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RESEARCH MEMORANDUM

ALTITUDE INVESTIGATION OF THRUST AUGMENTATION USING WATER-ALCOHOL INJECTION INTO THE COMBUSTION CHAMBERS OF AN AXIAL-FLOW TURBOJET ENGINE

By E. T. Jansen and P. E. Renas

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

February 12, 1953

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ALTITUDE INVESTIGATION OF THRUST AUGMENTATION USING WATER-ALCOHOL INJECTION INTO THE COMBUSTION CHAMBERS OF AN AXIAL-FLOW TURBOJET ENGINE

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2794

SUMMARY

An investigation was conducted in an altitude test chamber at the NACA Lewis laboratory to determine the altitude performance of an axial-flow turbojet engine using water-alcohol injection into the combustion chambers. Data were obtained over a range of altitudes from 30,000 to 50,000 feet at a flight Mach number of 0.80. At each flight condition the liquid-air ratios were varied between 0 and about 0.10 for each of four conical exhaust nozzles.

Net thrust augmentation ratios of 1.25, 1.23, and 1.21 were obtained at a liquid-air ratio of 0.10 for altitudes of 30,000, 40,000, and 50,000 feet, respectively. In order to have this thrust augmentation, which was obtained at limiting turbine-outlet temperature, a continuously variable-area exhaust nozzle would be required. Operation with only the standard exhaust nozzle and with a liquid-air ratio of 0.10 resulted in a variance of the augmented net thrust from 1.14 to 1.195 as the altitude was increased from 30,000 to 50,000 feet. The use of 10,000 gallons of water-alcohol mixture in 7.5 hours of engine running time resulted in no noticeable engine structural deterioration. Engine operation was smooth at all times regardless of the injection flow rates.

INTRODUCTION

The increasing need for higher thrust of short time duration so as to improve the take-off conditions and certain flight maneuvers places a new demand on the aircraft engine industry. The requirement of higher thrust for aircraft still in the design stage may often be satisfied by the turbojet engine fitted with an afterburner. On the other hand, the installation of afterburners in existing aircraft to improve flight performance is often an impossibility because of nacelle or fuselage space limitations. The need is therefore apparent for a more simplified method of augmenting turbojet engine thrust which will be applicable to existing aircraft. Theoretical studies of means of augmenting turbojet engine thrust (refs. 1 to 3) have shown that in addition to afterburning, liquid injection into the compressor, liquid injection into the combustion chamber, and separate burning of fuel in air bled from the compressor each provides promising increases in thrust. Of the systems mentioned, liquid injection into the combustion chamber has the advantage of simplicity with regard to existing aircraft engine installations.

Experimental investigations of liquid injection augmentation using a water-alcohol mixture at sea-level static conditions have been reported in references 4 to 7 for compressor-inlet, interstage, combustion-chamber, and combination injection systems and at altitude in reference 8 for compressor-inlet and interstage injection. These data show appreciable thrust gains, with reference 7 showing a thrust augmentation at sea level of 39.5 percent at a ratio of liquid flow to air flow of 0.148 with a combination of compressor sixth-stage and combustion-chamber injection. For combustion-chamber injection alone, the augmented thrust was 24.5 percent at a liquid-air ratio of 0.10. Liquid flow is defined as the total liquid injected into the engine excluding engine fuel flow. In the present investigation, because simplicity of installation and control are of paramount importance, the combustion-chamber injection system was chosen for investigation at altitude conditions.

The investigation reported herein, which was conducted in an NACA Lewis altitude test chamber, covered a range of altitudes from 30,000 to 50,000 feet at a flight Mach number of 0.8. Liquid-air ratios, the liquid being a water-alcohol mixture, were varied between 0 and about 0.10 for each of four conical exhaust nozzles having areas varying from the normal area to 93.7 percent of the normal area. The performance of the over-all engine along with component performance was obtained. From these data, performance cross plots of the engine operating at limiting turbine-outlet temperature, which represents the maximum thrust augmentation obtainable with combustion-chamber injection, were drawn. In addition, performance cross plots were obtained which represent practical engine configuration installations for existing aircraft. A comparison was also made between altitude and sea-level performance obtained by this method of thrust augmentation.

APPARATUS

Engine and Installation

The current production-model axial-flow turbojet engine used in this investigation has a twelve-stage compressor, eight tubular combustion chambers equipped by the manufacturer for water-alcohol injection, and a single-stage turbine. The engine has a static sea-level thrust rating of 6060 pounds at the rated speed of 7950 rpm and a turbine-outlet temperature of 1245° F based on the standard installation thermocouple harness.

The engine was installed in an altitude test chamber 10 feet in diameter and 60 feet in length and mounted on a thrust measuring bed as shown in figure 1. The thrust measuring bed was hung from four pendulum-type supports using flexure plates at each end of the supports. The thrust force was transmitted through a bell crank and lever system to a null-type force-measuring cell. To keep the thrust-cell forces always acting in one direction, a dead-weight preload was added to the thrust system. A front bulkhead, which incorporated a labyrinth seal around the forward end of the engine, separated the inlet and exhaust sections of the chamber and provided freedom of movement of the engine in an axial direction. A rear bulkhead was installed to act as a radiation shield and to prevent recirculation of the hot exhaust gases about the engine.

Liquid-Injection System

The liquid-injection nozzles used in this investigation (fig. 2(a)) were standard equipment for the engine and designed for sea-level conditions. The nozzles were located in the engine combustion chambers and the injection system consisted of four nozzles equally spaced around the periphery of each combustion chamber. Tips of the liquid-injection nozzles were located approximately flush with the combustion-chamber liner and injected the liquid into the primary zone of the combustion chamber normal to the direction of gas flow. The liquid spray, which was in the form of a hollow cone, provided a reasonable atomization of the injected liquid. The axial location of the injection nozzles is shown in figure 2(b).

Fuel and Liquid Mixture

Fuel supplied to the engine during this investigation conformed to specification MIL-F-5624A grade JP-4. The liquid injection mixture consisted of 30-percent MIL-A-6091 alcohol and 70-percent water by unmixed volume. The fuel and liquid flows were measured with calibrated rotameters, except at the high liquid flows where a calibrated orifice was used.

Instrumentation

The location of the instrumentation stations before and after each of the principal components of the engine is shown in figure 1. The 24 total-pressure tubes at station 1 were located at the centers of 24 equal areas and the 18 total-pressure tubes at station 9 were located 3 on each of 6 equal areas. The thermocouples at stations 1, 3, 4, 5, and 9 and the total-pressure tubes at stations 3, 4, and 5 were located on approximately equal spacings. The altitude pressure surrounding the jet nozzle was measured by four lip static probes located in the exhaust portion of the chamber. The air flow through the engine was measured at station 1, engine-inlet, and such air flow was corrected for any leakages existing in the compressor.

PROCEDURE

2794

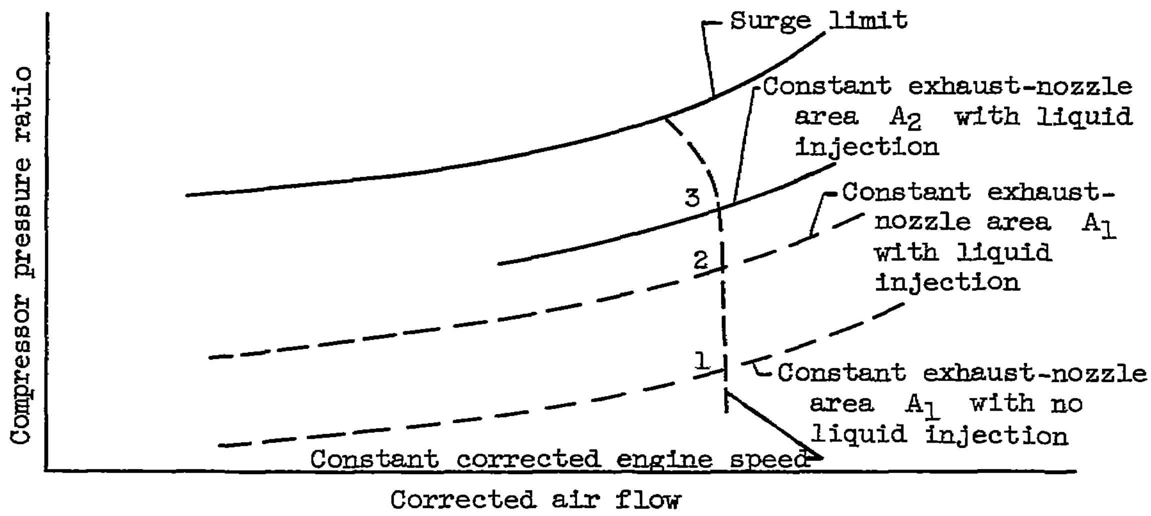
The performance data presented herein were obtained at simulated altitudes of 30,000, 40,000, and 50,000 feet at a flight Mach number of 0.8. The engine-inlet total pressure and temperature and the exhaust static pressure were held constant for each flight condition while data were obtained for various fixed-area nozzles over a range of water-alcohol injection rates with the engine operating at rated engine speed. The nozzle areas used in this investigation correspond to the standard area and 97.0, 96.2, and 93.7 percent of the standard area.

The range of liquid-flow rates was from zero to the flow which gave a liquid-air ratio of approximately 0.10. This injection rate was set by the manufacturer as an approximate maximum for safe operation due to the damaging effect of water hitting hot metal surfaces at the combustor outlet and turbine inlet.

RESULTS AND DISCUSSION

The principles of various types of liquid-injection systems used for obtaining thrust augmentation are discussed in reference 1. However, in order to facilitate the interpretation of the data reported herein, the principles involved in thrust augmentation by liquid injection into the combustion chambers are presented in the following paragraphs.

To augment the thrust of a turbojet engine, it is necessary to increase the exhaust-gas velocity or the mass flow. Liquid injection serves to increase both these quantities. With combustion-chamber injection, the mass flow through the turbine is increased and the turbine-discharge temperature is decreased at a constant engine speed and exhaust-nozzle area. Because the turbine nozzle is choked and the decrease in turbine temperature is insufficient to compensate for this increase in mass flow, the turbine-inlet total pressure must be raised to satisfy flow continuity. The required increase in turbine-inlet total pressure reflects back to the compressor outlet, resulting in an increase in compressor pressure ratio. The effect on the compressor is much the same as throttling the flow at the discharge; the corresponding change in operating point is noted on the following sketch of the compressor characteristic curve as a movement from point 1 to point 2.



If an exhaust nozzle of smaller area were used to maintain limiting turbine-outlet temperature, a similar shift in operating point would be encountered and the final compressor pressure ratio would have a correspondingly higher value for the same engine speed and liquid-injection rate (point 2 to point 3) than the larger nozzle would have. This increased compressor pressure ratio, being greater than the change in turbine pressure ratio, results in an increase in tail-pipe pressure ratio. The combined effect of increased tail-pipe pressure ratio and increased mass flow produce an increase in thrust. Further examination of the compressor curve indicates that the amount of liquid that can be injected or the allowable reduction in nozzle area may be limited by compressor surge.

Basic Performance Data

The data obtained during this investigation are presented as a ratio of augmented to normal engine performance over a range of liquid-air ratios for four fixed-area exhaust nozzles. The normal engine performance is the unaugmented performance which the engine delivers at each flight condition. Normal engine performance would be obtained with the engine equipped with a fixed-area exhaust nozzle properly sized at sea-level static conditions to give the limiting turbine-outlet temperature (1250° F) at rated engine speed. Therefore, at altitude, because of a decrease in component efficiencies, the rated turbine-outlet temperature, and consequently maximum unaugmented performance, was encountered before reaching rated engine speed. The augmented performance was obtained at rated engine speed and varying turbine-outlet temperature depending on the rate of liquid injection.

The effect of liquid-air ratio on pertinent engine and component performance variables for each of four fixed-area exhaust nozzles at altitudes of 30,000, 40,000, and 50,000 feet is shown in figures 3 to 10. The increase in compressor pressure ratio with increasing liquid-air ratio is shown in figure 3. For each nozzle size, the turbine pressure ratio (fig. 4) remained nearly constant over the range of liquid-air ratios, so that the exhaust-nozzle pressure increase was approximately proportional to the compressor pressure rise. With the 100-percent nozzle area and at zero liquid-air ratio, the ratio of compressor pressure ratios had a tendency to increase above unity and the ratio of 2794
turbine pressure ratios had a tendency to decrease below unity as the altitude was increased, because rated speed operation of the engine at these altitudes occurred above limiting exhaust-gas temperature. Other performance parameter ratios also tend to show a rise above a value of 1.0 because of this higher-than-rated exhaust-gas temperature operation at zero liquid-air ratio.

The air flow remained constant over the range of nozzle areas and liquid-air ratios investigated (fig. 5), indicating that the compressor was operating in the region where the constant engine-speed line was vertical on the compressor map. The total mass flow therefore increased linearly with liquid-air ratio (fig. 6). The combustion efficiency, based on total fuel and alcohol flow, decreased only slightly as the liquid-air ratio was increased (fig. 7), indicating that most of the alcohol was burning. Consequently, fuel flow (fig. 8) decreased almost linearly as liquid-air ratio was increased. The exhaust-gas temperature decreased as the liquid-air ratio was increased (fig. 9). However, the decrease in exhaust-gas temperature was more than offset by the increase in exhaust-nozzle pressure and mass flow with a resultant increase in net thrust (fig. 10). Inspection of the data indicate that all the parameters discussed exhibit the same trends for each of the three altitudes investigated.

Specific Modes of Operation

On the basis of the preceding performance characteristics, performance can be determined for several different modes of operation. The following discussion will describe the performance of three such modes of operation which would require the following types of exhaust nozzles - a standard fixed-area nozzle, a continuously variable-area nozzle, and a two-position nozzle. In addition to the exhaust nozzle and control, each mode of operation would require liquid tanks, piping, a pump, and a regulator on the engine fuel flow as well as the liquid flow which would maintain rated engine speed as the liquid-injection rate was varied. Operation will be considered at the limiting liquid-air ratio of 0.10, which, as mentioned previously, was set by the manufacturer as an approximate maximum for safe operation.

Standard fixed-area nozzle. - The data previously discussed for 100-percent nozzle area (figs. 3 to 10) showed the engine and component performance with a liquid-injection system installed on a standard engine with a standard fixed-area nozzle. This configuration represents the most simplified installation and could be incorporated on existing aircraft with a minimum of modifications. At a liquid-air ratio of 0.10, the net thrust gain with this configuration varied from 14.0 percent at 30,000 feet to 19.5 percent at 50,000 feet (fig. 10). At an altitude of 30,000 feet it was necessary to extrapolate the data for the
2794 100-percent nozzle-area curve from a liquid-air ratio of 0.073 to 0.10 because of a limitation on the liquid-system pumping capacity.

Continuously variable-area nozzle. - The optimum mode of operation consists of using a continuously variable-area exhaust nozzle in order to maintain limiting turbine-outlet temperature. This configuration will give the maximum thrust gain at any flight condition. The effect of liquid-air ratio on engine performance including component performance over a range of altitudes at limiting turbine-outlet temperature (1250° F) is shown in figures 11 to 14. Superimposed on these figures as dashed lines are data taken from reference 7. These reference data were obtained at static sea-level conditions using an earlier model of the same turbojet engine but operating at a limiting turbine-outlet temperature of 1275° F.

The compressor pressure ratio (fig. 11(a)) increased more rapidly and the turbine pressure ratio (fig. 11(b)) decreased less rapidly at altitude than at sea level for the higher liquid-air ratios. At a liquid-air ratio of 0.10, the over-all result is that the rise in exhaust-nozzle pressure ratio was 1 to 2 percent less at altitude than at sea level.

Combustion efficiency (fig. 12(a)) was nearly constant at altitude, whereas at sea level there was a rapid fall-off above a liquid-air ratio of 0.06. Therefore, at sea level the fuel flow (fig. 12(b)) increased above this liquid-air ratio, whereas at altitude the fuel flow decreased almost linearly with increasing liquid flow. The loss in combustion efficiency at sea level is attributed to greater penetration of the water-alcohol mixture into the combustor which tends to quench the flame, thereby requiring more fuel to maintain a constant exhaust-gas temperature. The greater penetration at sea level than at altitude resulted from the higher pressure required to inject the desired liquid flows. This decrease in combustion efficiency was also apparent at an altitude of 30,000 feet with maximum liquid injection (fig. 7).

The engine-inlet air flow (fig. 13(a)) was nearly constant for all flight conditions including sea level over the range of liquid-air ratios investigated. Therefore, the ratio of augmented to normal mass flow increased directly as the liquid-air ratio increased (fig. 13(b)).

Increasing the liquid-air ratio at a given altitude decreased the nozzle area required to maintain limiting turbine-outlet temperature (fig. 14(a)). Increasing the altitude at a given liquid-air ratio increased the required nozzle area to maintain limiting turbine temperature.

Increasing the liquid-air ratio at a given altitude increased the jet thrust (fig. 14(b)) approximately linearly. At a given liquid-air ratio, the augmented jet thrust ratio obtained at sea-level static conditions was nearly twice the amount obtained at altitude and a flight 2794
Mach number of 0.8. This result appears inconsistent because the percentage increase in engine pressure ratio and mass flow at a given liquid-air ratio was about the same irrespective of flight condition. The difference arises from the fact that the tail-pipe pressure ratio at sea-level static conditions was approximately 1.7 and at the higher flight Mach number, about 3.5. Consequently, a given percentage increase in tail-pipe pressure ratio resulted in a greater increase in jet velocity at sea-level static conditions than at a higher flight Mach number. The net thrust curve (fig. 14(c)) exhibits the same general trends as the jet thrust, and at a liquid-air ratio of 0.10, the net thrust augmentation was 1.25, 1.23, and 1.21 at altitudes of 30,000, 40,000, and 50,000 feet, respectively.

Two-position nozzle. - The third configuration considered would involve operation with a two-position variable-area nozzle at a constant liquid-air ratio of 0.10. The engine performance with possible nozzle areas for use in this type configuration is shown in figures 3 to 10. With this configuration, limiting turbine-outlet temperature operation at rated engine speed would be attained only at the designed flight condition for each nozzle position. Operation below the design altitude would be at reduced turbine-outlet temperature but at rated speed, while operation above the design altitude would be at rated temperature but at reduced speed. This loss in turbine-outlet temperature or the reduction in engine speed is more than offset by the increase in mass flow and exhaust-nozzle pressure ratio so as to result in a substantial net thrust increase over normal engine operation.

If limiting turbine-outlet temperature operation at rated engine speed were maintained for a given nozzle area by regulating the liquid-injection rate, the net thrust gain would be considerably decreased at the off-design conditions because of the decreased liquid-air ratio. It is therefore evident that this limiting temperature operation with a two-position nozzle is an impractical method of thrust augmentation.

Comparison of modes of operation. - The effect of altitude on thrust augmentation for the various nozzle configurations is shown in figure 15 at a constant liquid-air ratio of 0.10. With the continuously variable-area nozzle, limiting turbine-outlet temperature operation results in a maximum net thrust augmentation of 1.25, 1.23, and 1.21 for altitudes of 30,000, 40,000, and 50,000 feet, respectively. Using the standard fixed-area exhaust nozzle, the augmented net thrust varied from 1.14 to 1.195 as the altitude was increased from 30,000 to 50,000 feet. The decrease in thrust augmentation with the standard-area nozzle is due to operation at reduced turbine-outlet temperature. Limiting turbine-outlet temperature operation, and consequently maximum thrust augmentation, would be attained with the standard-area nozzle at an altitude of approximately 52,000 feet. The thrust augmentation ratios for nozzle areas 97.0 and 96.2 percent of the standard area are also presented in figure 15. The design altitudes for these nozzles, 97.0 and 96.2 percent of standard, are 42,500 and 37,000 feet, respectively. Therefore, the combination of a reduced area such as this with the standard area in the form of a two-position nozzle could reduce the off-design thrust penalties of the standard nozzle at the lower altitudes.

The effect of liquid-air ratio on the exhaust-nozzle inlet temperature profile for altitudes of 30,000 and 50,000 feet and the location of the standard control thermocouple for this engine is shown in figure 16. As the liquid-air ratio was increased, the negative slope of the temperature gradient from the shell to the center of the tail pipe became more severe. This change in temperature gradient becomes important if engine operation is maintained at a constant control temperature because as liquid-air ratio is increased at constant temperature as indicated by the standard control thermcouple, the average tail-pipe gas temperature decreases.

Operation Characteristics

The duration of this investigation covered 11.5 hours of engine operation with approximately 7.5 hours of this time being consumed operating the engine with liquid injection. The total quantity of water-alcohol mixture used was about 10,000 gallons. All the engine time was consumed with the engine operating either at or slightly above limiting turbine-outlet temperature with no liquid injection, or at slightly reduced turbine-outlet temperature with liquid injection. This operation therefore represents the severest condition to which any engine flight installation would be subjected, except possibly for low-altitude flight conditions requiring liquid-injection operation such as take-off or climb. The time consumed in this investigation (7.5 hr) also represents practically the total life of any one engine in a flight installation employing a liquid-injection system of this type because the augmented time would be approximately 1 percent of the total engine time. No noticeable deterioration of the engine which could be traced to the water-alcohol injection was detected. The engine had 36 hours of operating time following the last inspection, with the last 11.5 hours being the liquid-injection program. During the liquid-injection investigation, no turbine-shroud rubs were encountered. Following the investigation, the engine was disassembled from the combustors aft and the parts were inspected. This inspection revealed the existence of three cracks in the leading edges of three blades in ~~the turbine~~-nozzle diaphragm. Because of the large amount of limiting-temperature or over-temperature operation, these cracks could not definitely be attributed to liquid injection. Similar cracks in the turbine diaphragm have been experienced with normal engine operation.

During the entire program no combustion blow-outs were encountered, and the transition of engine operation from one liquid-flow rate to another flow rate was smooth. This ease of operation covered conditions ranging from minimum to maximum liquid-flow rate changes in a time interval of less than about 5 seconds. 2794

An additional factor that would tend to simplify an injection system on a turbojet engine would be the ability of the engine speed topping governor to regulate engine fuel flow as the liquid injection was varied so as to maintain rated engine speed. However, this characteristic of the engine was not determined during the present investigation.

SUMMARY OF RESULTS

The results of this investigation showed that with combustion-chamber injection of alcohol and water at a liquid-air ratio of 0.10 while the engine was operating at limiting turbine-outlet temperature and a flight Mach number of 0.80, net thrust augmentations of 1.25, 1.23, and 1.21 were possible at altitudes of 30,000, 40,000, and 50,000 feet, respectively. This thrust augmentation was available at all altitudes investigated if a continuously variable-area exhaust nozzle was used. When operating with only the standard exhaust nozzle at a liquid-air ratio of 0.10, the augmented net thrusts were 1.14 and 1.195 at altitudes of 30,000 and 50,000 feet, respectively.

The use of 10,000 gallons of water-alcohol mixture in 7.5 hours of engine running time resulted in no noticeable engine structural deterioration. Engine operation was smooth at all times regardless of the injection flow rates, and no adverse operational characteristics were encountered.

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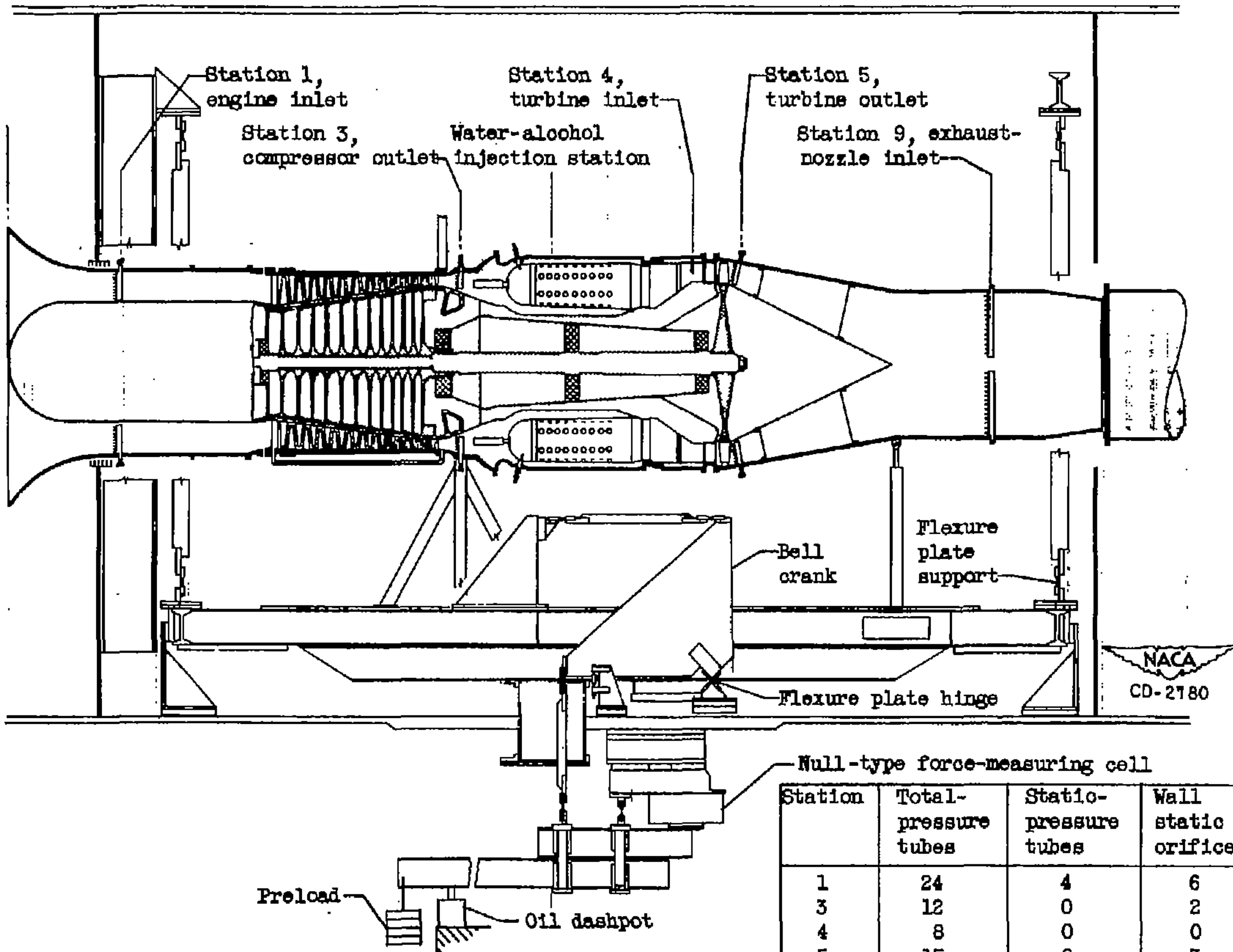
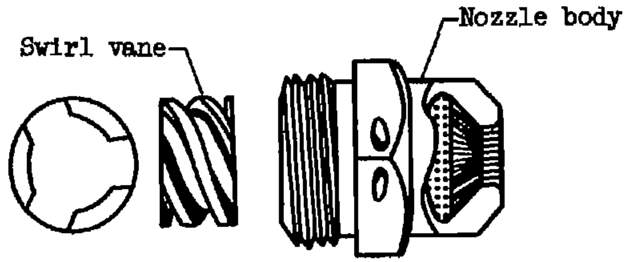
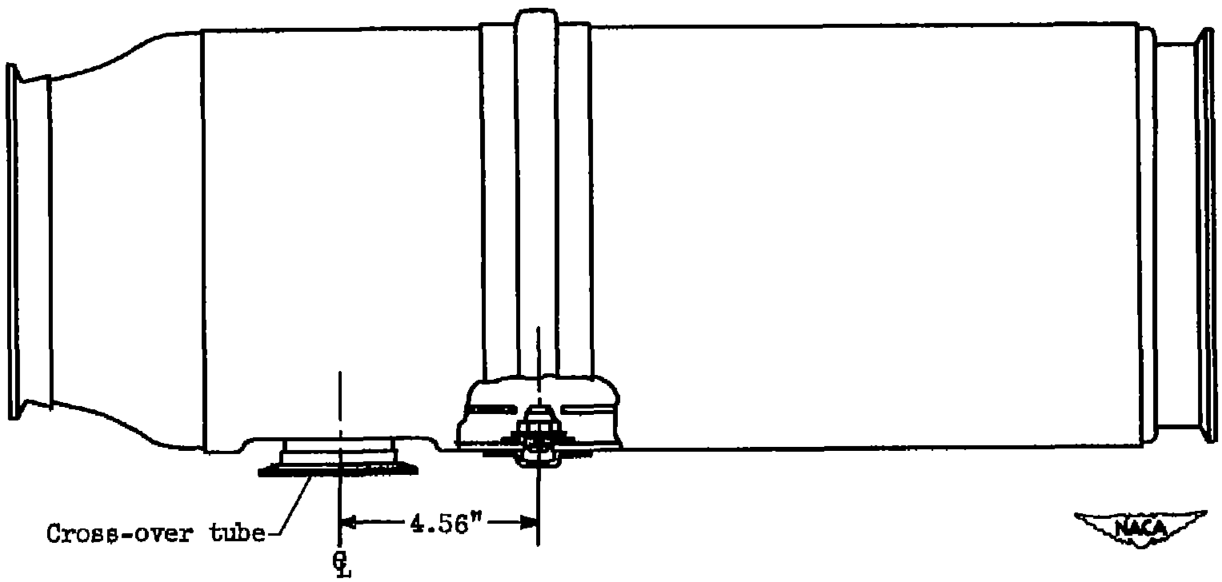


Station	Total-pressure tubes	Static-pressure tubes	Wall static orifices	Thermo-couples
1	24	4	6	12
3	12	0	2	12
4	8	0	0	8
5	15	0	3	12
9	18	0	5	25

Figure 1. - Cross section of engine mounted in altitude test chamber showing location of instrumentation stations.



(a) Water-alcohol injection nozzle.



(b) Location of nozzle in combustion chamber (four nozzles per combustion chamber equally spaced around the periphery).

Figure 2. - Sketch of water-alcohol injection nozzle and its location in combustion chamber.

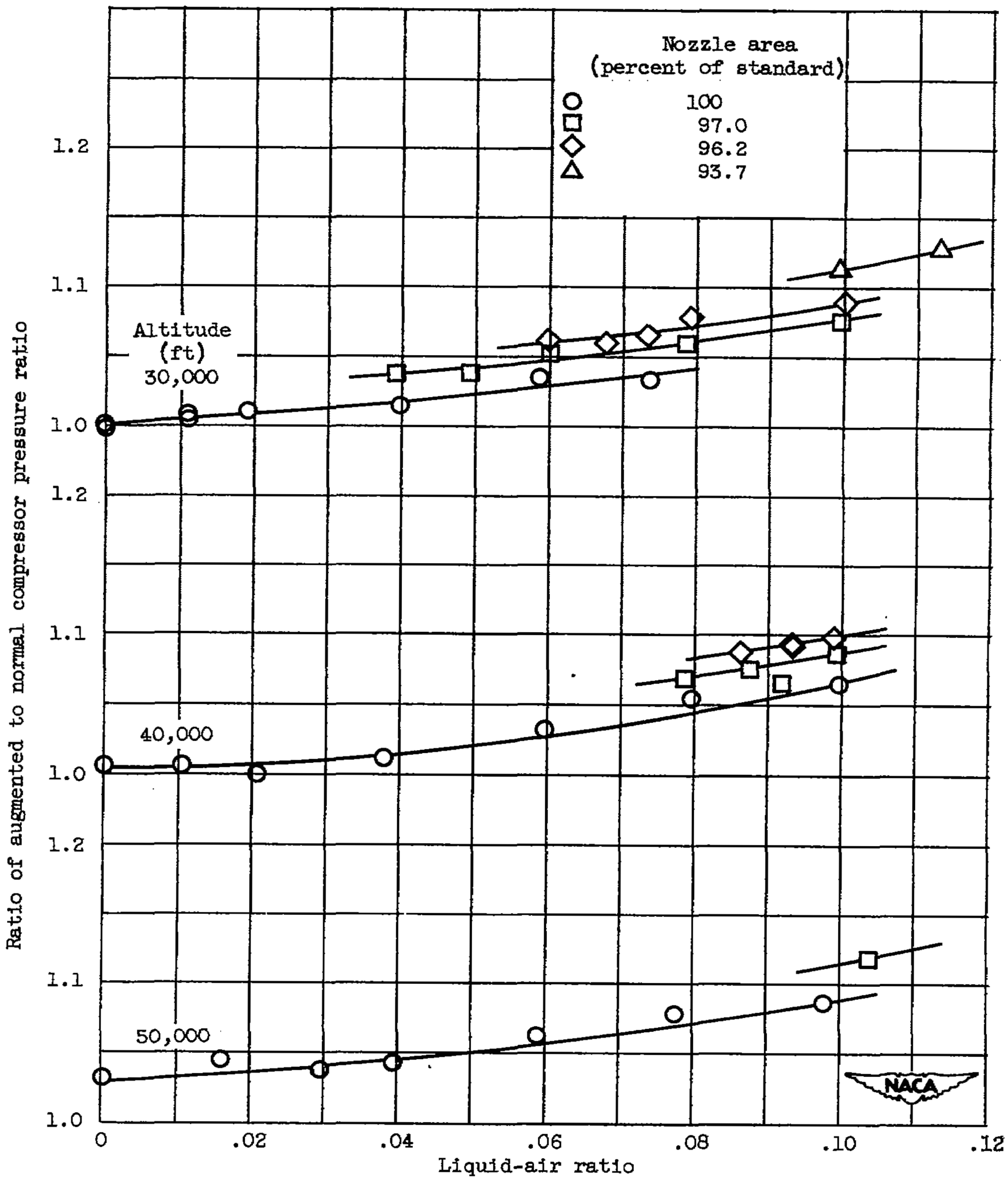


Figure 3. - Variation of ratio of augmented to normal compressor pressure ratio with liquid-air ratio. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature; flight Mach number, 0.80.

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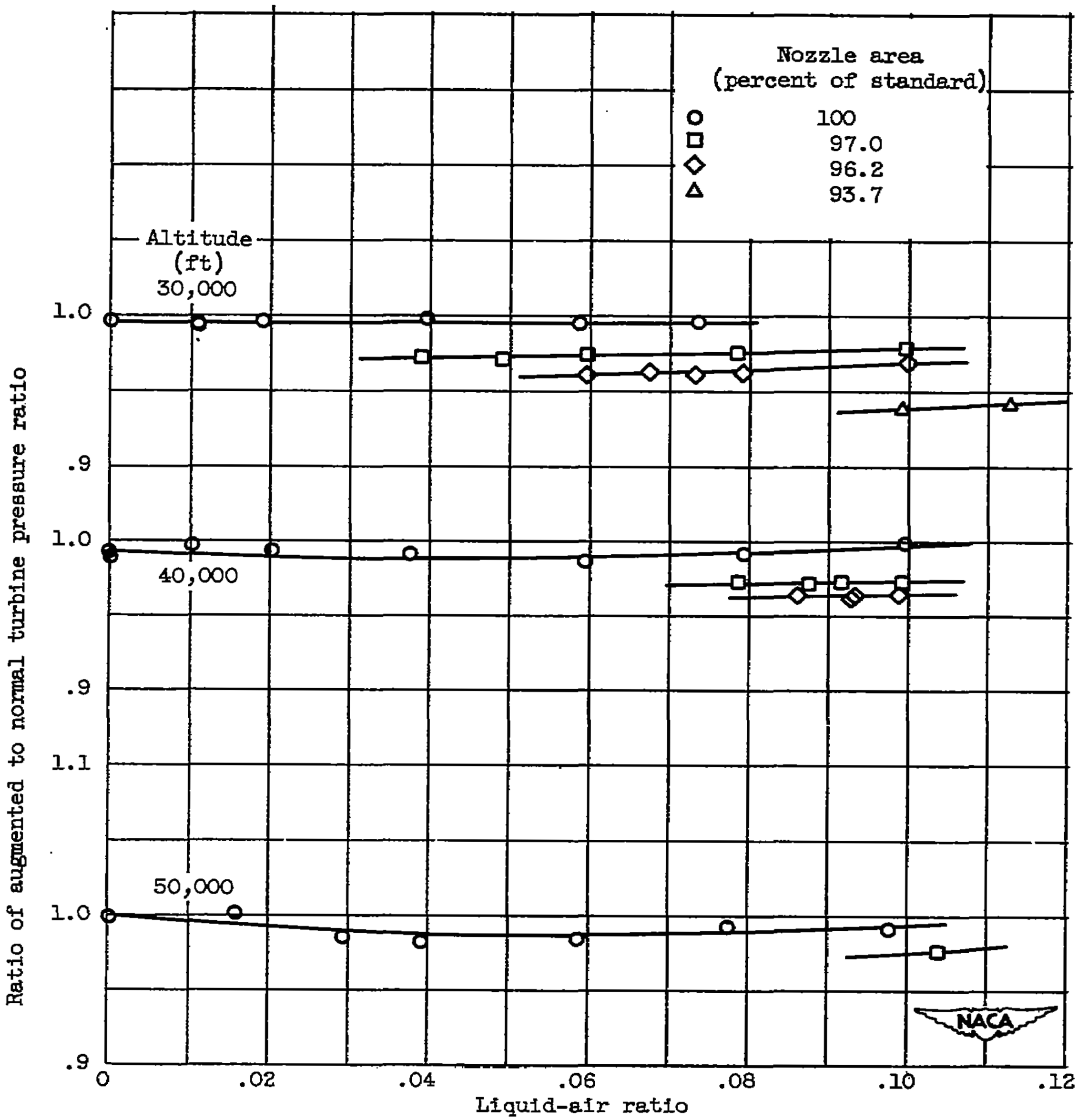

Figure 4. - Variation of ratio of augmented to normal turbine pressure ratio with liquid-air ratio. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature; flight Mach number, 0.80.

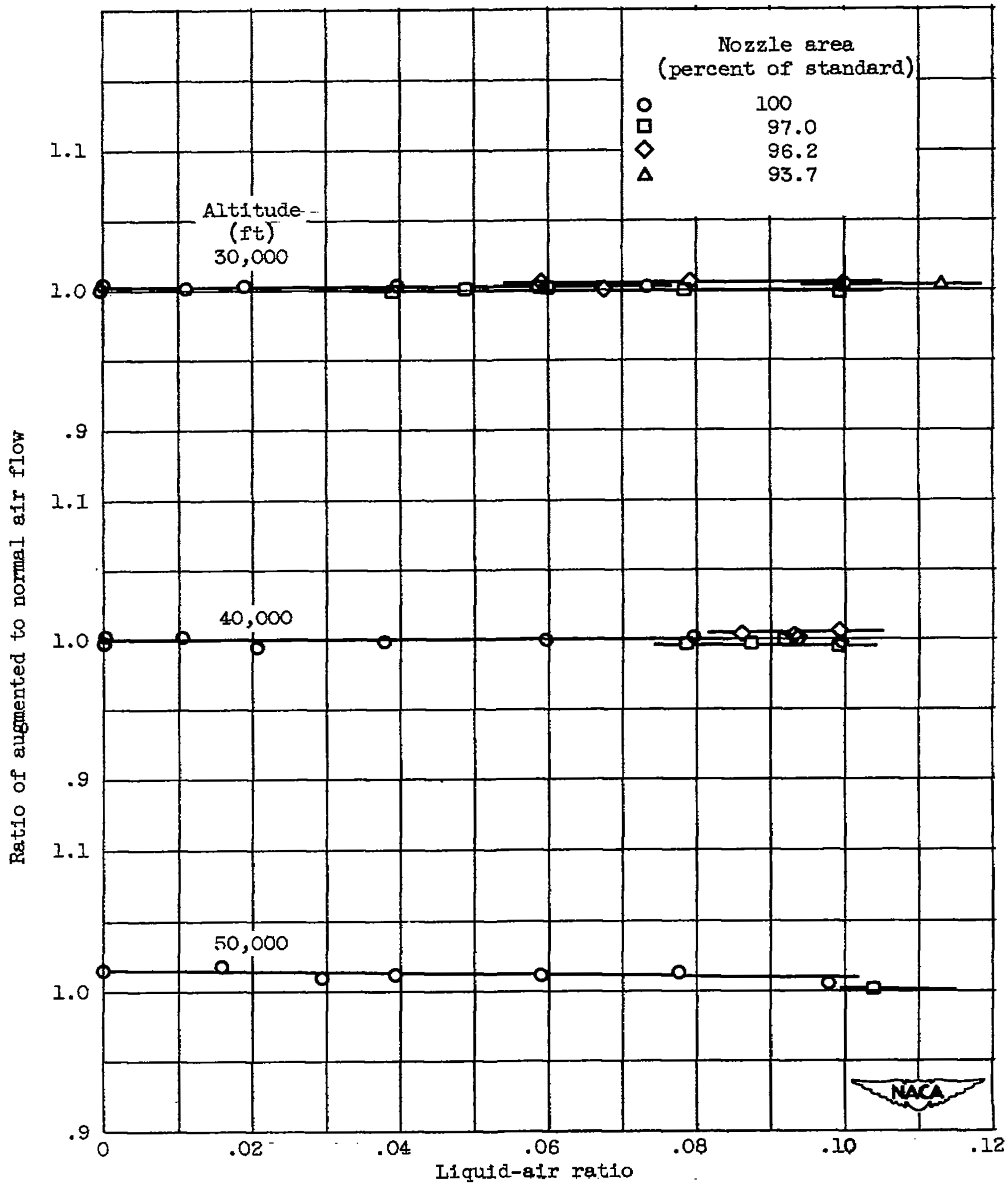

Figure 5. - Variation of ratio of augmented to normal air flow with liquid-air ratio. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature; flight Mach number, 0.80.

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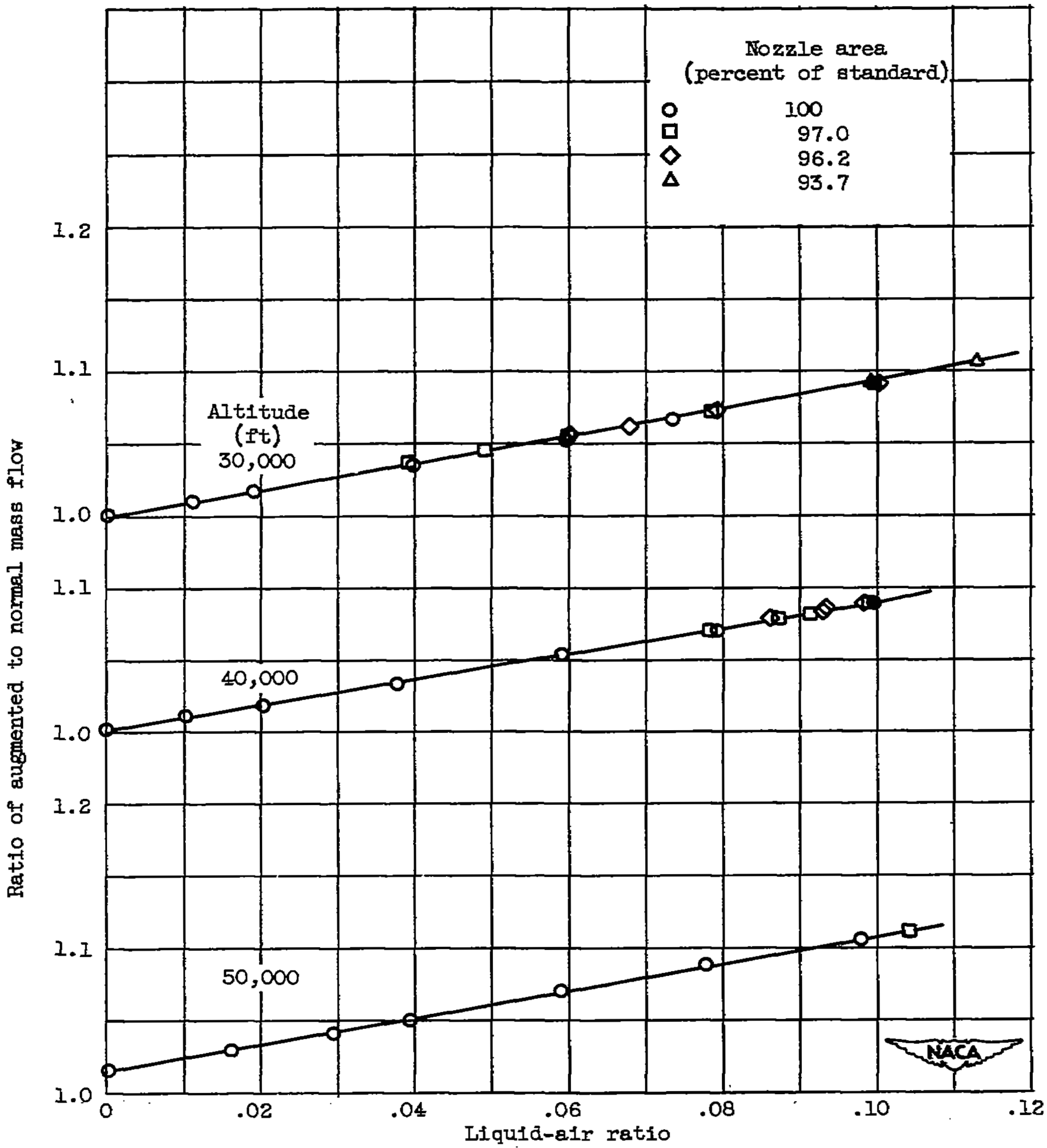

Figure 6. - Variation of ratio of augmented to normal mass flow with liquid-air ratio. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature; flight Mach number, 0.80.

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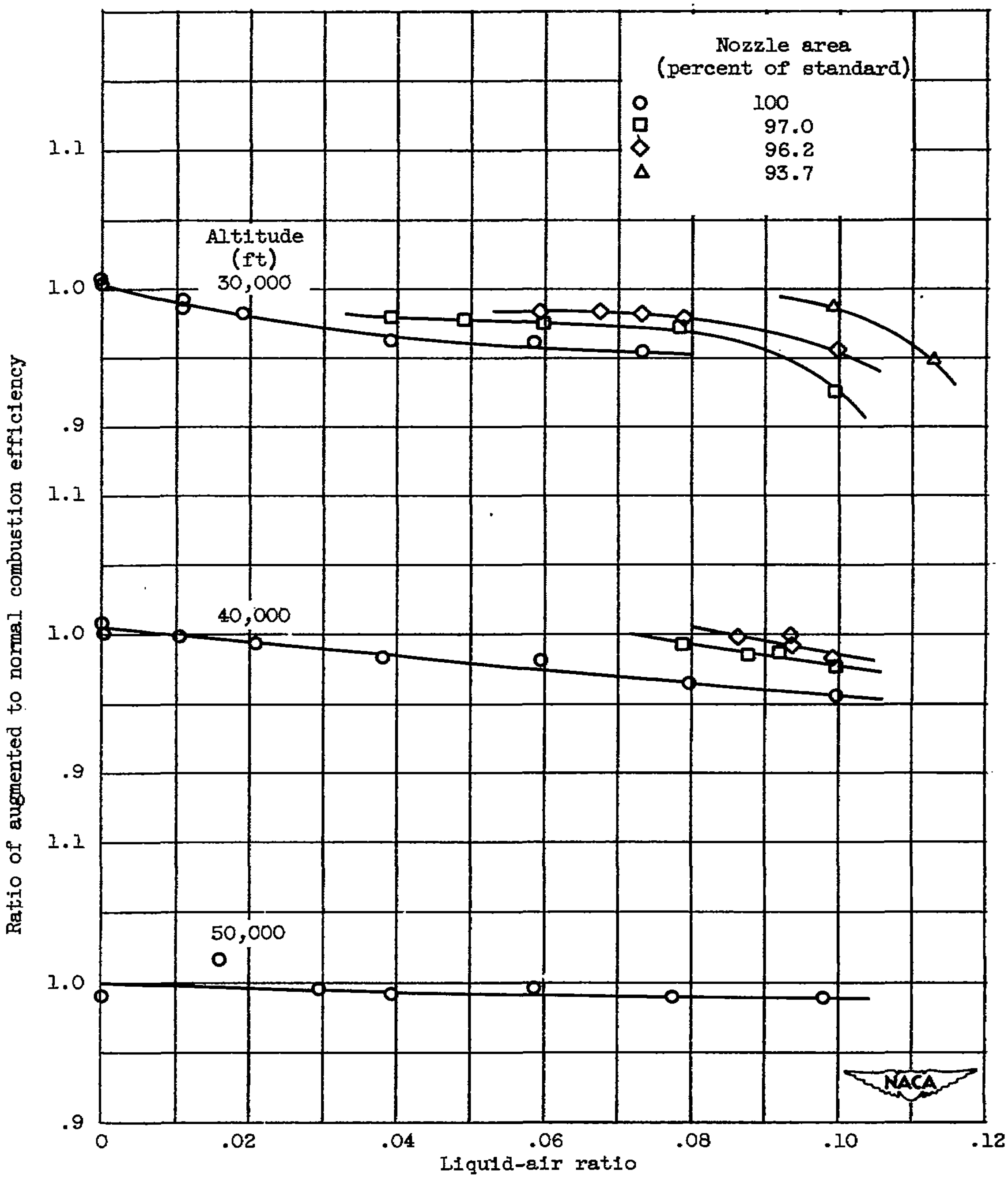

Figure 7. - Variation of ratio of augmented to normal combustion efficiency with liquid-air ratio. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature; flight Mach number, 0.80.

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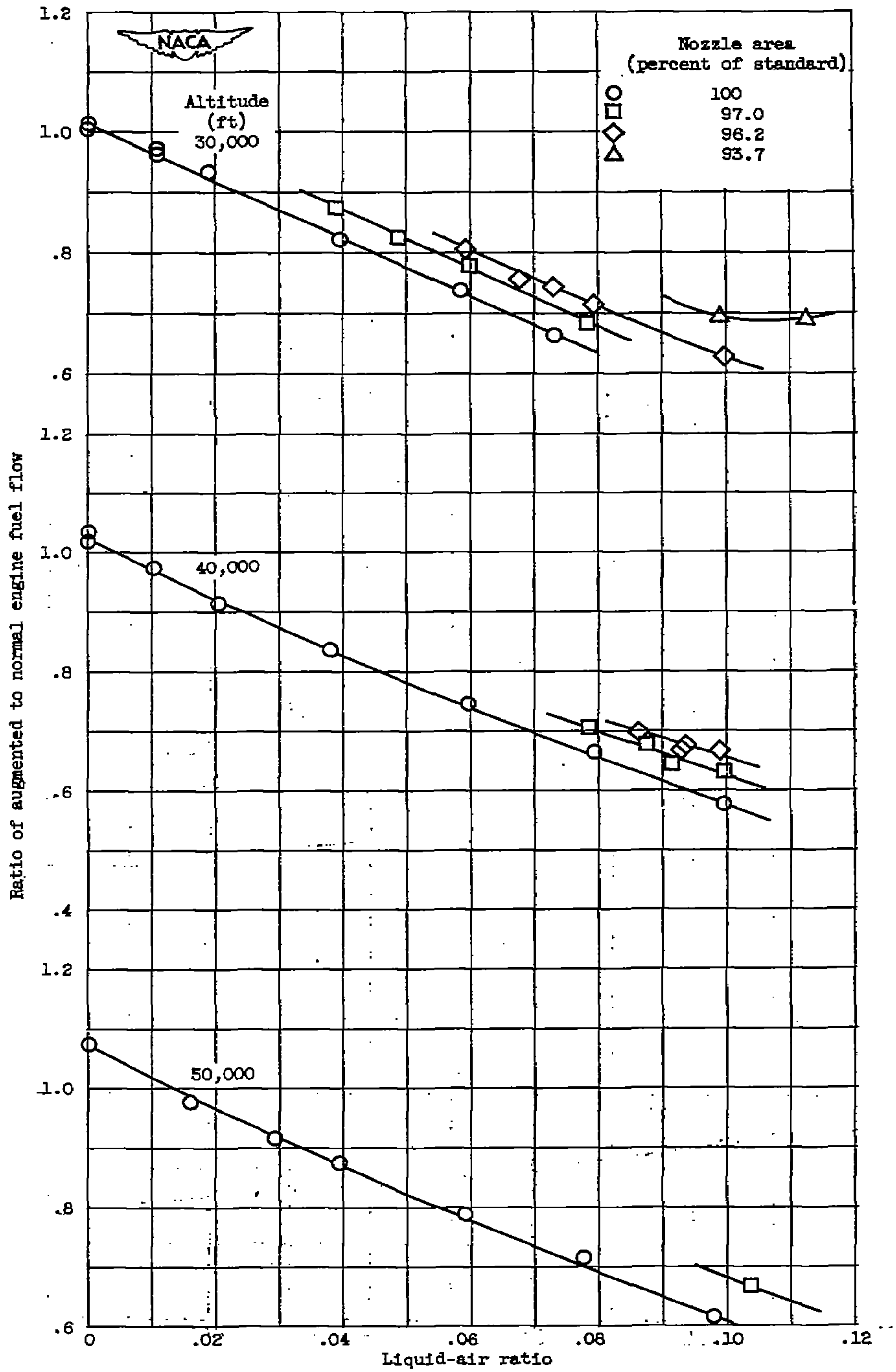

Figure 8. - Variation of ratio of augmented to normal engine fuel flow with liquid-air ratio. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature; flight Mach number, 0.80.

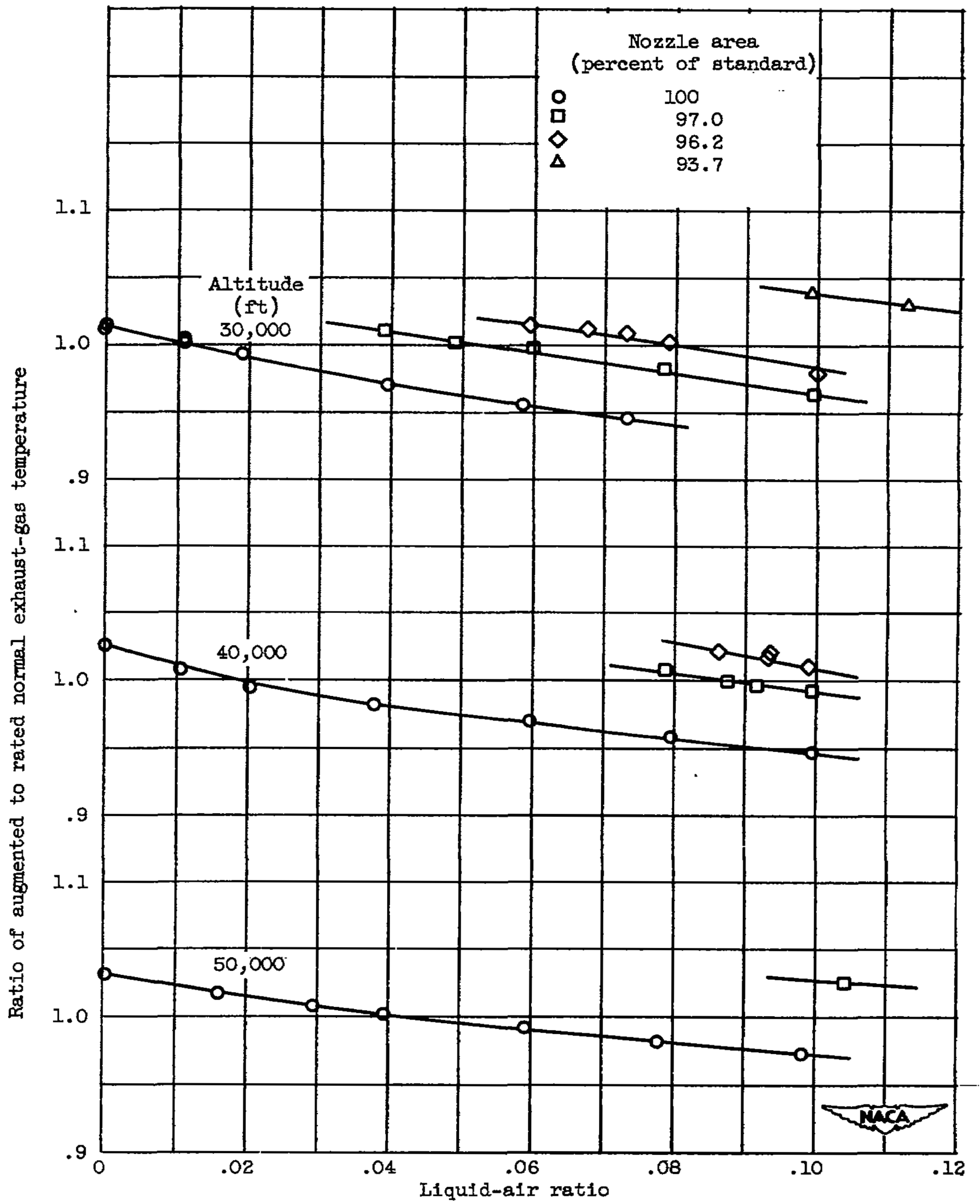

Figure 9. - Variation of ratio of augmented to rated normal exhaust-gas temperature with liquid-air ratio. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature; flight Mach number, 0.80.

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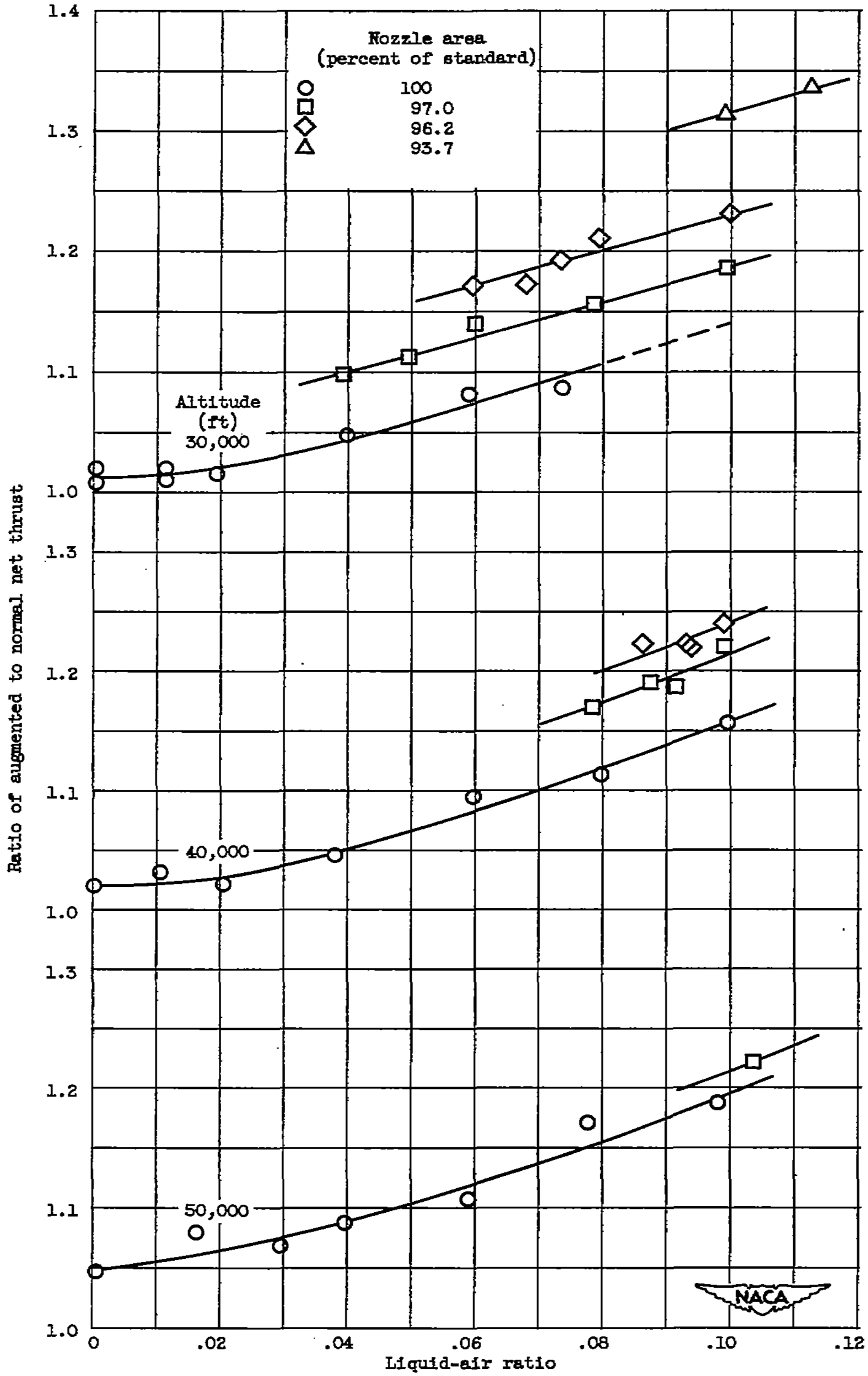

Figure 10. - Variation of ratio of augmented to normal net thrust with liquid-air ratio. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature; flight Mach number, 0.80.

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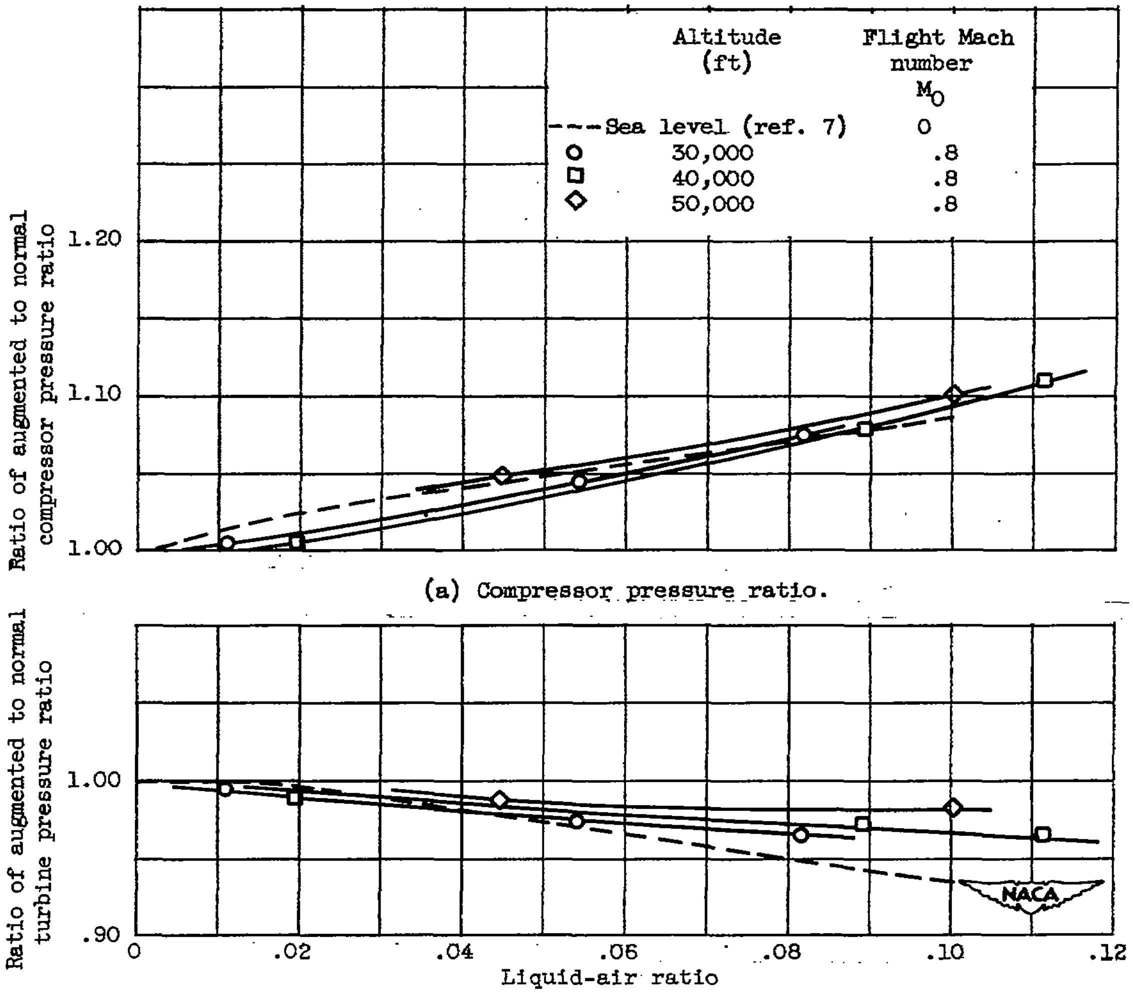

(a) Compressor pressure ratio.

(b) Turbine pressure ratio.

Figure 11. - Effect of liquid-air ratio on compressor and turbine pressure ratios with engine operating at limiting turbine-outlet temperature of 1250° F. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature.

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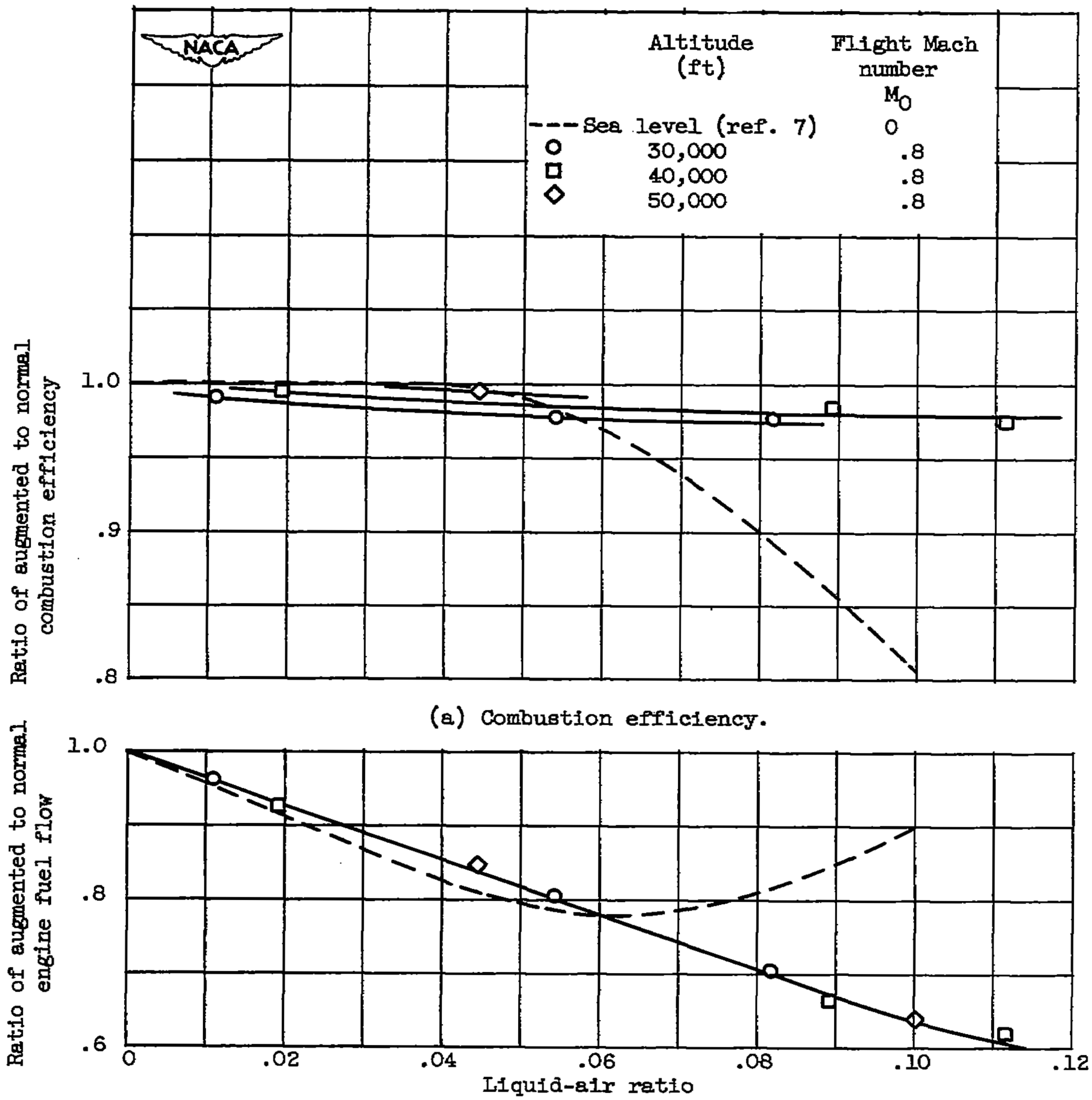

(a) Combustion efficiency.

(b) Engine fuel flow.

Figure 12. - Effect of liquid-air ratio on combustion efficiency and engine fuel flow with engine operating at limiting turbine-outlet temperature of 1250° F. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature.

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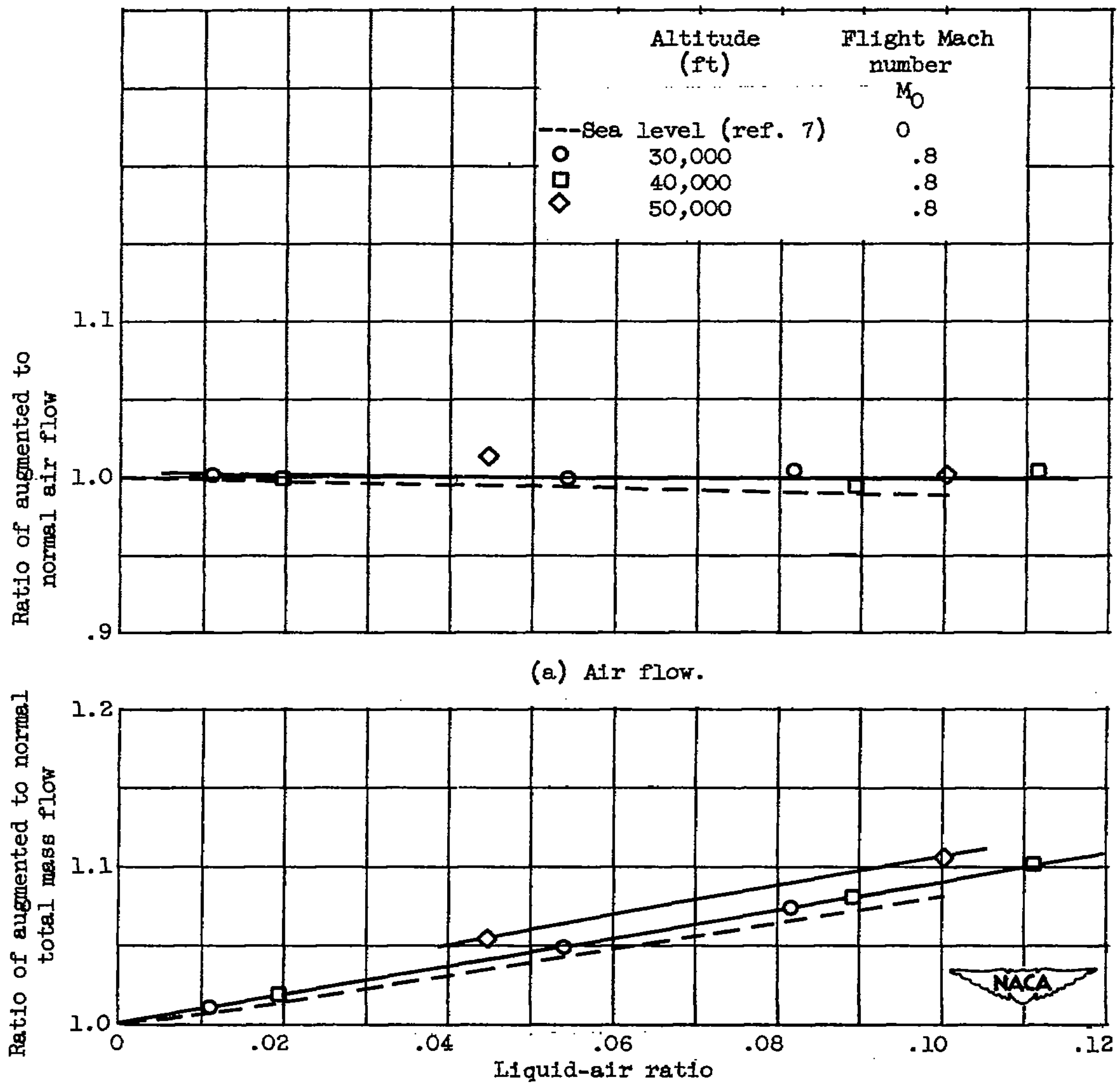


(a) Air flow.

(b) Total mass flow.

Figure 13. - Effect of liquid-air ratio on engine-inlet air flow and total mass flow with engine operating at limiting turbine-outlet temperature of 1250° F. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature.

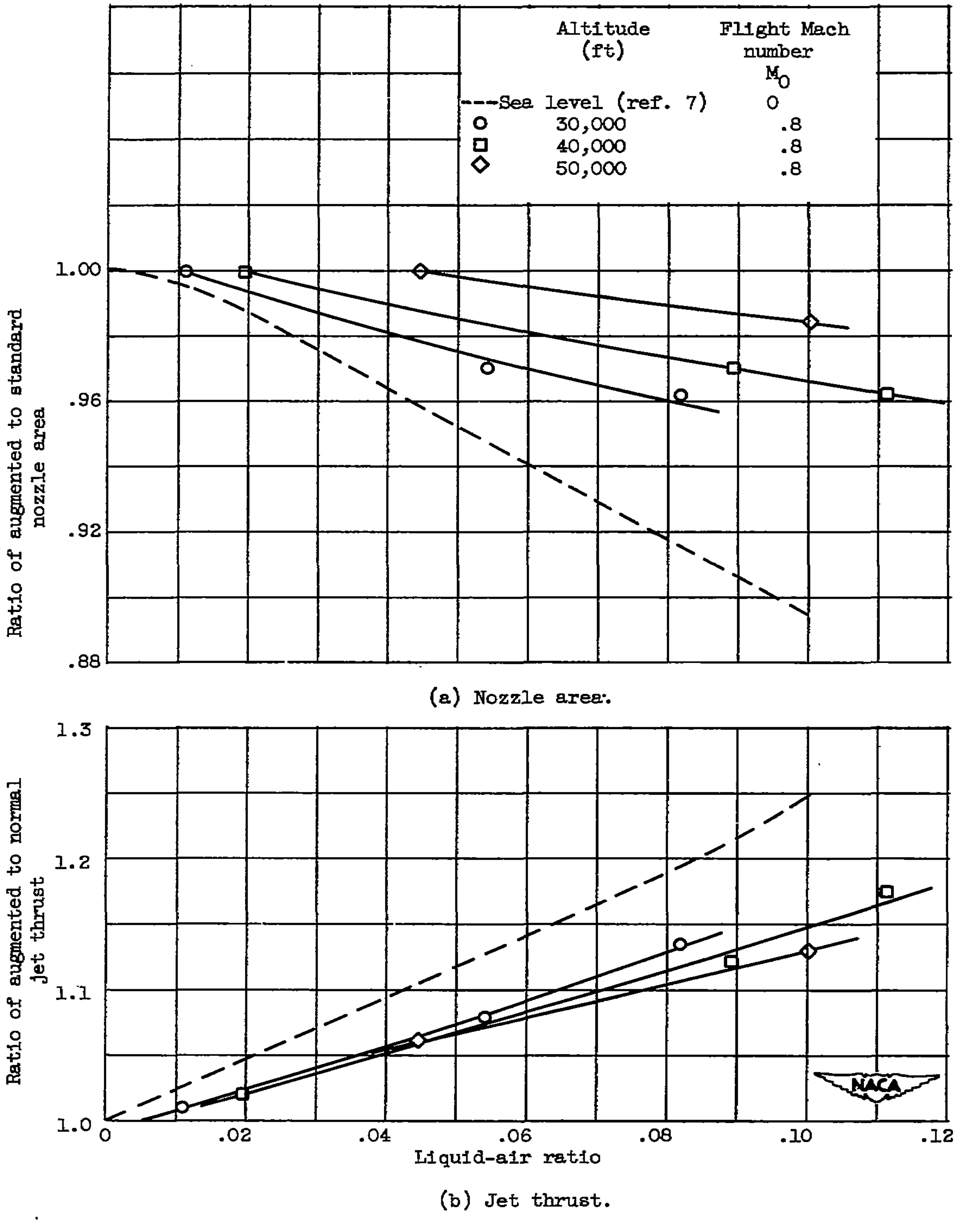
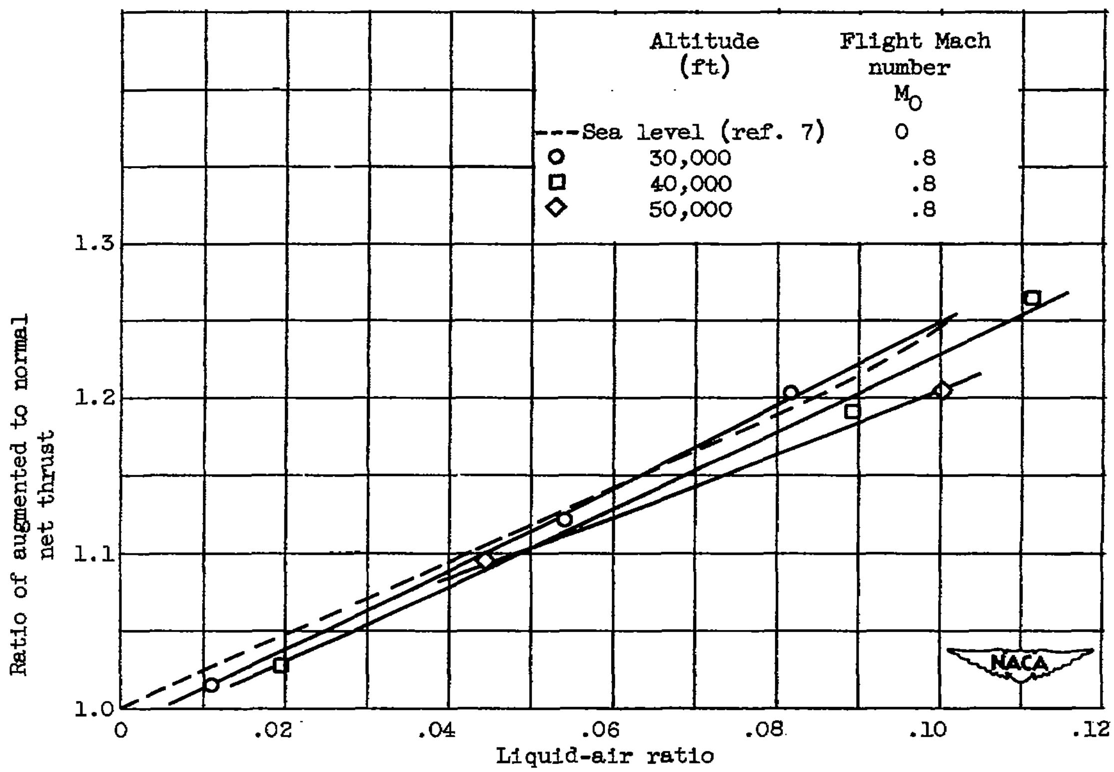


Figure 14. - Effect of liquid-air ratio on exhaust-nozzle area and jet and net thrusts with engine operating at limiting turbine-outlet temperature of 1250° F. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature.



(c) Net thrust.

Figure 14. - Concluded. Effect of liquid-air ratio on exhaust-nozzle area and jet and net thrusts with engine operating at limiting turbine-outlet temperature of 1250° F. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature.

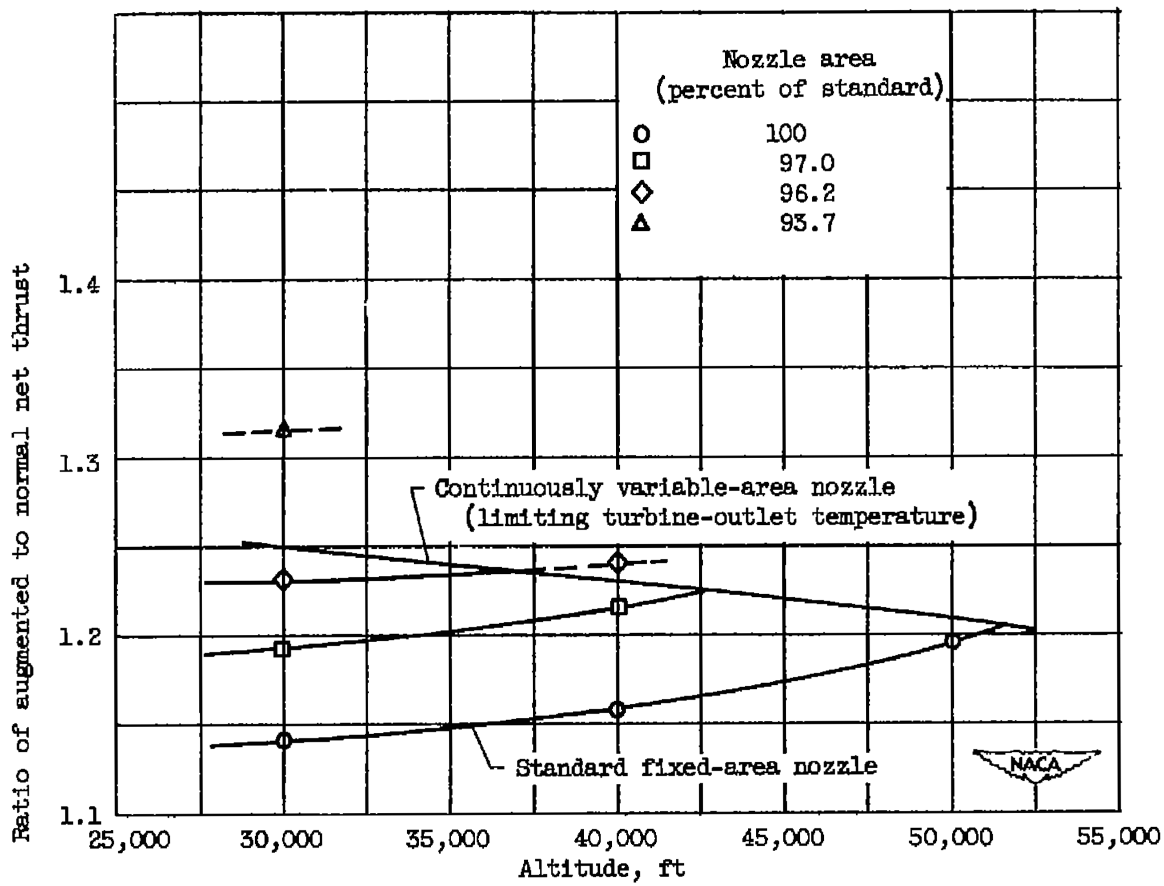
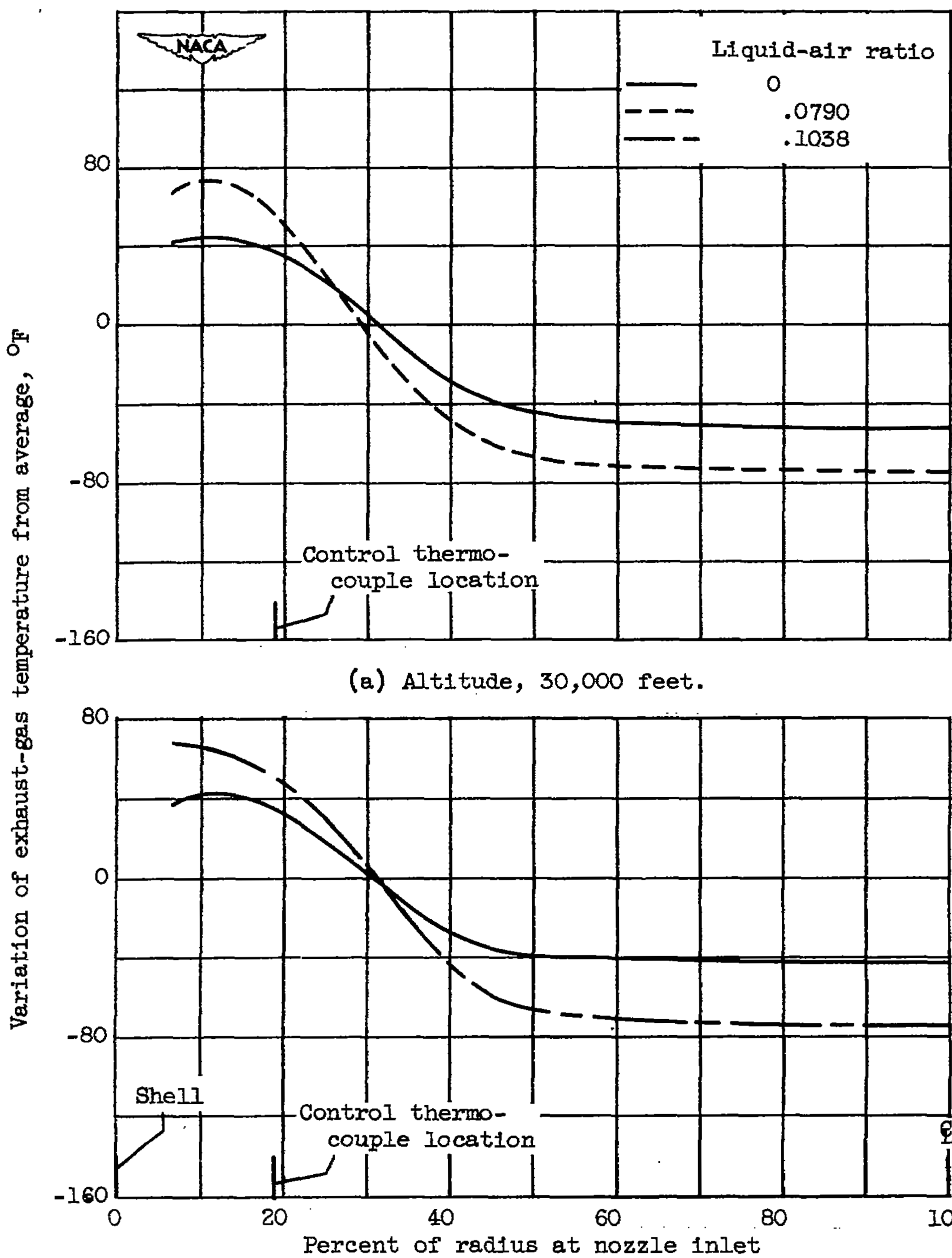


Figure 15. - Effect of altitude on thrust augmentation at given exhaust-nozzle area and liquid-air ratio of 0.10. Engine speed for augmented operation, 7950 rpm; engine speed for normal operation determined by rated turbine-outlet temperature; flight Mach number, 0.80.



(a) Altitude, 30,000 feet.

(b) Altitude, 50,000 feet.

Figure 16. - Exhaust-gas temperature profile at exhaust-nozzle inlet for two liquid-air ratios. Engine speed, 7950 rpm; flight Mach number, 0.80.

SECURITY INFORMATION

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