific thrust by heat rejection ahead of, within, and behind the compressor, since fuel consumption as well as engine thrust is decreased.

The performance variations due to heat rejection at a constant turbine-inlet temperature are presented herein by showing turbojet-engine performance with heat rejection relative to the case with no heat rejection.

As would be expected from the preceding discussion, figure 2 shows that the relative thrust of a turbojet engine decreases with increasing heat rejection. It will also be noted that the farther forward on the engine the heat is rejected, the higher will be the thrust losses due to heat rejection. When the heat is rejected behind the main compressor or in a sink external to the engine, the thrust loss results from a decrease in turbine gas temperature only. The thrust loss from added compressor and turbine work is higher with heat rejection ahead of the compressor than with heat rejection within the compressor, because compressor work is directly proportional to inlet temperature. The compressor-inlet temperature is highest when all the heat is rejected ahead of the compressor.

The specific fuel consumption increases when the heat is rejected within or ahead of the compressor or in a sink external to the engine, but it decreases when the heat is rejected behind the compressor. The reason for the improved specific fuel consumption when heat is rejected behind the compressor is that the effect on engine performance is exactly the same as if the turbine-inlet temperature were slightly reduced (the fuel-flow rate decreases in the same manner that θ_3 decreases from eq. (5)). The gas temperature for best specific fuel consumption for a turbojet engine is lower than the value of 2460° R used in the calculation of results shown in figure 2.

The results shown for heat rejection to a sink external to the engine represent the worst possible case with respect to specific fuel consumption, because all the heat is lost to the cycle. If some of this heat could be utilized for obtaining jet thrust or in heating the fuel, the specific fuel consumption would be lower. As stated previously, the performance for heat rejection to a sink external to the engine was considered for comparison purposes only. No consideration was given as to the practicality of such an arrangement. The turbojet-engine performance resulting when heat is rejected in a sink external to the engine will not be discussed further. This performance can be obtained, however, from any of the curves presented, since the relative thrust is the same as for heat rejection behind the main compressor, and the relative specific fuel consumption is the reciprocal of the relative thrust.

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In order to determine the engine performance that can be expected from liquid-cooled engines for a range of engine and flight conditions, the heat-rejection rates obtained from the study in reference 4 were incorporated into engine performance calculations. Heat-rejection-rate variations that occur with changes in turbine-inlet temperature, compressor pressure ratio, flight Mach number, compressor equivalent weight flow, and flight altitude are taken from reference 4 and are shown in figure 3. The relative performances obtained with the heat-rejection rates shown are presented in figures 4 to 7. These performance results show only the effects of heat rejection as influenced by turbine-inlet temperature, compressor pressure ratio, flight Mach number, and so forth. The direct effects of turbine-inlet temperature, compressor pressure ratio, flight Mach number, compressor equivalent weight flow, and flight altitude on engine power (thrust or horsepower) and fuel consumption are not shown. These direct effects are included in other studies (such as refs. 6 and 7). The combined effect of liquid-cooling, turbine-inlet temperature, compressor pressure ratio, and so forth, can be obtained by multiplying the relative performance values given herein by the absolute performance values without heat rejection such as shown in references 6 and 7.

Effect of compressor pressure ratio. - In figure 4 relative thrust and relative specific fuel consumption are plotted against sea-level compressor pressure ratio for heat rejection ahead of, within, and behind the main compressor. The curves are shown for turbine-inlet temperatures of 2460° and 2800° R, a flight Mach number of 2.0 at an altitude of 50,000 feet, and a sea-level static compressor equivalent weight flow of 35.0 pounds per second per square foot. Both nonafterburning and afterburning liquid-cooled turbojet engines are considered.

Nonafterburning engine: For the one-stage turbines operating at a turbine-inlet temperature of 2460° R (fig. 4(a)), the thrust reductions due to heat rejection ahead of, within, and behind the compressor are roughly $2\frac{1}{2}$, 2, and $1\frac{1}{2}$ percent, respectively. The amount of heat that must be rejected for the two-stage turbines is almost double that for the one-stage turbines (see fig. 3(a)). This increase in $(Q/w_T)_{tot}$ is reflected in larger thrust reductions for the two-stage turbines. For a turbine-inlet temperature of 2460° R and change in $(p_2'/p_1')_{sl}$ from 6 to

12, the thrust is decreased from about 5 to 8 percent for heat rejection ahead of the compressor. At these same conditions, thrust reductions of about 4 to 6 and 2.5 to 3 percent are observed for heat rejection within and behind the compressor, respectively.

Reflecting the relative thrust results of figure 4(a) into relative specific fuel consumption shows a small effect due to heat rejection, as explained previously. The specific fuel consumption for heat rejection behind the compressor is less than that for no heat rejection for the
4007 one- and two-stage turbines.

When the turbine-inlet temperature is increased to 2800° R (fig. 4(b)), the same trends observed in figure 4(a) are repeated, the percentage reductions in thrust being greater at the higher turbine-inlet temperature.

Afterburning engine: Figures 4(c) and (d) show the performance variations due to heat rejection for afterburning turbojet engines operating at turbine-inlet temperatures of 2460° and 2800° R with an afterburner temperature of 3500° R. The addition of the afterburner tends to cancel the thrust reductions due to heat rejection. As a result, the largest decrease in thrust occurring for the afterburning engine is only about 3 percent.

The specific-fuel-consumption results shown in figures 4(c) and (d) are affected directly by the thrust changes, since the fuel consumption remains practically constant regardless of heat rejection, because the over-all engine temperature ratio for a specified afterburner temperature is independent of transfer of heat from one place to another inside the engine. The percentage increase in specific fuel consumption is therefore about the same as the percentage decrease in thrust. The percentages are not exactly the same, because the combustion efficiency of the afterburner is lower than that of the primary burner.

As a further explanation for a relative specific fuel consumption greater than 1.0 for heat rejection behind the main compressor, reference 6 shows that the specific fuel consumption of afterburning engines improves as turbine-inlet temperature increases. Since cooling the turbine and rejecting the heat behind the compressor have the same effect as decreasing the turbine-inlet temperature, the relative specific fuel consumption therefore increases opposite to the case for nonafterburning engines.

Effect of flight Mach number. - The effects of flight Mach number M on liquid-cooled turbojet-engine performance when rejecting heat ahead of, within, and behind the compressor are illustrated in figure 5. Only a one-stage turbine with $(p_2'/p_1')_{sl} = 6.0$ and a two-stage turbine with

$(p_2'/p_1')_{sl} = 12.0$ are considered for a nonafterburning engine operating at a turbine-inlet temperature of 2460° R, an altitude of 50,000 feet, and a compressor equivalent weight flow $(w_C\sqrt{\theta_1}/A_C\delta_1)_{sl}$ of 35 pounds per second per square foot. The effects of other values of $(p_2'/p_1')_{sl}$, a turbine-inlet temperature of 2800° R, and use of an afterburner on the results presented in figure 5 may be inferred from figure 4. Effects of variations in $(w_C\sqrt{\theta_1}/A_C\delta_1)_{sl}$ and altitude are discussed in the following sections. 4007

As shown in figure 3(b) for a one-stage turbine, $(Q/w_T)_{tot}$ decreases about 37 percent as M changes from 0.8 to 3.0. The effect of this change on the relative thrust is shown in figure 5(a); the relative-specific-thrust changes due to liquid-cooling are of approximately the same trend and magnitude as indicated in figure 2 for corresponding values of Q/w_T. Flight Mach number effects on heat rejection cause little change in thrust specific fuel consumption for the one-stage turbine. This effect is somewhat at variance with figure 2 and is caused by a change in optimum turbine-inlet temperature for a given compressor pressure ratio as flight Mach number is increased.

The effect of flight Mach number on the specific thrust of a two-stage turbine engine (fig. 5(b)) is different from that for the one-stage turbine. There is a slight decrease in thrust as the flight Mach number is increased even though the heat-rejection rates decrease. An explanation for this behavior will be given in the following discussion on specific fuel consumption variations with flight Mach number.

The specific fuel consumption for an engine with a sea-level static compressor pressure ratio of 12 (fig. 5(b)) rises rather rapidly as the flight Mach number increases. This increase is out of proportion to that shown in figure 2 for corresponding variations in Q/w_T or when compared with corresponding changes in relative thrust. These large effects on specific fuel consumption result from changes in over-all engine pressure and temperature ratios as flight Mach number increases. At the high Mach numbers the engine temperature ratio is decreasing, and heat removal at the turbine by liquid-cooling results in a more significant loss in thrust than occurs at lower flight speeds or lower compressor pressure ratios. In addition the engine is in a less efficient region of operation. As a result, the specific fuel consumption increases beyond that expected from an examination of figure 2. At a lower compressor pressure ratio, as shown in figure 5(a), this effect is not observed, because the best-economy turbine-inlet temperature has not been exceeded for that pressure ratio. This trend gives evidence, similar to that obtained in many other unpublished cycle analyses, that high compressor pressure ratios are not desirable at high supersonic flight speeds.

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<u>Effect of compressor equivalent weight flow</u>. - Figure 6 shows the effects of sea-level compressor equivalent weight flow, $(w_C\sqrt{\theta_1}/A_C\delta_1)_{sl}$, for engines operating at a flight Mach number of 2.0, all other conditions being the same as in figure 5. For the one-stage turbine, no significant effect is observed for either relative thrust or relative specific fuel consumption as $(w_C\sqrt{\theta_1}/A_C\delta_1)_{sl}$ is varied.

As $(w_C\sqrt{\theta_1}/A_C\delta_1)_{sl}$ is varied from 20 to 35 pounds per second per square foot for the two-stage turbine, the relative thrust tends to increase slightly. This increase is due to the decreasing $(Q/w_T)_{tot}$, as shown in figure 3(c). Changes in relative specific fuel consumption with $(w_C\sqrt{\theta_1}/A_C\delta_4)_{sl}$ are a direct result of the thrust changes.

<u>Effect of altitude</u>. - The effects of altitude on the performance of liquid-cooled turbojet engines due to heat rejection are presented in figure 7. The decreases in thrust obtained for both the one- and two-stage turbines as the altitude varies from 50,000 to 80,000 feet are again a direct result of the increases in $(Q/w_T)_{tot}$ over this range of altitude (see fig. 3(d)). For the one-stage turbine, the thrust is reduced from about 3 to 5 percent and 1.5 to 2.5 percent with heat rejection ahead of and behind the compressor, respectively, as altitude is increased from 50,000 to 80,000 feet. For these same conditions, thrust reductions from about 8 to 13 percent and 3 to 5 percent occur for the two-stage turbine. The specific fuel consumption behaves in the same manner as shown in figure 2 as the relative specific thrust decreases.

Turboprop-Engine Performance

The very small blades of turboprop engines are generally more difficult to cool with air than the larger turbojet blades. The reasons are that it is more difficult to provide the required augmented heat-transfer surface inside small air-cooled blades and that the heat transferred to the gas per unit of blade surface area is higher for the smaller blades. Consequently, liquid-cooling of the very small turboprop blades may be more promising than air-cooling. For this reason, the effect of heat rejection on the performance of liquid-cooled turboprop engines was investigated. As in the turbojet-engine study, heat was rejected ahead of, within, and behind the main compressor and in a sink external to the engine. Calculations were made for a turbine-inlet temperature of 2460° R, a sea-level static compressor pressure ratio of 12, and a flight Mach number of 0.8 at sea level. Other assigned values necessary for these calculations are listed in the table in the section ANALYTICAL PROCEDURES.

The relative equivalent horsepower and relative specific fuel consumption are shown in the following table for heat rejection ahead of, within, and behind the main compressor and in a sink external to the engine:

Location of liquid-cooling heat-rejection sink	Relative equivalent horsepower	Relative specific fuel consumption
Ahead of compressor	0.883	1.067
Within compressor	.916	1.041
Behind compressor	.966	1.006
External to engine	.966	1.035

In order to compare the effects of heat rejection on turboprop- and turbojet-engine performance, calculations were made for a liquid-cooled turbojet engine operating at the same flight conditions and same compressor pressure ratio as the turboprop engine. The results are shown in the following table:

Location of liquid-cooling heat-rejection sink	Relative thrust	Relative specific fuel consumption
Ahead of compressor	0.947	1.013
Within compressor	.960	1.011
Behind compressor	.986	.995
External to engine	.986	1.014

Comparison of the results from the two tables shows that the performance of a liquid-cooled turboprop engine is affected more than that of a liquid-cooled turbojet when heat is rejected at the same location of each engine.

Comparison of Engine Performance with Liquid- and Air-Cooling

Variations in performance due to cooling result, for the most part, from completely different reasons for air- and liquid-cooling. The only similarity is that in both cases heat is removed from the gases by the blades. This heat removal results in a slight reduction in turbine work due to reducing the turbine gas temperature. With liquid-cooling, the heat removed is unavailable for producing jet thrust. With air-cooling, however, the heat removed from the gases is transferred to the cooling air. The cooling air mixes with the gases after leaving the turbine blade, so that the heat temporarily removed by cooling is still available for producing jet thrust. For this reason the heat removal due to cooling is less serious for air-cooling than for liquid-cooling for turbojet engines. The effect is similar for turboprop engines.

<u>Turbojet engines</u>. - As discussed in reference 6, the primary effects of air-cooling on turbojet engines are (1) the exhaust-gas temperature for a given turbine-inlet temperature is reduced, because part of the compressor air (the air used for cooling) bypasses the burner and is mixed with the combustion gases at or downstream of the turbine, and (2) additional turbine work is required because of cooling-air pumping both in the compressor and in the turbine rotor. This additional turbine work results in added temperature and pressure drops across the turbine. Both of these effects reduce the engine thrust and usually cause an increase in specific fuel consumption. The first effect is by far the largest. The second is often made even smaller by recovery of part of the pumping work by the reaction of cooling-air on the blades as it is discharged into the gas stream. This effect will be discussed later.

For the liquid-cooling schemes that use heat rejection in or ahead of the compressor, the primary cause for losses in performance is that the compressor work (and in turn the turbine work) is increased because of higher air temperatures. This additional work reduces the engine thrust and increases the specific fuel consumption as discussed previously. When the heat is rejected at the compressor exit, the thrust reduction results entirely from a reduction in turbine gas temperature. Generally, the specific fuel consumption is improved for a constant turbine-inlet temperature and heat rejection at the compressor exit (see fig. 2).

In order to compare the effects of liquid- and air-cooling on performance, results presented in reference 6 for air-cooling, using air bled from the compressor exit and a flight Mach number of 2, were cross-plotted so that they could be presented in the same form as the liquid-cooling results presented herein. The effect of air-cooling on specific thrust and specific fuel consumption is shown in figure 8 as a function of sea-level static compressor pressure ratio for both afterburning and nonafterburning turbojet engines for turbine-inlet temperatures of 2460° and 2800° R. By comparing this figure with figure 4, the effects of air- and liquid-cooling on engine performance can be compared.

The previously mentioned comparison can be more easily seen by plotting the performance of air- and liquid-cooled engines on the same curve, as shown in figure 9 for nonafterburning turbojet engines and in figure 10 for afterburning turbojet engines. The comparison is made for a flight Mach number of 2.0 at an altitude of 50,000 feet for a sea-level static compressor equivalent weight flow of 35 pounds per second per square foot of frontal area.

It is difficult to evaluate the two cooling methods from figures 9 and 10 alone, because the required dilution (ratio of cooling-air flow to compressor-inlet flow) is not known until calculations are made for a particular engine and turbine blade configuration. Dilution increases with turbine-inlet temperature and flight Mach number and decreases as the blade cooling effectiveness is improved. An approximate indication of air-cooling requirements can be given, however, for the flight conditions considered in figures 9 and 10. A good air-cooling system using air bled from the compressor exit would probably require between 3 and 6 percent cooling air at a turbine-inlet temperature of 2460° R and between 6 and 9 percent at 2800° R at the conditions specified in figures 9 and 10. The type of liquid-cooling scheme that can be used depends on factors discussed earlier. 4007

For both nonafterburning and afterburning engines (figs. 9 and 10), better engine performance can be obtained with liquid-cooling than with air-cooling if the heat from liquid-cooling can be rejected at the compressor exit. If, however, the complication is too great or the compressor-discharge temperatures are too high to reject heat at this location, it is often questionable whether there is any advantage to liquid-cooling over air-cooling based on engine performance.

Although not shown herein, the comparison between air- and liquid-cooling would be similar at other altitudes and other compressor equivalent weight flows. If air-cooling is used at flight Mach numbers of 2.5 and higher, it may be necessary to use heat exchangers to reduce cooling-air temperatures because of excessively high compressor-discharge temperatures. If air-to-air heat exchangers are used at the compressor inlet for reducing cooling-air temperature, the engine performance will suffer because of heat addition to the compressor air. In this case the air-cooled engine performance would be poorer than that indicated in figures 9 and 10, and liquid-cooling would possibly look more promising in comparison. At these flight conditions compressor-discharge temperatures would be too high to make liquid-cooling heat rejection at the compressor exit feasible. Only heat rejection ahead of or within the compressor could be considered for liquid-cooling.

From this study, therefore, it appears for turbojet engines that (1) higher flight Mach numbers are possible with a liquid-cooling system with heat rejection in a heat exchanger ahead of the engine compressor than with an air-cooling system using compressor bleed without refrigeration; (2) when it is possible to use a liquid-cooling system in which heat is rejected at the compressor exit, the engine performance will be superior to that obtained with air-cooling; and (3) with heat rejected from liquid-cooling at locations other than the compressor exit, air-cooled engine performance will probably be superior until a flight Mach number is reached at which some device is required for cooling the cooling air after it is bled from the engine compressor.

Turboprop engines. - In this investigation on liquid-cooling effects on engine performance, turboprop calculations were made at a turbine-inlet temperature of 2460° R, a flight Mach number of 0.8 at sea level, and a compressor pressure ratio of 12. Air-cooled engine performance was not calculated at exactly these same conditions in reference 6; but from the trends of the results, the estimated performance can be predicted at the same engine and flight conditions used in the liquid-cooling calculations. In the following table the engine performance with liquid-cooling is tabulated, and the approximate air-cooling dilution that would result
4007 in the same performance is shown. The values of equivalent horsepower and equivalent specific fuel consumption are relative to the case with no heat rejection:

Location of liquid-cooling heat-rejection sink	Relative equivalent horsepower with liquid-cooling	Approximate air-cooling dilution for same relative equivalent horsepower as with liquid-cooling	Relative specific fuel consumption with liquid-cooling	Approximate air-cooling dilution for same relative specific fuel consumption as with liquid-cooling
Ahead of compressor	0.883	0.08	1.067	0.12
Within compressor	.916	.06	1.041	.09
Behind compressor	.966	.02	1.006	.015
External to engine	.966	.02	1.035	.07

From the results shown in this table for a turboprop engine and those shown in figures 9 and 10 for turbojet engines, it appears that, relative to air-cooled engine performance, the performance of liquid-cooled turboprop engines suffers more than that of liquid-cooled turbojet engines. In other words, air-cooled turboprop engines can use more cooling air than air-cooled turbojet engines before the performance is worse than for the respective liquid-cooled engine. There is the possibility, however, that the required ratio of cooling-air flow to gas flow may be higher for turboprop engines than for turbojet engines because of higher gas-to-blade heat-transfer rates that result with small blades. The exact quantities of cooling air required could only be obtained by use of extensive analysis. It can be seen from the table, however, that an air-cooled turbine would have to require in excess of 9 and 12 percent of the compressor air for cooling before liquid-cooled engine performance would be superior with heat rejection within and ahead of the compressor, respectively. It is estimated, however, that, if the heat from a liquid-cooling system can be rejected behind the compressor, the turboprop-engine performance will be better than could be expected with air-cooling schemes.

It is difficult to reach definite conclusions as to the relative advantages of air- and liquid-cooling. Studies of fabrication and operational problems are required to determine which system would be more practical. As an example, turboprop blades may be so small that air-cooling may not be feasible. In such a case larger performance losses resulting from liquid-cooling with heat rejection within or ahead of the compressor may have to be tolerated in order to realize the benefits of higher turbine-inlet temperatures. From a study of reference 6 and the air-cooling dilutions that correspond to the same liquid-cooled engine performance, it appears that, if a liquid-cooled turboprop engine could 4007
be operated at a turbine-inlet temperature 500° F above that possible for an uncooled engine and if the heat were rejected within the compressor, the power output of the engine could be increased a little over 50 percent with about a 4-percent saving in specific fuel consumption.

A factor that favors air-cooling should also be considered when comparing the results of air- and liquid-cooling on performance of both turbojet and turboprop engines. The performance of air-cooled engines will probably be somewhat superior to that reported in reference 6. Since publication of reference 6, an investigation has been conducted on the effects of air-cooling on turbine efficiency. The results obtained experimentally from two turbines are reported in reference 8. This reference shows that the discharge of cooling air at the turbine blade tips can result in added turbine work, with a resulting improvement in efficiency above that used in the calculations of reference 6. An analytical study reported in reference 9 indicates that this added work is primarily the result of reaction of the cooling air on the turbine blade after it is discharged at the blade tip. This efficiency improvement will vary with different turbines, so that the exact amount of performance improvement due to this effect is difficult to predict. It can be said, however, that air-cooled engine performance will generally be no worse than that shown in reference 6. For turbojet engines the thrust and specific fuel consumption could be improved up to about 2 percent because of efficiency improvement. For turboprop engines the power and specific fuel consumption could be improved up to about 10 percent. These improvements are based on a possible 5-percent improvement in turbine efficiency and the results shown in reference 6.

CONCLUDING REMARKS

As a result of an analytical investigation of the effects of liquid-cooling on turbojet- and turboprop-engine performance, the following conclusions can be drawn:

(1) It may be desirable to use liquid-cooling rather than air-cooling for some gas-turbine-engine applications. With the proper type of liquid-cooling system, higher flight speeds appear feasible, and the cooling of very small turbine blades will probably be easier than with air-cooling.

(2) The practicality of liquid-cooling can be greatly improved for gas-turbine engines if coolants such as liquid metals or water at supercritical pressures can be used so that coolant temperatures can be increased to the point that heat-rejection rates are substantially reduced. A suggested method of obtaining this improvement is to utilize a rotating heat exchanger to eliminate coolant seals between stationary and rotating parts. Such a system could reject heat ahead of, within, or behind the engine compressor.

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(3) Liquid-cooled engine performance is substantially affected by the location of heat rejection within the engine. For the best engine performance, heat rejection should occur at the compressor discharge. The performance is considerably poorer if heat is rejected at the compressor inlet. Heat rejection at the compressor inlet would be desirable, however, to permit cooled-engine operation at very high flight speeds. This location is also advantageous in that a system with adequate coolant pumping characteristics may be provided, and it appears practical with respect to fabrication and operation.

(4) For turbojet engines it appears that: (a) Higher flight Mach numbers are possible with a liquid-cooling system with heat rejection in a heat exchanger ahead of the engine compressor than with an air-cooling system using compressor bleed air without refrigeration. (b) When it is possible to use a liquid-cooling system with heat rejection at the compressor exit, the engine performance will be superior to that obtained with air-cooling. (c) With heat rejected from liquid-cooling at locations other than the compressor exit, air-cooled engine performance will probably be superior until a flight Mach number is reached where some device is required for cooling the air after it is bled from the compressor.

(5) For turboprop engines it appears that cooling of small turbine blades may be more successful with liquids than with air, but the heat from the liquid coolant will probably have to be rejected at the compressor exit in order to also ensure engine performance that is superior to that with air-cooling.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, February 14, 1956

APPENDIX A

SYMBOLS

The following symbols are used in this report:

A_C compressor frontal area, sq ft

c_p specific heat at constant pressure, Btu/(lb)(^{o}R) 4007

F_n net thrust, lb

f fuel-air ratio

H enthalpy, Btu/lb

M flight Mach number

p pressure, lb/sq ft

Q/w_T heat-rejection rate, Btu/lb

T temperature, ^{o}R

w weight-flow rate, lb/sec

δ ratio of total pressure to NACA standard sea-level pressure of 2116 lb/sq ft

θ ratio of total temperature to NACA standard sea-level temperature of 518.7^{o} R

Subscripts:

C compressor

f denotes portion of work or temperature ratio for cube root of total compressor pressure ratio with no heat rejection

no no heat rejection

sl sea-level static

T turbine

tot	total
0	ahead of engine
1	compressor inlet
2	compressor outlet
3	turbine inlet
4	turbine outlet
5	exhaust nozzle

Superscript:

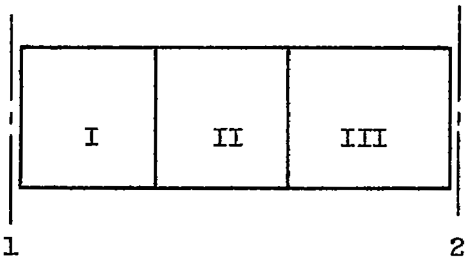
'	stagnation conditions

APPENDIX B

DERIVATION OF EQUATIONS FOR COMPRESSOR SPECIFIC WORK AND OUTLET

TEMPERATURE FOR HEAT REJECTION WITHIN MAIN COMPRESSOR

Assume that the main compressor is divided into three parts of equal compressor pressure ratio with one-third of the heat being added at the entrance of each part. For constant polytropic efficiency the equivalent specific work of each section will also be equal. Thus, when the compressor is represented schematically, 4007



there results

$$\left(\frac{p_2'}{p_1'}\right)_I = \left(\frac{p_2'}{p_1'}\right)_{II} = \left(\frac{p_2'}{p_1'}\right)_{III} = \left(\frac{p_2'}{p_1'}\right)^{1/3} \tag{B1}$$

The total compressor specific work is

$$\Delta H_C' = (\Delta H_C')_I + (\Delta H_C')_{II} + (\Delta H_C')_{III} \tag{B2}$$

Equation (B2) may be written as

$$\Delta H_C' = \left(\frac{\Delta H_C'}{\theta_1}\right)_I (\theta_1)_I + \left(\frac{\Delta H_C'}{\theta_1}\right)_{II} (\theta_1)_{II} + \left(\frac{\Delta H_C'}{\theta_1}\right)_{III} (\theta_1)_{III} \tag{B3}$$

The terms $(\theta_1)_I$, $(\theta_1)_{II}$, and $(\theta_1)_{III}$ can be written as (assuming that heat is added equally at the entrance of each section)

$$(\theta_1)_I = \left(\theta_1 + \frac{\Delta\theta_C}{3}\right) \tag{B4}$$

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$$(\theta_1)_{II} = (\theta_2)_I + \frac{\Delta\theta_C}{3} = \left(\frac{\theta_2}{\theta_1}\right)_I (\theta_1)_I + \frac{\Delta\theta_C}{3} = \left(\frac{\theta_2}{\theta_1}\right)_I \left(\theta_1 + \frac{\Delta\theta_C}{3}\right) + \frac{\Delta\theta_C}{3} \tag{B5}$$

$$(\theta_1)_{III} = (\theta_2)_{II} + \frac{\Delta\theta_C}{3} = \left(\frac{\theta_2}{\theta_1}\right)_{II} (\theta_1)_{II} + \frac{\Delta\theta_C}{3}$$

$$= \left(\frac{\theta_2}{\theta_1}\right)_{II} \left[\left(\frac{\theta_2}{\theta_1}\right)_I \left(\theta_1 + \frac{\Delta\theta_C}{3}\right) + \frac{\Delta\theta_C}{3}\right] + \frac{\Delta\theta_C}{3} \tag{B6}$$

where $\Delta\theta_C$ is obtained from equation (1). Now, let the subscript f denote the portion of temperature or equivalent work for the cube root of the total compressor pressure ratio and for a constant polytropic efficiency through the compressor. Then,

$$\left(\frac{\theta_2}{\theta_1}\right)_I = \left(\frac{\theta_2}{\theta_1}\right)_{II} = \left(\frac{\theta_2}{\theta_1}\right)_{III} = \left(\frac{\theta_2}{\theta_1}\right)_f = \left(\frac{\theta_2}{\theta_1}\right)^{1/3} \tag{B7}$$

and

$$\left(\frac{\Delta H_C'}{\theta_1}\right)_I = \left(\frac{\Delta H_C'}{\theta_1}\right)_{II} = \left(\frac{\Delta H_C'}{\theta_1}\right)_{III} = \left(\frac{\Delta H_C'}{\theta_1}\right)_f \tag{B8}$$

Combining equations (B3) to (B8) and simplifying give the total compressor specific work for heat rejection within the compressor as

$$\Delta H_C' = \left(\frac{\Delta H_C'}{\theta_1}\right)_f \left(\theta_1 + \Delta\theta_C + \left(\frac{\theta_2}{\theta_1}\right)_f \left\{\left[1 + \left(\frac{\theta_2}{\theta_1}\right)_f\right] \left(\theta_1 + \frac{\Delta\theta_C}{3}\right) + \frac{\Delta\theta_C}{3}\right\}\right) \tag{7}$$

For the case where there is no heat rejection, $\Delta\theta_C$ is set equal to zero and equation (7) becomes

$$\left(\frac{\Delta H_C'}{\theta_1}\right)_f = \frac{\left(\frac{\Delta H_C'}{\theta_1}\right)_{no}}{1 + \left(\frac{\theta_2}{\theta_1}\right)_f \left[1 + \left(\frac{\theta_2}{\theta_1}\right)_f\right]} \tag{B9}$$

where, from equation (B7),

$$\left(\frac{\theta_2}{\theta_1}\right)_f = \left(\frac{\theta_2}{\theta_1}\right)_{no}^{1/3} \tag{B10}$$

The compressor-outlet temperature may be obtained from the following:

$$\theta_2 = \left(\frac{\theta_2}{\theta_1}\right)_{III} (\theta_1)_{III} \tag{B11}$$

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Now, combining equations (B11), (B7), (B6), and (B4) and simplifying,

$$\theta_2 = \left(\frac{\theta_2}{\theta_1}\right)_f \left\{\left(\frac{\theta_2}{\theta_1}\right)_f \left[\left(\frac{\theta_2}{\theta_1}\right)_f \left(\theta_1 + \frac{\Delta\theta_C}{3}\right) + \frac{\Delta\theta_C}{3}\right] + \frac{\Delta\theta_C}{3}\right\} \tag{8}$$

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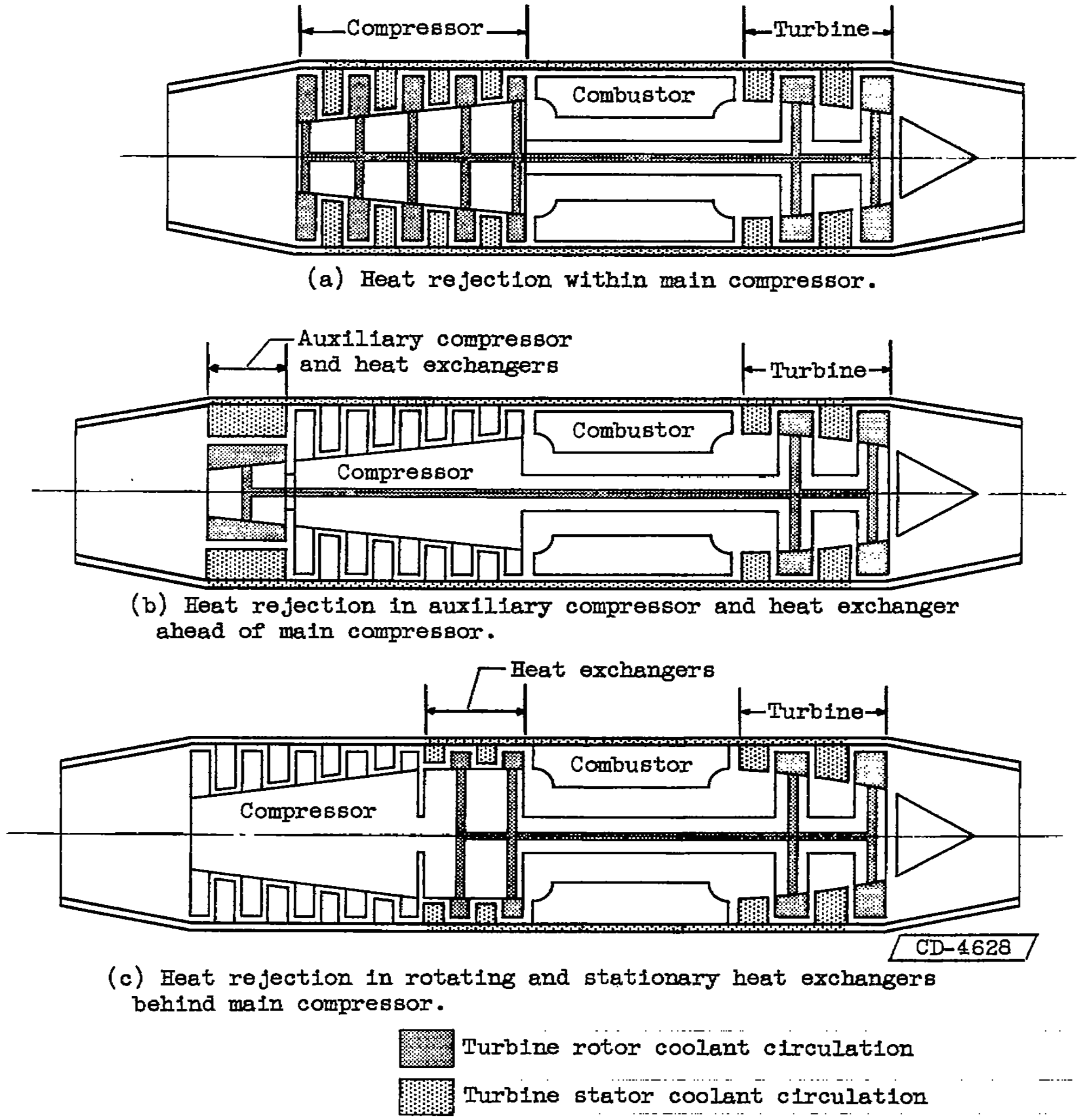

(a) Heat rejection within main compressor.

(b) Heat rejection in auxiliary compressor and heat exchanger ahead of main compressor.

(c) Heat rejection in rotating and stationary heat exchangers behind main compressor.

Figure 1. - Three possible methods of rejecting heat in liquid-cooled turbine engines.

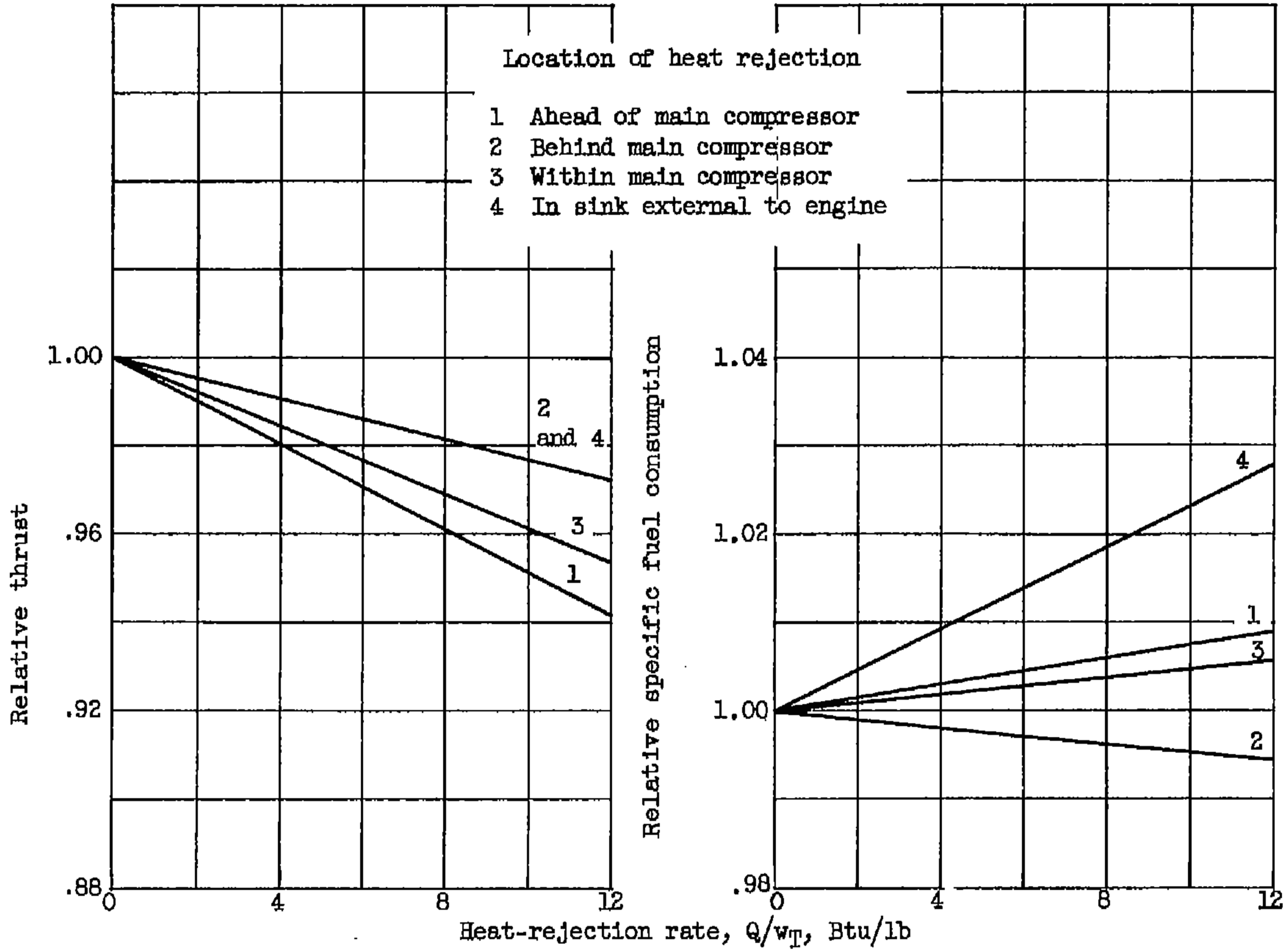

Figure 2. - Effect of heat-rejection rate on performance of nonafterburning liquid-cooled turbojet engine. Flight Mach number, 2.0 (in stratosphere); sea-level static compressor pressure ratio, 6; turbine-inlet temperature, 2460° R.

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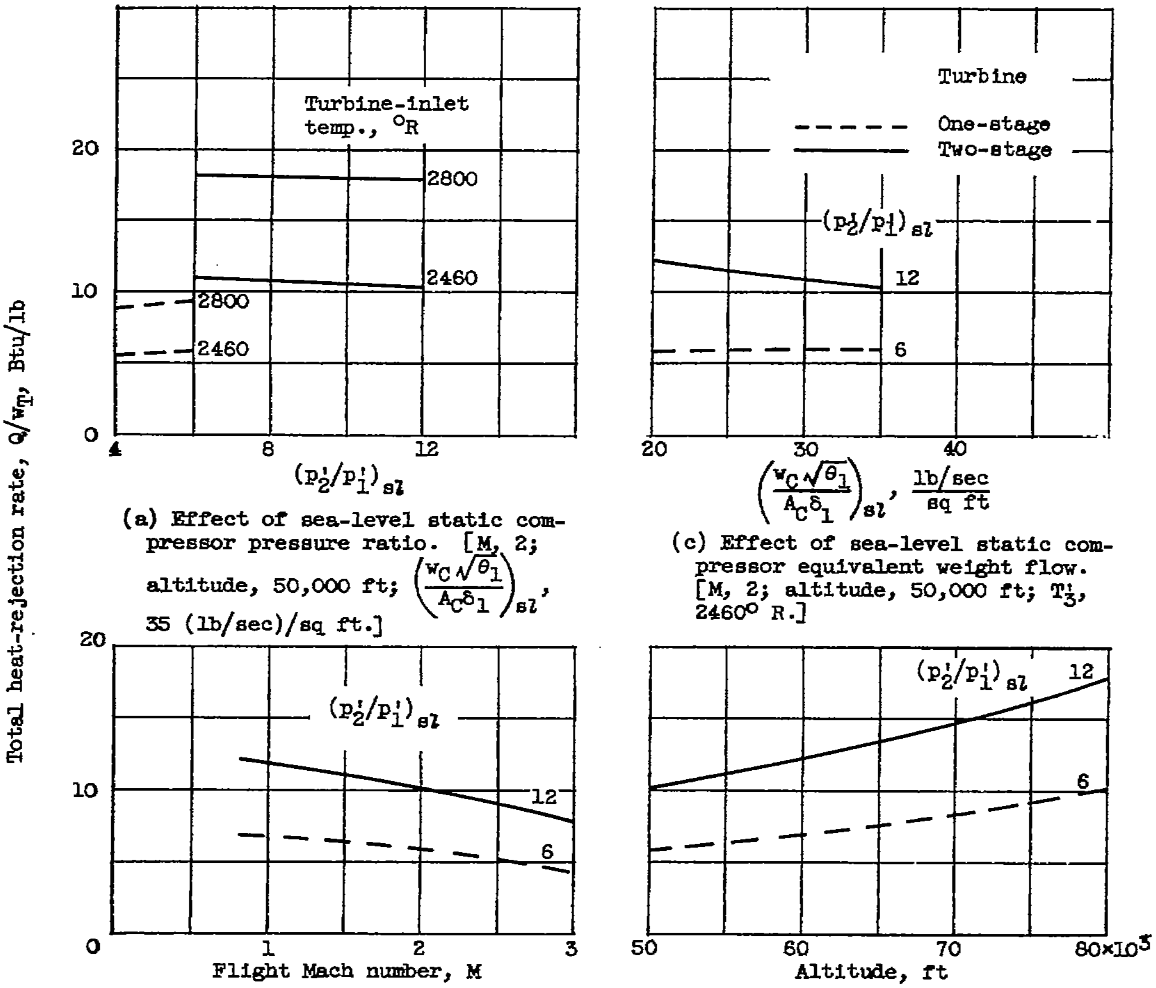

(a) Effect of sea-level static compressor pressure ratio. [M, 2; altitude, 50,000 ft; $\left(\frac{w_C\sqrt{\theta_1}}{A_C\delta_1}\right)_{sl}$, 35 (lb/sec)/sq ft.]

(b) Effect of flight Mach number. [T_3', 2460° R; altitude, 50,000 ft; $\left(\frac{w_C\sqrt{\theta_1}}{A_C\delta_1}\right)_{sl}$, 35 (lb/sec)/sq ft.]

(c) Effect of sea-level static compressor equivalent weight flow. [M, 2; altitude, 50,000 ft; T_3', 2460° R.]

(d) Effect of flight altitude. [M, 2; T_3', 2460° R; $\left(\frac{w_C\sqrt{\theta_1}}{A_C\delta_1}\right)_{sl}$, 35 (lb/sec)/sq ft.]

Figure 3. - Effect of engine and flight conditions on liquid-cooling heat-rejection rates (ref. 4).

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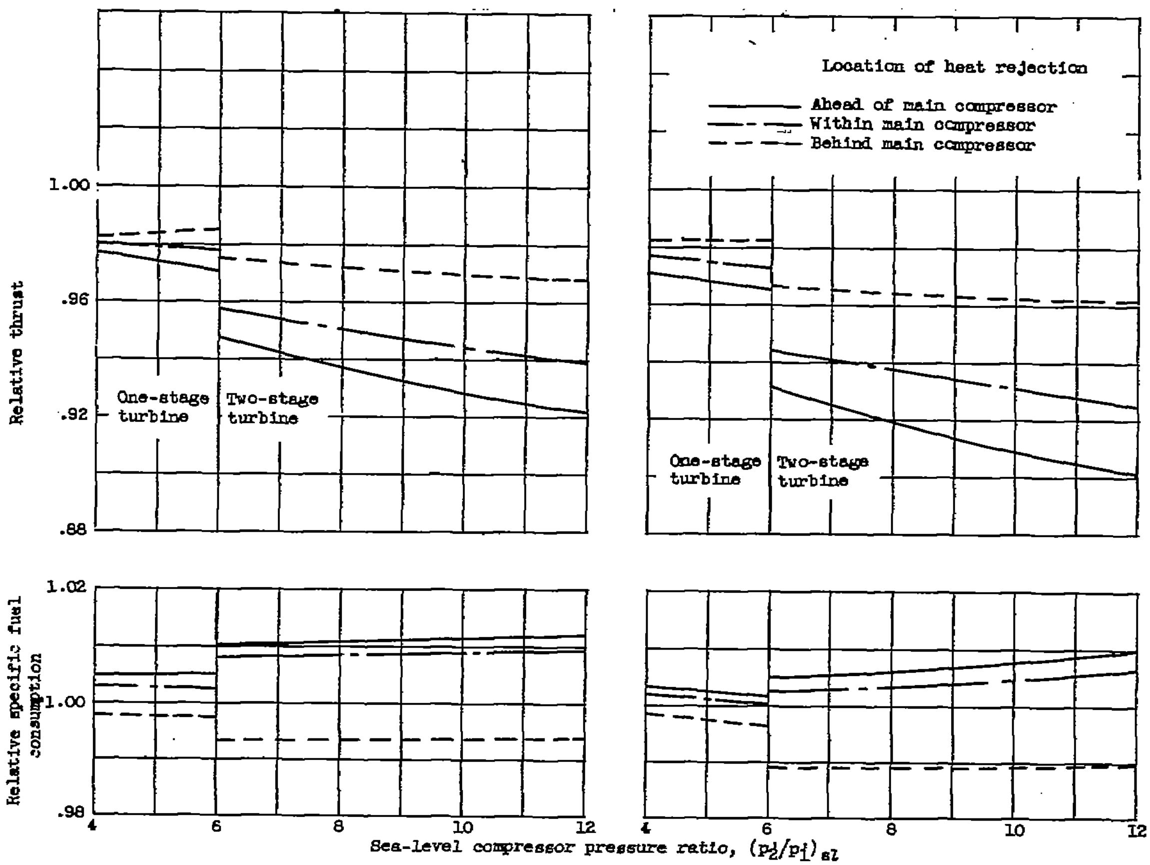

(a) Turbine-inlet temperature, 2460° R; nonafterburning engine.

(b) Turbine-inlet temperature, 2800° R; nonafterburning engine.

Figure 4. - Relative turbojet-engine performance resulting from effects of compressor pressure ratio on liquid-cooling heat-rejection rates. Flight Mach number, 2.0; altitude, 50,000 feet; sea-level static compressor equivalent weight flow, 35 pounds per second per square foot.

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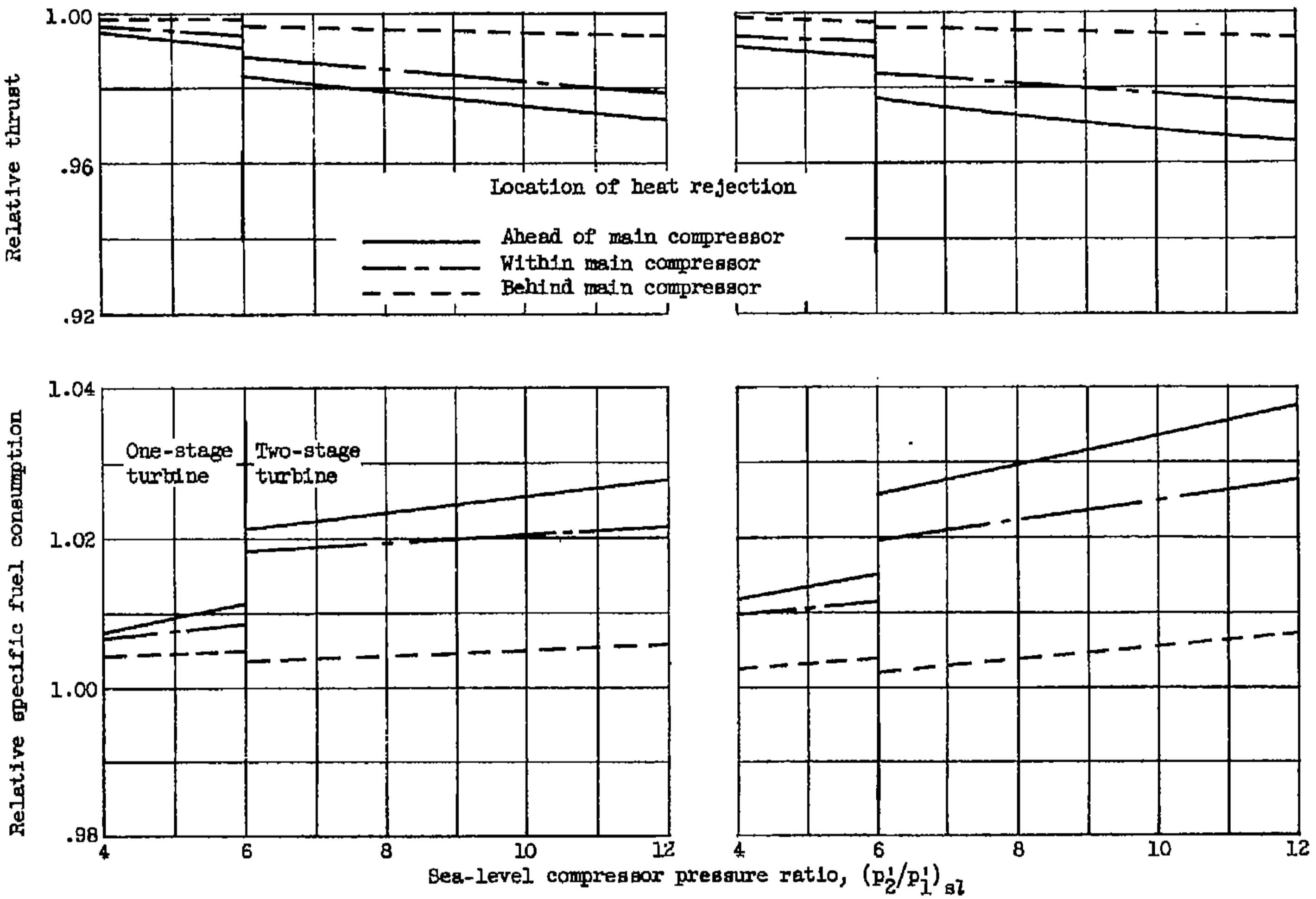
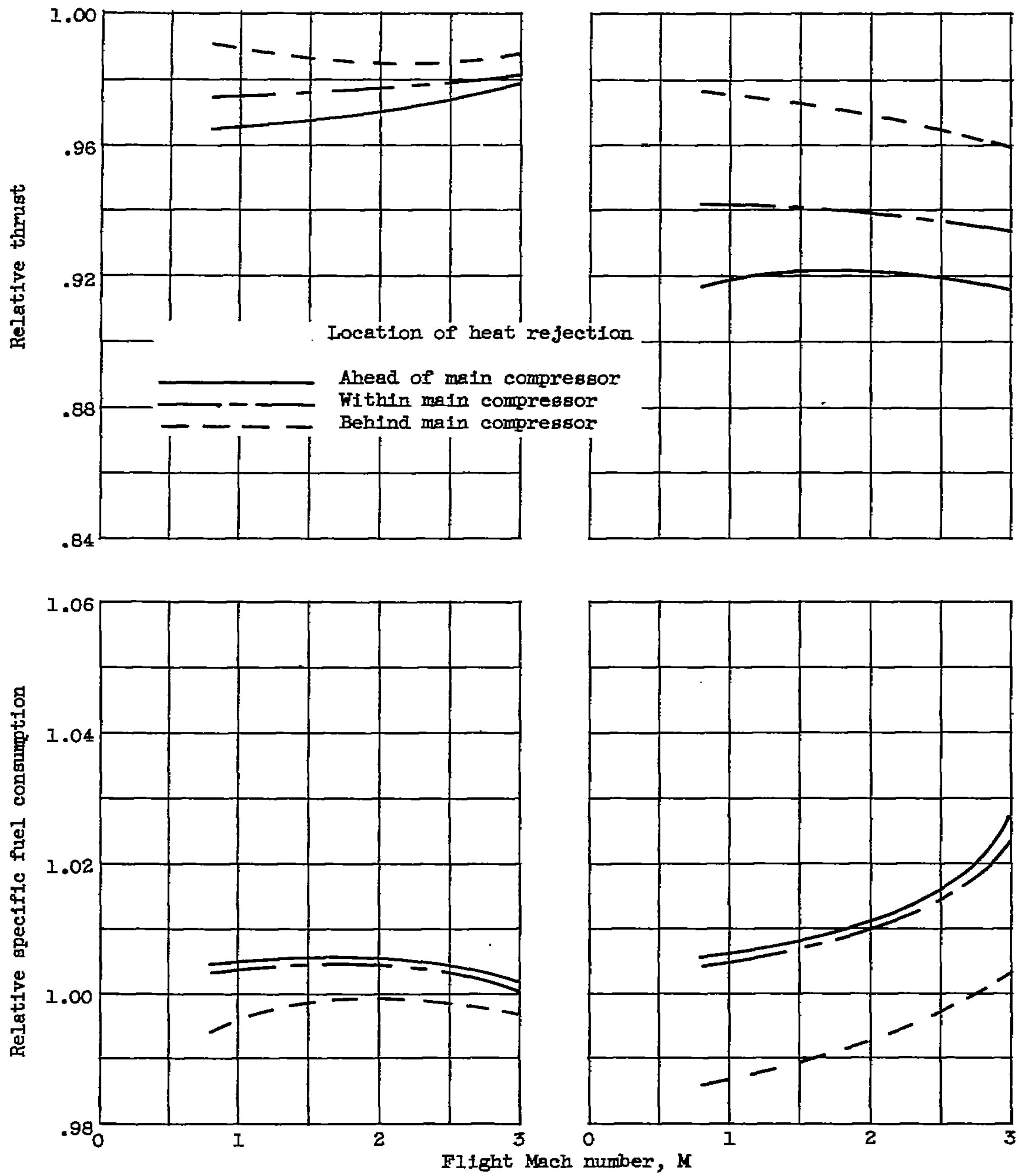


(c) Turbine-inlet temperature, 2460° R; afterburning engine; afterburner temperature, 3500° R.

(d) Turbine-inlet temperature, 2800° R; afterburning engine; afterburner temperature, 3500° R.

Figure 4. - Concluded. Relative turbojet-engine performance resulting from effects of compressor pressure ratio on liquid-cooling heat-rejection rates. Flight Mach number, 2.0; altitude, 50,000 feet; sea-level static compressor equivalent weight flow, 35 pounds per second per square foot.



(a) One-stage turbine; compressor pressure ratio, 6.0.

(b) Two-stage turbine; compressor pressure ratio, 12.0.

Figure 5. - Relative performance of nonafterburning turbojet engine resulting from effects of flight Mach number on liquid-cooling heat-rejection rates. Turbine-inlet temperature, 2460° R; altitude, 50,000 feet; sea-level static compressor equivalent weight flow, 35 pounds per second per square foot.

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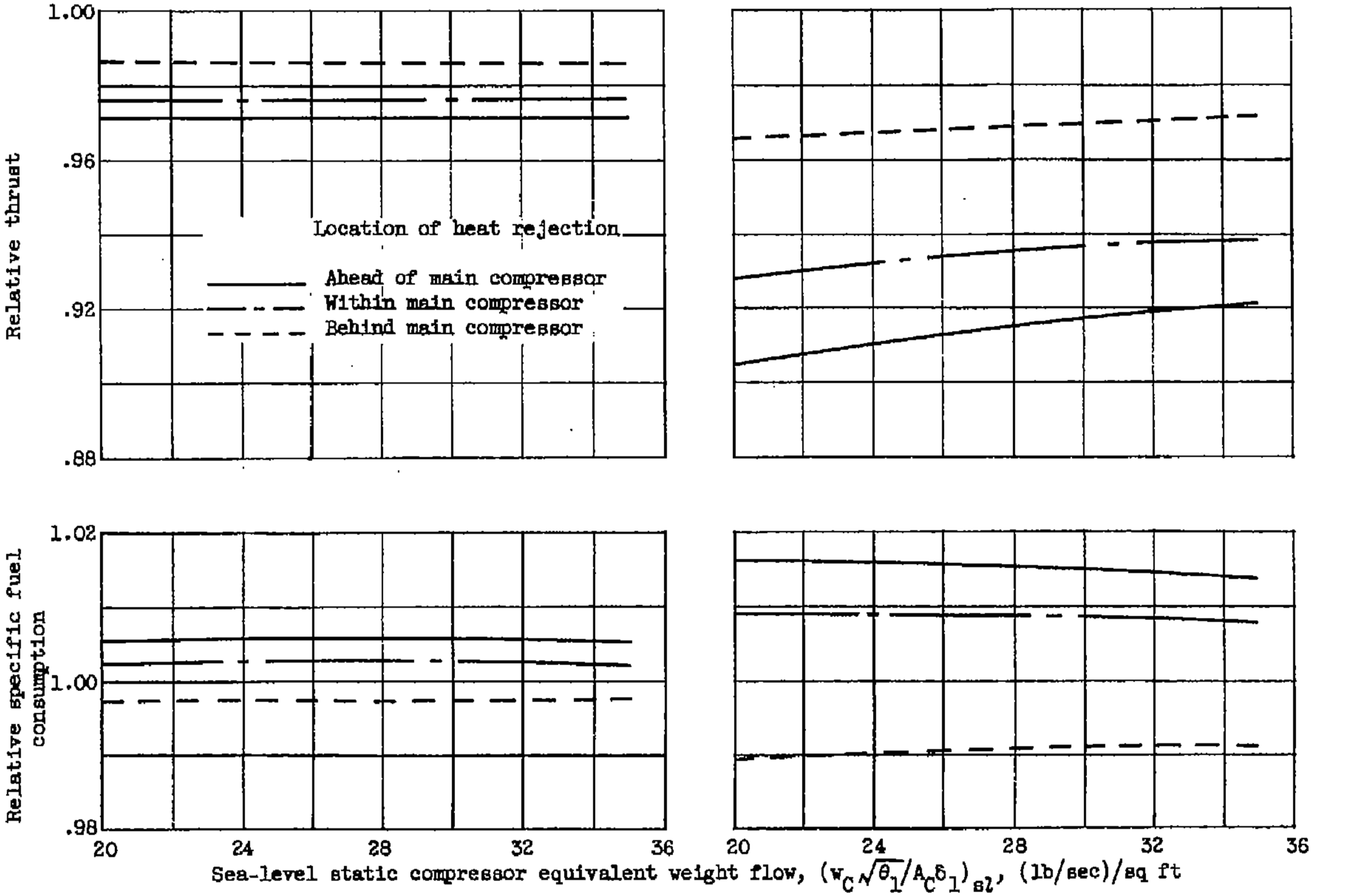
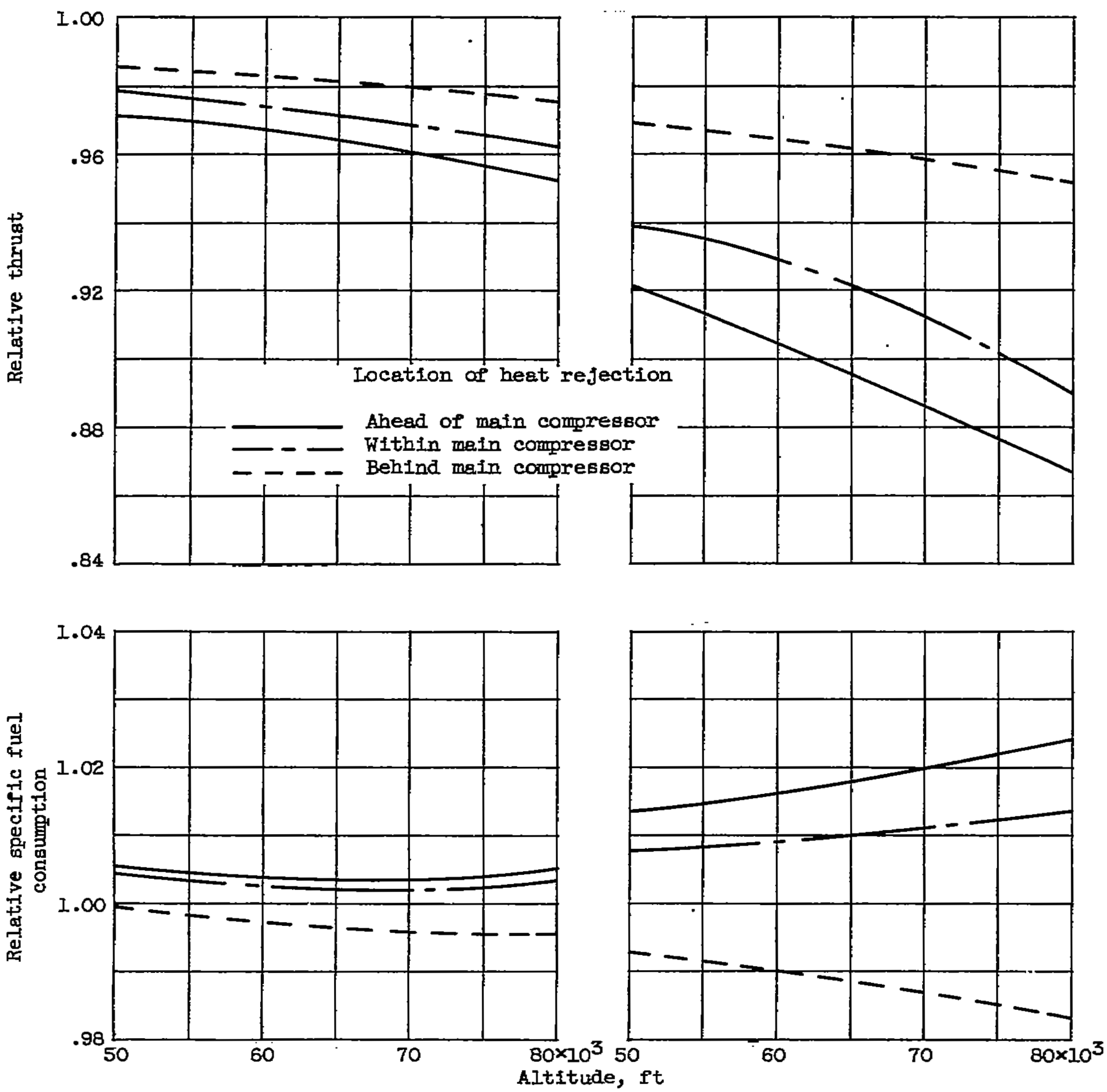


(a) One-stage turbine; compressor pressure ratio, 6.0. (b) Two-stage turbine; compressor pressure ratio, 12.0.

Figure 6. - Relative performance of nonafterburning turbojet engine resulting from effects of compressor equivalent weight flow on liquid-cooling heat-rejection rates. Turbine-inlet temperature, 2460° R; altitude, 50,000 feet; flight Mach number, 2.

4007

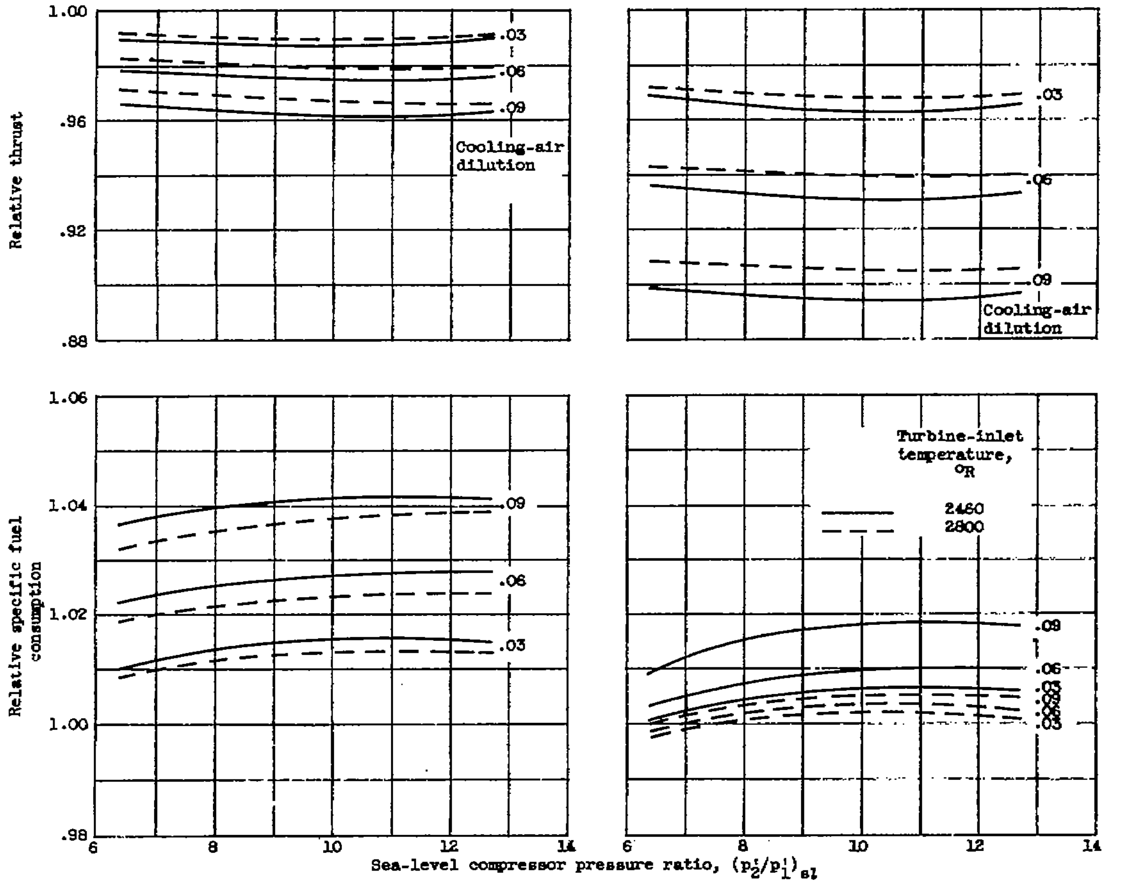


(a) One-stage turbine; compressor pressure ratio, 6.0.

(b) Two-stage turbine; compressor pressure ratio, 12.0.

Figure 7. - Relative performance of nonafterburning turbojet engine resulting from effects of flight altitude on liquid-cooling heat-rejection rates. Turbine-inlet temperature, 2460° R; flight Mach number, 2; sea-level static compressor equivalent weight flow, 35 pounds per second per square foot.

4007

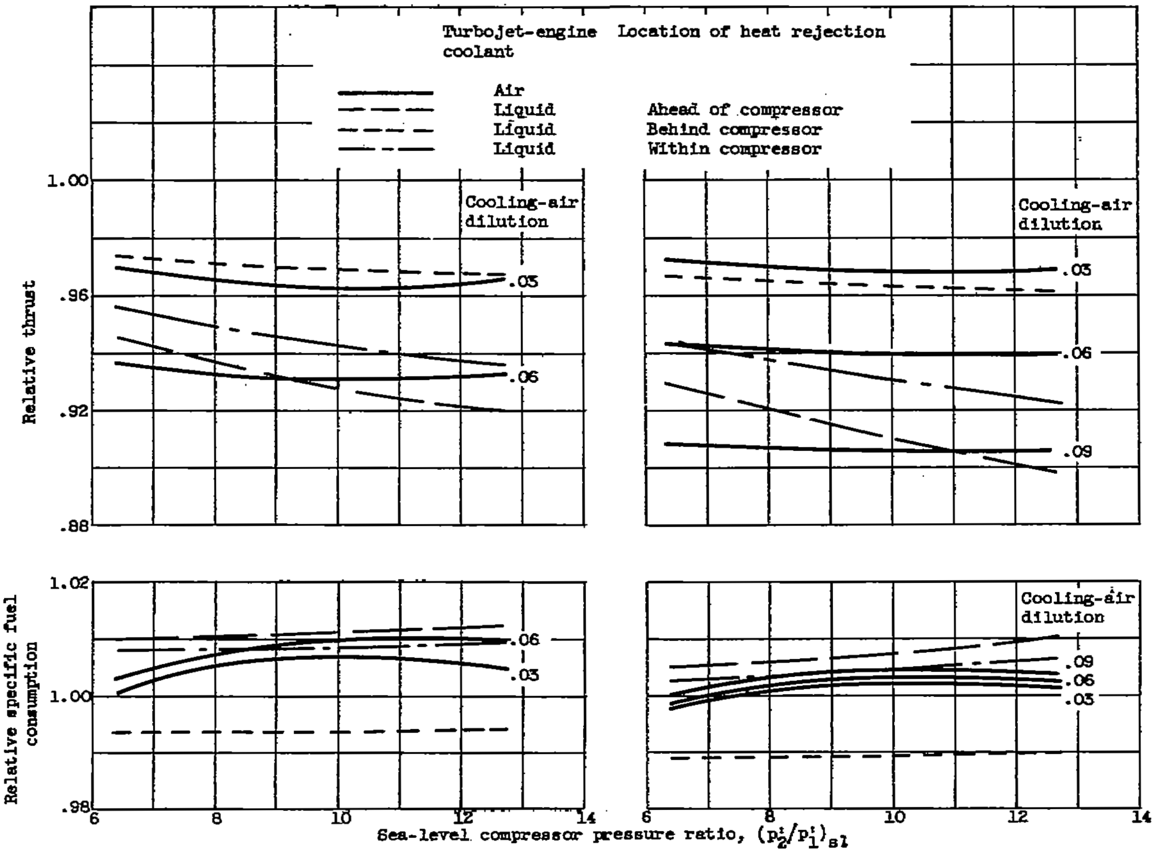


(a) Afterburning turbojet engine.

(b) Nonafterburning turbojet engine.

Figure 8. - Effect of air-cooling on turbojet-engine performance for a range of compressor pressure ratios. Flight Mach number, 2; altitude, 50,000 feet.

4007



(a) Turbine-inlet temperature, 2460° R.

(b) Turbine-inlet temperature, 2800° R.

Figure 9. - Comparison of performance of nonafterburning air- and liquid-cooled turbojet engines. Flight Mach number, 2.0; altitude, 50,000 feet; sea-level static compressor equivalent weight flow, 35 pounds per second per square foot.

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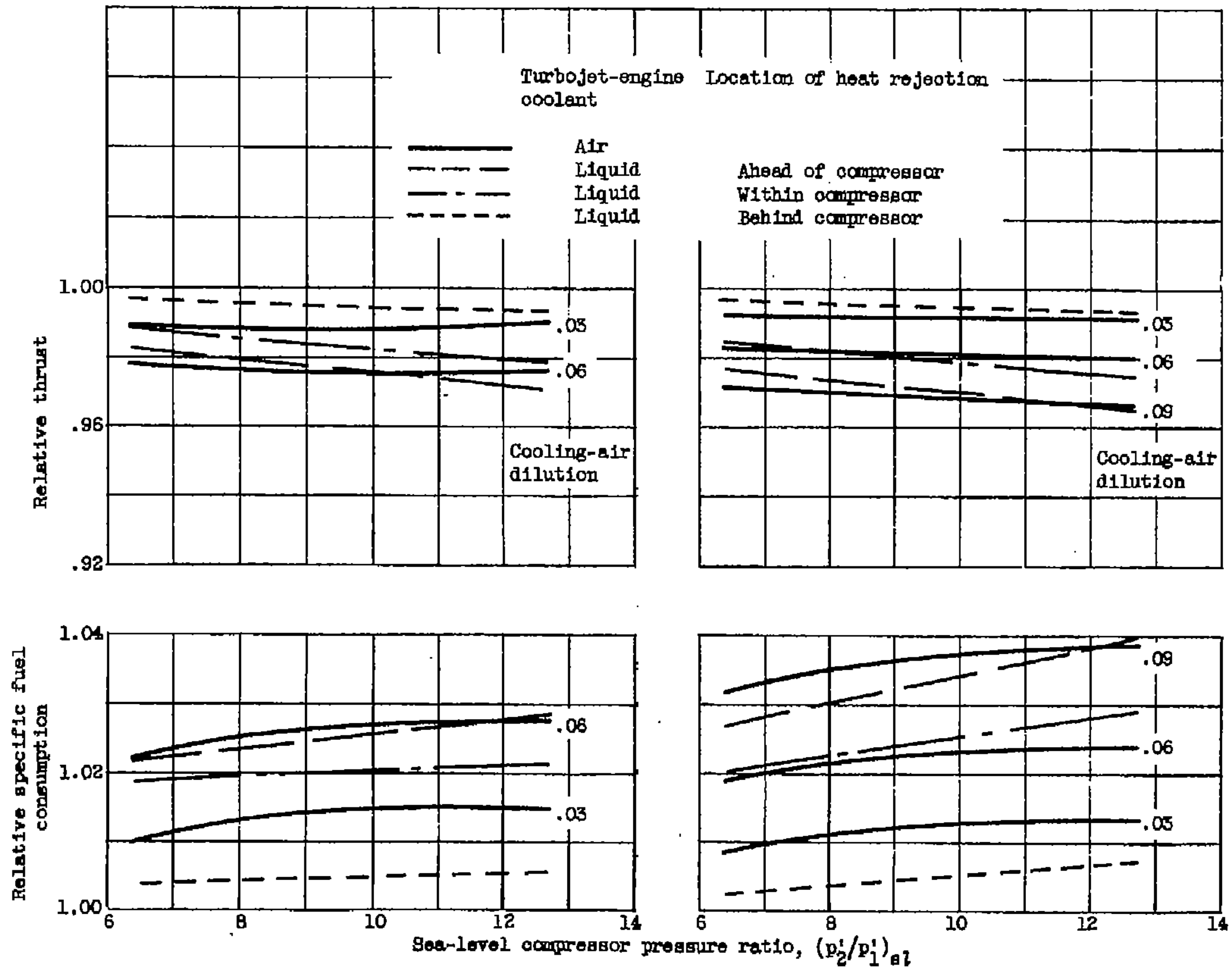

(a) Turbine-inlet temperature, 2460° R.

(b) Turbine-inlet temperature, 2800° R.

Figure 10. - Comparison of performance of afterburning air- and liquid-cooled turbojet engines. Flight Mach number, 2.0; altitude, 50,000 feet; sea-level static compressor equivalent weight flow, 35 pounds per second per square foot.

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