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RESEARCH MEMORANDUM

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AND AFTERBURNING FOR TURBOJET

THRUST AUGMENTATION

By James W. Useller, S. C. Huntley, and David B. Fenn

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COMBINED COMPRESSOR COOLANT INJECTION AND AFTERBURNING FOR TUROBJET THRUST AUGMENTATION

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SUMMARY

Investigations of the use of combined compressor coolant injection and afterburning in a turbojet engine have revealed that certain basic requirements must be considered in the design and operation of each part of the system to achieve maximum effectiveness of the whole. A discussion of some of these requirements and of the application of the augmented performance to high-speed, high-altitude flight is presented.

Efficient and compatible performance with combined compressor coolant injection and afterburning can be obtained only by selecting a coolant that is suited to the flight operating conditions, introducing the coolant uniformly at the most advantageous location in the compressor, minimizing any tendency toward reducing the compressor performance, as well as giving proper consideration to the influence of the coolant on afterburner performance. Because of its high heat of vaporization, water is a satisfactory compressor coolant at the normal air temperatures associated with sea-level take-off conditions, while liquid ammonia possesses properties favorable for evaporative cooling at the low temperatures associated with high-altitude flight. Both water and ammonia exhibited adverse effects on the combustion performance of the afterburner at high coolant flow rates. For example, with an afterburner stoichometric fuel-air ratio a combustion limit was reached at an ammonia-air ratio of 0.04.

An analysis based on the experimental results shows that for a flight Mach number of 2.5 at an altitude of 35,000 feet, an augmented net-thrust ratio of 6.9 is possible from the combined use of stoichiometric afterburning and a compressor injection ammonia-air ratio of 0.04. Stoichiometric afterburning alone produces an augmented-thrust ratio of only 4.4.

INTRODUCTION

Afterburning is the system most often used to augment standard engine thrust. Occasionally, however, it is insufficient to meet the thrust requirements during take-off, acceleration, and climb of the aircraft, and additional means of thrust augmentation are necessary. When maximum permissible thrust is needed for a short time, a combination of compressor coolant injection and afterburning is a promising method of thrust augmentation. Such a combined system of augmentation employs evaporative cooling before and during the compression stage of the engine and afterburning in the exhaust stage. This use of combined compressor coolant injection and afterburning to produce a maximum level of thrust augmentation has been investigated at conditions simulating sea-level take-off (ref. 1) and high-altitude flight (ref. 2).

This report summarizes the available information on combined augmentation systems using compressor coolant injection and afterburning and also discusses the effects of the compressor coolant on component performance and afterburning. In addition, an extrapolation of the experimental data to high flight speeds has been made, taking into account the anticipated performance variation or limitation caused by compressor coolant injection.

APPARATUS

Sea-level investigation. - A schematic diagram of the test vehicle employed for the sea-level investigation is presented in figure 1. The turbojet engine was an early production model incorporating an 11-stage axial-flow compressor, eight cylindrical combustors, and a single-stage turbine. The nominal sea-level static military rating of the engine was 4000 pounds thrust at a rotor speed of 7700 rpm. The afterburner was designed to operate at stoichiometric fuel-air ratio. The water-alcohol mixture was injected ahead of the compressor inlet and at the sixth stage of the compressor. The inlet-injection system consisted of 34 conventional atomizing nozzles installed in a ring, while the interstage system consisted of 20 individual nozzles of 0.045 inch diameter (details in ref. 1). The interstage nozzles were designed using the spray characteristic information contained in reference 3. When the inlet and interstage injection systems were used concurrently, the coolant flow was evenly divided between the two stations.

Altitude investigation. - The axial-flow turbojet engine used during the altitude investigation developed 3000 pounds thrust at sea-level, zero-ram conditions with an average turbine-outlet gas temperature of 1625° R and a rotational speed of 12,500 rpm. The primary engine components included an 11-stage compressor with a compressor-outlet mixer, a double annular combustor, a two-stage turbine, a shrouded air-cooled afterburner, and a water-cooled, variable-area exhaust nozzle. A sectional view of the engine, the compressor injection system, and the afterburner is shown in figure 2. Two ammonia injection configurations were investigated to determine the effect of distribution on the augmentation. The ammonia injection system of configuration A consisted of 20 radial spray bars each having 16 orifices 0.021 inch in diameter that were radially spaced in pairs so as to introduce the ammonia at the centers of eight equal areas (ref. 2). The spray bars were designed using the penetration characteristics described in reference 4. Configuration B was comprised of 24 conventional atomizing nozzles installed in a ring. The investigation was conducted in an altitude test chamber in which a range of desired ram-pressure ratios and altitude static pressures could be obtained.

PROCEDURE

During the sea-level investigation of reference 1, the augmented performance was determined at zero-ram conditions with an average engine-inlet temperature of 80° F. The engine was operated at rated speed; data at rated turbine-outlet temperature were obtained at various injection rates by using a series of fixed-area exhaust nozzles.

For the altitude investigation of reference 2, the flight condition simulated was an altitude of 35,000 feet and a flight Mach number of 1.0 (engine-inlet temperature, 13° F). The engine was operated at rated speed, and rated turbine-outlet temperature was maintained by adjustment of a variable-area exhaust nozzle. The coolant- to air-flow ratio for the sea-level investigation was varied from 0 to about 0.085, while coolant- to air-flow ratio for the altitude investigation was varied from 0 to 0.045.

The procedure used in extrapolating the experimental performance to high flight speeds is discussed in the text.

RESULTS AND DISCUSSION

The reported investigations of combined augmentation systems have revealed certain factors that must be considered in the design and operation of such systems in order to achieve maximum effectiveness during both augmented and unaugmented operation. Therefore, the design and performance factors of a compressor coolant system to be used in conjunction with afterburning that must be considered, such as type of coolant, place of injection, and effect of coolant on engine and afterburner performance, are discussed herein. The first section of the discussion compares the experimental results with the theoretical calculations and points out some of the considerations involved in the selection of the coolant system. The second section discusses a proposed application of a specific combined augmentation system to high-speed, high-altitude flight. Whereas theoretical applications of combination systems to high-speed flight are discussed in previously published reports, the application reported herein is an extrapolation of experimental results of a particular engine-afterburner-coolant configuration and includes the anticipated performance limitations imposed by the compressor coolant injection.

Experimental Performance of Combined System

Thrust augmentation. - Theoretical calculations of thrust augmentation from the combined use of compressor evaporative cooling and stoichiometric afterburning as reported in reference 8 are compared in figure 3 with data obtained during the experimental sea-level investigation of reference 1 for sea-level static flight condition. In the calculations of reference 5, a coolant with the heat of vaporization of water was used and the coolant flows were varied from that sufficient to saturate the compressor-inlet air to that required to saturate the compressor-outlet air. The coolant used in reference 1 was a mixture of 70 percent water and 30 percent alcohol. The experimental results exhibit the same trend as the theoretical up to an augmented liquid ratio (ratio of total liquid flow to engine fuel flow required for the particular flight condition and augmentation system) of 5.5, with the amount of thrust augmentation being about 5 percent less. However, for augmented liquid ratios in excess of 5.5 (coolant- to air-flow ratios greater than 0.033), the experimental performance was limited to that shown in the figure by a combustion instability that will be discussed in the section entitled "Effect of coolant on afterburner performance."

A similar comparison of the experimental data (from ref. 2) and a theoretical calculation are shown for an altitude of 35,000 feet and a flight Mach number of 1.0 in figure 4. For the simulated flight condition, liquid ammonia was used as the compressor coolant and was injected ahead of the compressor. The experimental results are limited to a maximum augmented-thrust ratio of 2.13 by combustion blow-out. There was no sharp decrease in augmentation when higher ammonia-air ratios were investigated, as was the case with water injection, and the experimental thrust augmentation was at the most only 3 percent lower than that theoretically predicted. The experimental results of figure 4 agree more closely with the theoretical results than those of figure 3 because the afterburner combustion efficiency assumed for the theoretical calculations was nearer to the value obtained with the actual afterburner. (See appendix B for method of computation.)

Introduction of coolant. - The most favorable station for injection of the coolant is ahead of the compressor if the coolant-flow rate is low enough to permit complete evaporation before compression. This station is also most favorable as to simplicity of installation. If, however, complete evaporation of the coolant cannot be accomplished ahead of the compressor, the excess coolant enters the compressor in liquid form. Before it can evaporate, it is centrifuged out to the compressor casing where it causes thermal contraction that may result in blade rubbing and deterioration of engine performance (ref. 6). Because of this effect of the liquid on the compressor, interstage injection of that portion of the coolant which cannot be evaporated ahead of the compressor may be required.

The maximum amount of evaporation of any coolant is obtained from a uniform distribution of the coolant in the flow passage. A comparison of the performances of two injection systems having different distribution systems is presented in figure 5. Configuration A (from ref. 2) had 320 orifice sources of coolant injection, while configuration B (from ref. 7) had 24 atomizing-nozzle sources. Although the actual distributions were not measured, it is evident that the coolant distribution for configuration A was more uniform than that for configuration B, and the thrust augmentation increased as much as 5 percent with the improved distribution.

Selection of coolant. - Because the thrust augmentation obtained from compressor coolant injection is a result of the heat absorption during evaporation of the coolant, the coolant used must be selected in relation to the atmospheric conditions associated with the flight application and from a consideration of its physical and chemical properties. The primary factors involved with respect to physical properties include heat of vaporization, flammability limit, and boiling and freezing point temperatures. A survey of the physical properties of various potential coolants considered resulted in the following list:

Coolant	Heat of vaporization, Btu/lb	Flammability limit, percent by volume	Boiling point at atmospheric pressure, °F
Water	1079	Incombustible	212
Hydrogen peroxide	587	Heat of decomposition, 1242 Btu/lb	306
Ammonia (liquid)	561	0.16 - 0.25	-27
Ethyl alcohol	406	0.03 - 0.19	172
Methyl chloride	180	0.08 - 0.17	-12
Carbon dioxide (liquid)	110	Incombustible	-109
Air (liquid)	93	Supports combustion	-320

The high heat of vaporization of water makes it the most attractive coolant. The addition of alcohol to the water is usually necessary to prevent freezing at lower temperatures and to permit constant-throttle operation, which is possible because the alcohol aids the engine fuel in heating and vaporizing the water. For the low temperatures encountered during altitude operation, water is unsuitable because very little evaporative cooling can be obtained inasmuch as small amounts of water saturate the air. When a water-alcohol mixture was injected at the engine inlet during an altitude investigation (ref. 8), evaporation of the coolant was inadequate as a result of the low inlet air temperature and no augmentation was obtained. In addition, failure of the coolant to evaporate ahead of the compressor permitted liquid coolant to enter the compressor and adversely affected the compressor performance (to be discussed in detail in section entitled "Effect of coolant on compressor performance").

Hydrogen peroxide presents a distinct problem since it is spontaneously combustible at temperatures approximating the compressor-discharge temperature. Methyl chloride, liquid air, and liquid carbon dioxide all have heats of vaporization approximately 10 percent of that of water; thus they can absorb only small quantities of heat during evaporation unless much greater quantities of coolant are used.

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The low temperatures encountered at high-altitude flight indicate the need for a coolant such as ammonia that will evaporate in appreciable quantities at low temperatures. Liquid ammonia also has a relatively favorable heat of vaporization, approximately one-half that of water, and its combustible nature is also desirable for it would tend to reduce the primary fuel requirement of the engine. Liquid ammonia does not have universal application, however, since it is readily absorbed by the water vapor present in the air. This absorption process is exothermic and would tend to negate the cooling due to evaporation. Since atmospheric humidity is generally highest at sea level, ammonia would prove an undesirable coolant for sea-level applications, although it appears to have promise for use where the air is relatively dry.

Offsetting these advantages is the fact that ammonia is corrosive in the presence of certain metals and its use would require that care be taken not to use metals such as copper and brass in the ammonia system or in the engine where the ammonia may have contact with them. Special measures must also be taken that ammonia vapors are not contained in air taken from the compressor for cockpit pressurization and bearing cooling.

Effect of atmospheric conditions. - Although the afterburner is relatively insensitive to small changes in atmospheric temperature and humidity, the influence of these conditions on the evaporative cooling system can be expected to be more significant. In general, increased engine-inlet air temperatures will permit greater cooling and result in more favorable augmentation at equivalent coolant- to air-flow ratios. The increasing augmented-thrust ratio obtained with increasing inlet air temperature for a coolant- to air-flow ratio of 0.06 during operation at sea-level zero-ram conditions is shown in figure 6(a). For both the centrifugal engine (ref. 9) and the axial-flow engine (ref. 6), the increases in augmentation due to increasing inlet air temperature are similar. The effects of temperature on augmented-thrust ratio as found in the investigations of references 2, 7, and 8 at coolant-air ratios of 0.04 and during altitude flight operation are shown in figure 6(b) for water-alcohol and ammonia coolants. The altitude performance exhibits the same trends as the sea-level data.

The sea-level investigation of reference 9 indicated that at a given inlet air temperature and coolant-air ratio, the compressor pressure ratio and resultant thrust decreased slightly with increasing humidity of the inlet air; this effect became more pronounced as the coolant-flow rate was increased. The decrease in unaugmented performance due to high humidity was found to be a result of changes in the physical properties of the air (γ and c_p), and has been reported in reference 10 to be quite small in magnitude. The effect of humidity increases with coolant-flow rate because as saturation of the air is approached less coolant can evaporate and thus less cooling and less augmentation are provided. It has already been pointed out that high humidity decreased the desirability of liquid ammonia as a coolant since the absorption of water by anhydrous ammonia is exothermic and tends to negate any cooling due to the evaporation of the ammonia. Although no data have been obtained as to the effects of humidity on the altitude flight use of coolant injection, the effects are expected to be small because relatively little moisture can be present at the low temperatures normally encountered at high altitudes.

Effect of coolant on compressor performance. - The major influences of coolant injection are, of course, on the compressor performance and the afterburner combustion. For a coolant-flow rate less than that required to saturate the air at the compressor inlet, the variation of compressor performance with increasing coolant-flow rate is the same as the variation that would result from a variation in corrected engine speed. The variation of adiabatic compressor efficiency with corrected engine speed is shown in figure 7. The data points indicated are for calculated compressor efficiency over a range of ammonia-air ratios from 0.01 to 0.045 (corresponding to corrected engine speeds from 12,800 to 14,700 rpm), while the curve represents the normal compressor efficiency variation without coolant injection.

When the compressor-inlet air is saturated and liquid coolant enters the compressor, only limited verification of experimental data can be made because of the difficulties encountered in making accurate temperature measurements at the compressor inlet and outlet (see ref. 11 for qualitative discussion). When a water-alcohol mixture is used as the coolant at the compressor inlet, the saturation condition may be exceeded even at very low coolant-air ratios. For example, at the ram temperature associated with sonic flight at an altitude of 35,000 feet (13° F), the compressor inlet would be saturated with a coolant-air ratio of only 0.003. With unevaporated coolant entering the compressor, the compressor actually operates at different Mach numbers and loadings throughout, depending on stagewise evaporation. Because the stagewise pattern of evaporation of the coolant was unknown, the compressor was divided into four hypothetical axial sections having equal pressure ratios to facilitate the calculation of compressor efficiency when an aqueous coolant was used. A stepwise iteration was made to calculate temperature and efficiency through the compressor based on the assumption that for each step the adiabatic compression was followed by a constant-pressure cooling process until saturation was reached. Equal compressor efficiencies were initially assumed for the axial sections of the compressor, and a compressor-outlet temperature was calculated. This calculated temperature was compared with the measured temperature and the procedure repeated using altered compressor section efficiencies until the two temperatures agreed. The assumed compressor efficiency for the last iteration was taken to be the over-all compressor efficiency. The resultant variation of compressor efficiency with ratio of water-alcohol to air is shown in figure 8. While the injection of ammonia caused no appreciable decrease of compressor efficiency from the normal performance, the injection of a 70 percent water - 30 percent alcohol mixture resulted in as high as a 20 point drop in compressor efficiency.

Effect of coolant on afterburner performance. - Unfortunately, most liquids that have proven satisfactory as compressor coolants exhibit a negative reaction toward the kinetics of the combustion process. This fact is illustrated in figure 9, which shows the effect of increasing coolant-air ratio on afterburner temperature and combustion efficiency for operation at equivalence ratios of 1.0 and 0.8 for both the water-alcohol mixture (figs. 9(a) and (b)) and the liquid ammonia coolants (figs. 9(c) and (d)). These data are taken from references 1 and 2, respectively. At an equivalence ratio of 0.8, both water-alcohol and liquid ammonia injection tend to decrease the afterburner temperature and combustion efficiency as the coolant-air ratio is increased. At an equivalence ratio equal to 1.0 (stoichiometric mixture) the ammonia had relatively little effect on the combustion efficiency, whereas the water-alcohol coolant was only slightly less effective than at an equivalence ratio of 0.8. Although both coolants, in general, tend to decrease the combustion reaction rate, it is believed that the high temperature associated with stoichiometric operation dissociates the ammonia into more readily combustible products and thus reduces the influence of the ammonia on the reaction rate. A detailed discussion of the influence of water vapor and ammonia on combustion may be found in references 12, 13, and 14.

As appreciable quantities of a coolant are introduced, the range of equivalence ratios over which the afterburner operates with a reasonable degree of stability narrows. This effect is shown in figure 10 where for an equivalence ratio of 1.0 the maximum possible coolant-air ratios are 0.045 for ammonia injection and 0.065 for water-alcohol injection. In the investigations summarized herein (refs. 1 and 2), two types of combustion instability were noted. The combustion instability encountered during the sea-level investigation with the use of water-alcohol coolant was characterized by high-frequency vibration of a destructive nature characterized as screech. Because the afterburner demonstrated some tendency toward this form of instability without coolant injection, it is believed that the coolant tended only to aggravate the condition producing the instability. Although it was possible to eliminate the instability during independent operation of the afterburner, this was not always possible when the combined system was used. The instability encountered with ammonia took the form of combustion blow-out. Although this form was not destructive, it definitely limited the amount of augmentation obtainable from the ammonia injection phase of the combined system.

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Improvement of combined augmentation system performance. - Greater augmentation with the combined system obviously necessitates improved afterburner performance (see ref. 15) or, at the least, minimized adverse influence of the coolants on the afterburner performance. A liquid that will qualify as a good coolant and still possess desirable combustion properties is difficult to find. An alternative approach would be the use of an additive in the afterburner fuel to counteract the deleterious effects of the coolant. The small-scale tests of reference 16 showed that afterburner operation was possible with water-air ratios of 0.17 when a magnesium slurry composed of 60 percent by weight of powdered magnesium suspended in a hydrocarbon fuel was used. The slurry (or high-energy) fuel also has the added advantage of greater heat release and a subsequent higher temperature during combustion. Although no data have been obtained as yet, it is believed that the higher temperatures achieved with the slurry fuels would reduce the negative influence of the ammonia on the combustion process, and thus it might be possible to improve the combination performance below stoichiometric when ammonia is used as the coolant. The use of a slurry fuel presents several problems of a mechanical nature, however, such as handling, fuel metering, spray characteristics, and pumping of a stabilized slurry. These details are discussed extensively in reference 17.

Application of Augmented Performance to High-Speed Flight

With the foregoing design, operational, and performance information, the potential application of a combined augmentation system to the demands for high magnitudes of thrust augmentation envisioned by aircraft designers may be logically analyzed. The experimental data have therefore been extrapolated to indicate the amount of thrust augmentation available at high flight speeds.

The potential thrust augmentation available to a hypothetical turbojet engine operating at high-speed, high-altitude conditions has been computed using experimental data obtained from this compressor coolant injection and stoichiometric afterburning investigation for flight at an altitude of 35,000 feet and flight Mach numbers from 1.0 to 2.5. Stoichiometric afterburner operation with compressor injection of ammonia at an ammonia-air ratio of 0.04 has been assumed so as to keep the data in the region of stable afterburner operation determined in the investigation of reference 2. The complete evaporation of liquid ammonia before compression has been assumed, and adjustments to the calculated air flows and variations in gas properties have been made for ammonia injection. While previous theoretical calculations have assumed constant compressor and afterburner combustion efficiencies, the application presented herein has allowed for the variation in these efficiencies indicated by available data. The analysis has assumed a constant turbine-discharge gas temperature which requires the use of a variable-area exhaust nozzle to take full advantage of the available augmentation.

A list of the symbols used in the calculations is included in appendix A. A detailed explanation of the assumptions and extrapolation of the data of reference 2 are included in appendix B. The air-flow curve shown in figure 11 has been assumed for the hypothetical engine and is typical of that of current turbojet engines. It must be recognized that the slope of the linear portion of the curve is closely related to the augmentation achievable, inasmuch as a greater slope would provide larger air-flow increases per unit change in corrected speed. A curve of the pumping characteristics of the hypothetical engine is shown in figure 12, which is also typical of current turbojet engines. Similarly, the rate of change of the pressure ratio with the temperature ratio will significantly influence the augmentation actually available.

The results of the thrust augmentation calculation are shown on figure 13, which presents a comparison of the performance obtained with stoichiometric afterburning alone and with stoichiometric afterburning in combination with a compressor-inlet injection ammonia-air ratio of 0.04 for an altitude of 35,000 feet and flight Mach numbers from 1.0 to 2.5. At a flight Mach number of 1.0 the ammonia injection resulted in only a 0.22 increase in augmented net thrust over stoichiometric afterburning. At a flight Mach number of 2.5, the augmented net-thrust ratio computed for the combination of ammonia injection and afterburning was 6.9, while with afterburning alone it was only 4.4.

The augmented liquid ratios associated with the data of figure 13 are presented in figure 14, where the augmented liquid ratio is defined as the ratio of total liquid flow to the engine fuel flow required at the particular flight condition for that particular augmentation system. The increase in augmented liquid ratio with flight Mach number results from the increased engine-inlet air temperatures which reduce the engine fuel flow. With the afterburning configuration at a flight Mach number slightly in excess of 3.0, the ram-temperature rise will equal the normal temperature rise across the engine combustor, and the augmented liquid ratio will numerically approach infinity. With the combination system this condition is reached at a flight Mach number of 2.5 because the ammonia serves as an engine fuel. For engines capable of operating at higher turbine-inlet temperatures, this limiting Mach number would occur at a higher value. Of course, engine operation would be limited to some flight Mach number less than this limiting Mach number by compressor structural considerations and engine speed control and stability. It should be noted that the rapid rise of the augmented liquid ratio curve at the limiting Mach number is a result of the decreasing engine fuel flow required with higher inlet temperatures at the higher Mach numbers and is not an indication of the increase in total liquid consumption. For flight speeds between Mach 2.0 and Mach 2.5, favorable thrust characteristics for application to supersonic aircraft requiring short periods of increased acceleration are indicated.

The engine performance variables (compressor-inlet temperature, fraction of rated actual air flow, and exhaust-nozzle pressure ratio) directly related to the thrust augmentation are shown in figure 15. Both the air flow and the exhaust-nozzle pressure ratio (related to jet velocity) are shown to increase over the values obtained with afterburning alone because of the effect of the coolant on engine-inlet temperature. Although no substantiating data are available, it may be expected that with the high ram temperatures encountered in flight at Mach numbers of the order of 2.5, aqueous coolants would prove useful. This might be especially desirable if afterburner operation were limited to some value below stoichiometric as a result of a compromise with the afterburner shell cooling requirements. Water would exert a smaller negative influence on the combustion efficiency than ammonia at the combustion temperatures associated with operation at equivalence ratios less than 1.0. If the combustion instability limitation encountered at an ammonia-air ratio of 0.045 could be eliminated to permit greater cooling from higher ammonia-flow rates, even greater augmentation could be anticipated.

CONCLUDING REMARKS

Turbojet engine thrust augmentation provided by a combination of compressor evaporative cooling and afterburning has been experimentally demonstrated to be feasible. The selection of a suitable coolant for the compressor injection involves primarily a consideration of the atmospheric conditions associated with the flight application and the physical properties of the coolant.

Because of its high heat of vaporization, water is a satisfactory compressor coolant at the normal air temperatures associated with sea-level take-off conditions. However, at the low temperatures encountered during high-altitude subsonic flight, water is unsatisfactory as a coolant because small amounts of water saturate the air and very little evaporative cooling can be obtained. Liquid ammonia, on the other hand, possesses properties favorable to evaporative cooling at low temperatures. The greatest increases in air flow and compressor pressure ratio are to be realized from use of coolants which give the greatest reduction in temperature entering the compressor. However, if saturation is realized at the compressor inlet, stagewise evaporation is beneficial. For three axial-flow engines at high coolant-flow rates where complete evaporation could not take place ahead of the compressor, the use of interstage injection in conjunction with the inlet injection was necessary to prevent centrifugation of the unevaporated coolant onto the compressor casing and attendant blade rubbing due to differential expansion.

Both water and ammonia exhibited adverse effects on the combustion performance of the afterburner, and at high coolant flow rates their presence usually caused some form of combustion instability. The nature and severity of this condition vary with the coolant and are a result of the molecular behavior of the coolant during combustion.

An analysis based on the experimental results extended to include flight at a Mach number of 2.5 and an altitude of 35,000 feet shows that an augmented-net-thrust ratio of approximately 4.4 can be obtained by stoichiometric afterburning alone. The addition of compressor coolant injection to this stoichiometric afterburning within experimentally determined combustion limits (0.04 coolant-air ratio) will increase the augmented-net-thrust ratio to approximately 6.9.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, July 14, 1954

APPENDIX A

SYMBOLS

The following symbols are used in the analysis:

A	area, sq ft
c_p	specific heat at constant pressure, Btu/(lb)($^{\circ}$R)
d	ammonia-air ratio, lb ammonia/lb air
F_j	jet thrust, lb
F_n	net thrust, lb
f	fuel-air ratio, lb fuel/lb air
g	acceleration due to gravity, 32.17 ft/sec^2
h	enthalpy, Btu/lb
$h_{c,d}$	lower heat of combustion of ammonia, Btu/lb
$h_{c,f}$	lower heat of combustion of fuel, Btu/lb
N	engine speed, rpm
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
T	total temperature, $^{\circ}$R
V	velocity, ft/sec
W	weight flow, lb/sec
γ	ratio of specific heats
δ	ratio of total pressure to static sea-level pressure, P/2116
η	efficiency, ratio of energy increase to ideal energy increase
θ	ratio of total temperature to static sea-level temperature, T/519

Subscripts:

a air

b afterburner

c compressor

d ammonia

e engine

g gas

m manifold

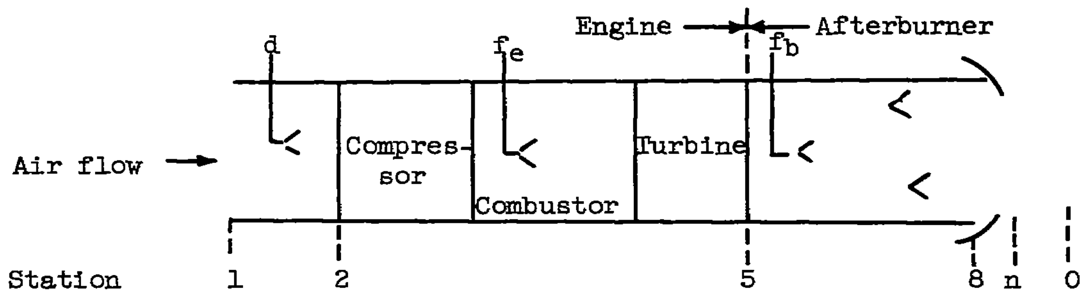
n vena contracta of jet nozzle

The numbered subscripts refer to stations indicated in figs. 1 and 2 and the sketch in appendix B.

APPENDIX B

METHOD OF ANALYSIS

The hypothetical engine under consideration in the extrapolation of the augmentation data to high speed was assumed to operate at rated mechanical speed with a turbine-outlet gas temperature T_5 of 1660° R maintained constant by the use of an ideal, convergent variable-area exhaust nozzle. A schematic view of the engine indicating the injection points of ammonia and fuel and also the various engine stations considered in the analysis is shown in the following sketch:



The following assumptions are necessary for the analysis:

(1) Inlet pressure and temperature, station 1, corresponded to NACA standard atmosphere with ram recovery that is representative of current supersonic inlet design.

(2) No pressure loss was incurred with the ammonia injection equipment, that is, P_2 equals P_1.

(3) The engine total-pressure ratio P_5/P_2 was a function of the total-temperature ratio T_5/T_2 as shown in figure 12.

(4) The afterburner total-pressure friction loss was 0.06 P_5, and with stoichiometric afterburning the total-pressure loss including friction was 0.12 P_5. These pressure losses correspond to the pressure loss measured for the afterburner in the altitude investigation as presented in this report.

(5) The corrected compressor air flow $\frac{W_a\sqrt{\theta_2}}{\delta_2}$ was a function of the corrected engine speed $\frac{N}{\sqrt{\theta_2}}$, as shown in figure 11.

(6) The engine combustion efficiency η_e was 0.97 with no ammonia injection and 0.62 with an ammonia-air ratio of 0.04.

(7) The afterburner efficiency η_b was 0.92 at a stoichiometric mixture.

(8) The manifold temperature of the ammonia and fuel was 80° F. The corresponding lower heats of combustion were 7500 Btu per pound and 18,725 Btu per pound, respectively. The hydrogen-carbon ratio of the fuel was 0.170.

The total pressure and total temperature at each station were calculated using the preceding assumptions for three cases: (1) normal engine performance, (2) stoichiometric afterburning, and (3) combined stoichiometric afterburning and an ammonia injection ratio d of 0.04.

The drop in inlet air temperature from T_1 to T_2 with an ammonia injection ratio of 0.04 was calculated from the energy balance between stations 1 and 2.

$$h_{a,1} + dh_{d,m(liq)} = h_{a,2} + dh_{d,2} \tag{1}$$

Enthalpies of ammonia were determined from reference 18.

The air flow was determined from the fraction of rated corrected speed, which is equivalent to $\frac{1}{\sqrt{\theta_2}}$, using figure 11. The air flow with ammonia injection was corrected for the difference between density of ammonia vapor and that of air in the following manner:

$$W_{a,2} (1 + 1.70d) = W_{g,2} = 0.989\ W_{a,2}\ (\text{indicated}) \tag{2}$$

where 1.70 is the ratio of density of air to that of ammonia. It was determined during the experimental investigations that 98.9 percent of the indicated air flow without ammonia injection could be achieved with an ammonia-air ratio of 0.04. Therefore, this factor was introduced to provide achievable air flow increases.

The engine fuel-air ratio f_e was calculated from an energy balance across the engine:

$$h_{a,1} + \eta_e\ (f_e h_{c,f} + dh_{c,d}) = (1 + f_e + d)h_{g,5} \tag{3}$$

and the afterburner fuel-air ratio f_b was calculated from the over-all stoichiometric relation:

$$f_b + f_e + \frac{0.0675}{0.1642} d = 0.0675 \tag{4}$$

where 0.0675 is the stoichiometric fuel-air ratio and 0.1642 is the stoichiometric ammonia-air ratio.

The combustion temperature T_8 was then determined from an energy balance across the afterburner:

$$(1 + f_e + d)h_{g,5} + \eta_b\left[(1 + f_e + f_b + d)h_{g,T_{max}} - (1 + f_e + d)h_{g,5}\right]$$
$$= (1 + f_e + f_b + d)h_{g,8} \tag{5}$$

The value of $h_{g,T_{max}}$ was determined from the ideal energy modified by an energy difference to take into account the increase in chemical energy in the products of combustion due to the effect of dissociation. The value of this energy difference was determined from data contained in reference 19.

The jet thrust was calculated from the following relation:

$$F_j = \frac{W_{g,8}}{g} V_n + A_n (p_n - p_0) \tag{6}$$

using the procedure given in reference 20 and $W_{g,8} = W_{a,1} (1 + f_e + f_b + d)$.

The net thrust was then calculated:

$$F_n = F_j - \frac{W_{a,1}}{g} V_0 \tag{7}$$

and the augmented-net-thrust ratios for stoichiometric afterburning and for combined stoichiometric afterburning with an ammonia injection ratio of 0.04 were calculated by dividing the respective net thrust for each case by that obtained for normal engine performance.

The thermodynamic properties of the exhaust gas were determined from the products of an ideal combustion using the weighted averaging process and values obtained from reference 21.

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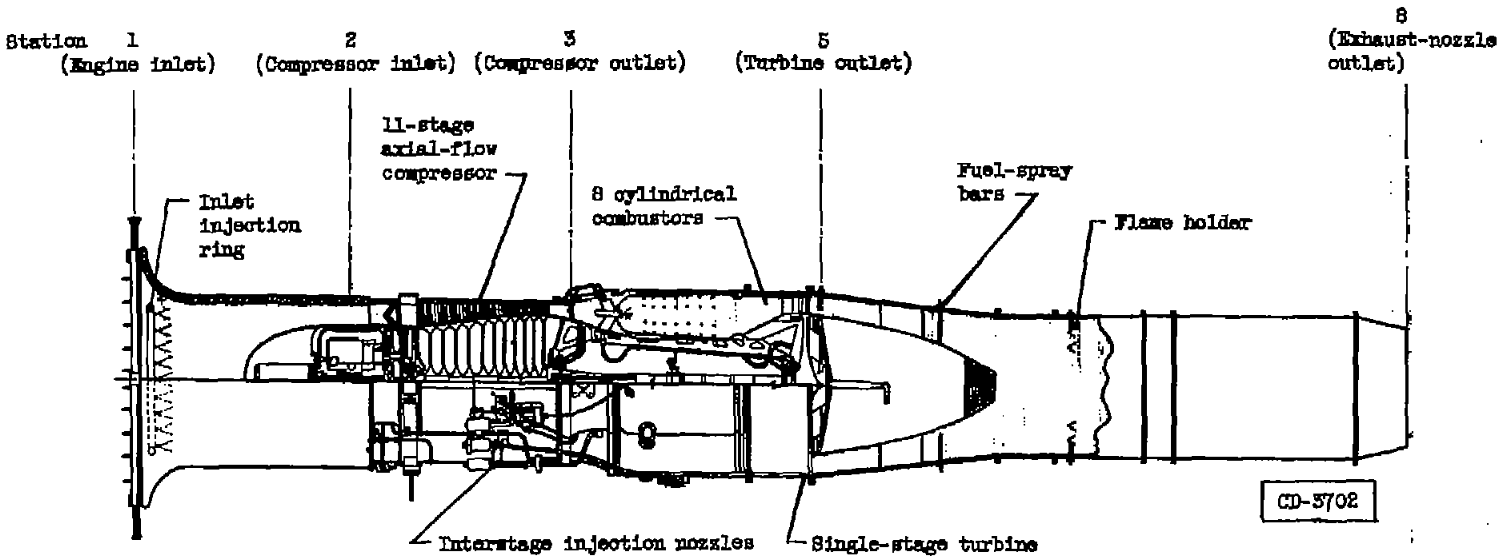

Figure 1. - Sectional view of turbojet engine with configuration of combined augmentation system used during sea-level static investigation.

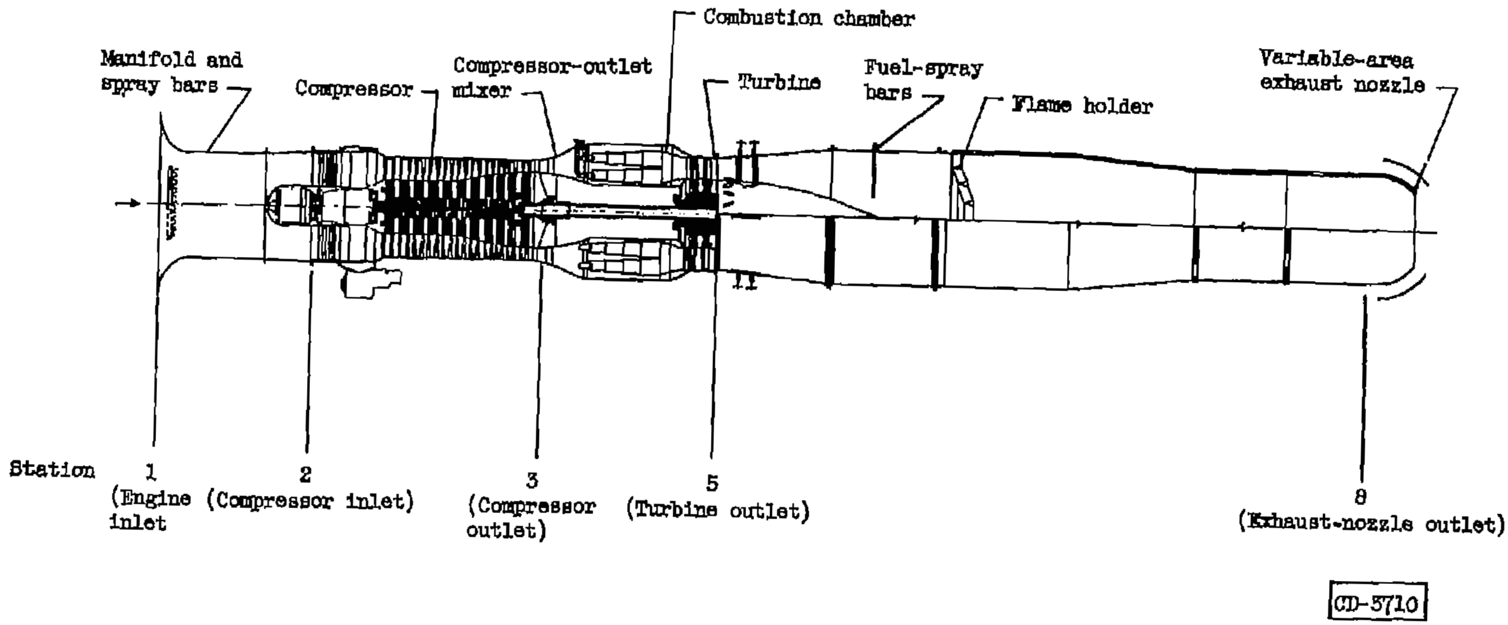

Figure 2. - Sectional view of turbojet engine with combined augmentation system used during simulated altitude flight investigation.

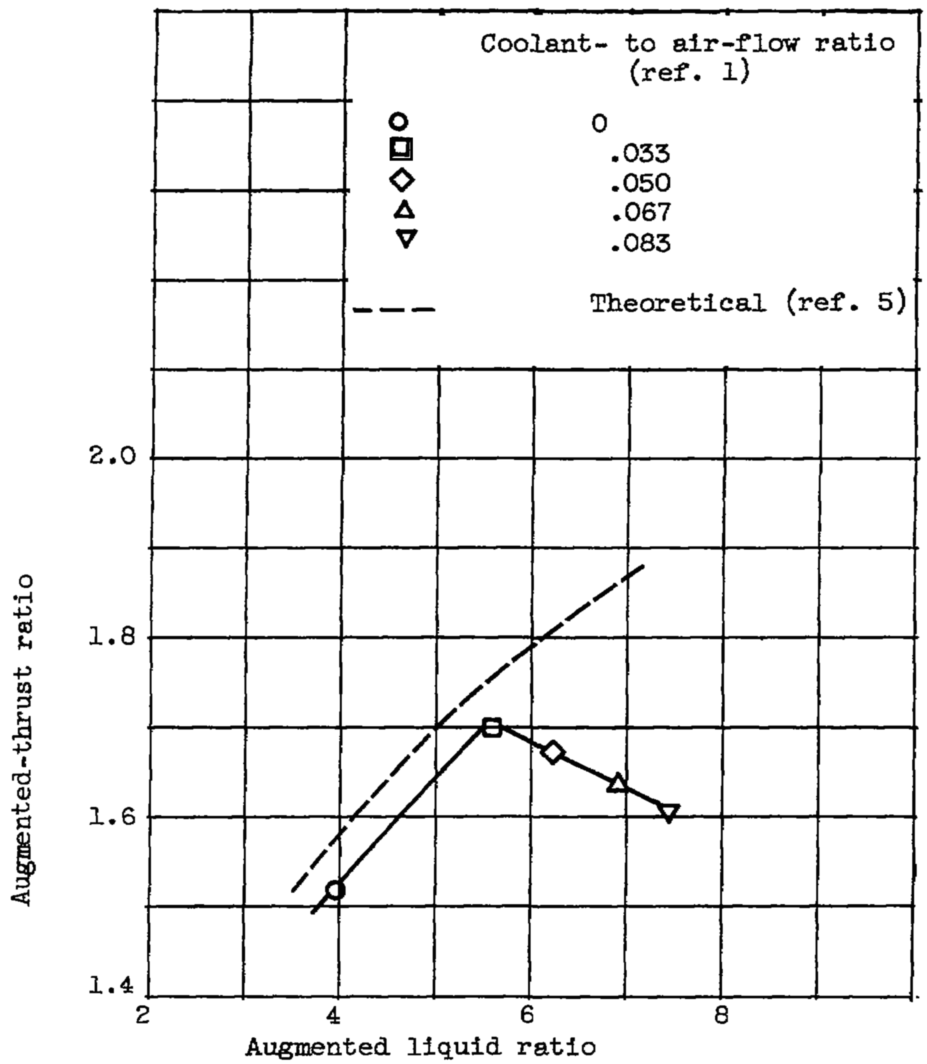

Figure 3. - Theoretically predicted and experimental thrust augmentation with combined inlet and interstage compressor water-alcohol injection and stoichiometric afterburning. Engine operation at sea-level, zero-ram conditions.

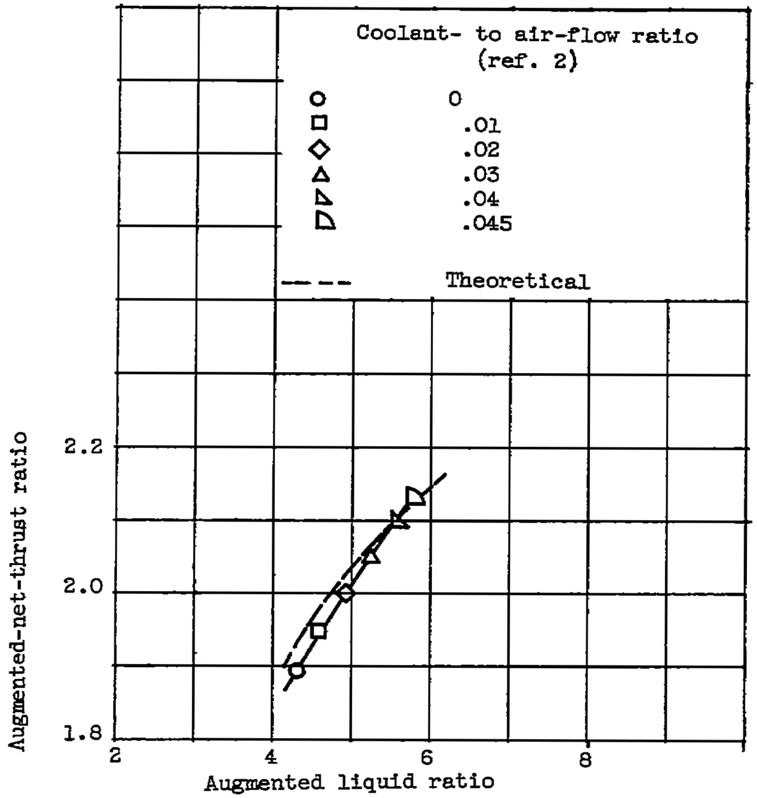

Figure 4. - Theoretically predicted and experimental net-thrust augmentation with combined compressor ammonia injection and stoichiometric afterburning. Engine operation at altitude of 35,000 feet and flight Mach number of 1.0.

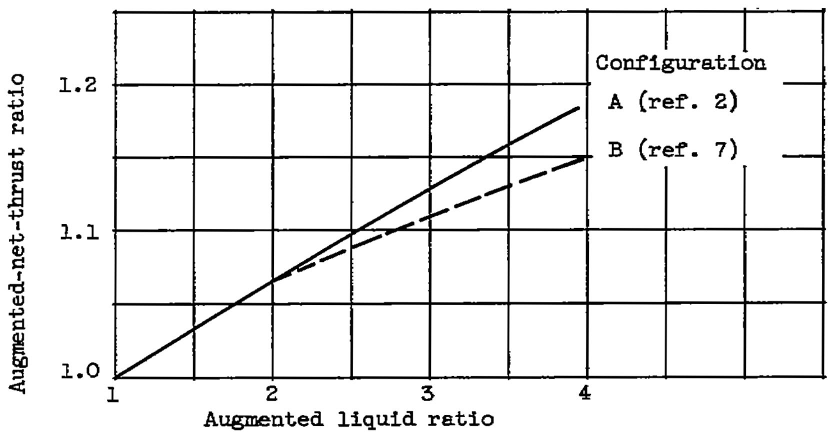

Figure 5. - Additional thrust augmentation achieved with improved distribution of ammonia and air at engine inlet. Engine operation at altitude of 35,000 feet, flight Mach number of 1.0, and engine-inlet air temperature of 13° F.

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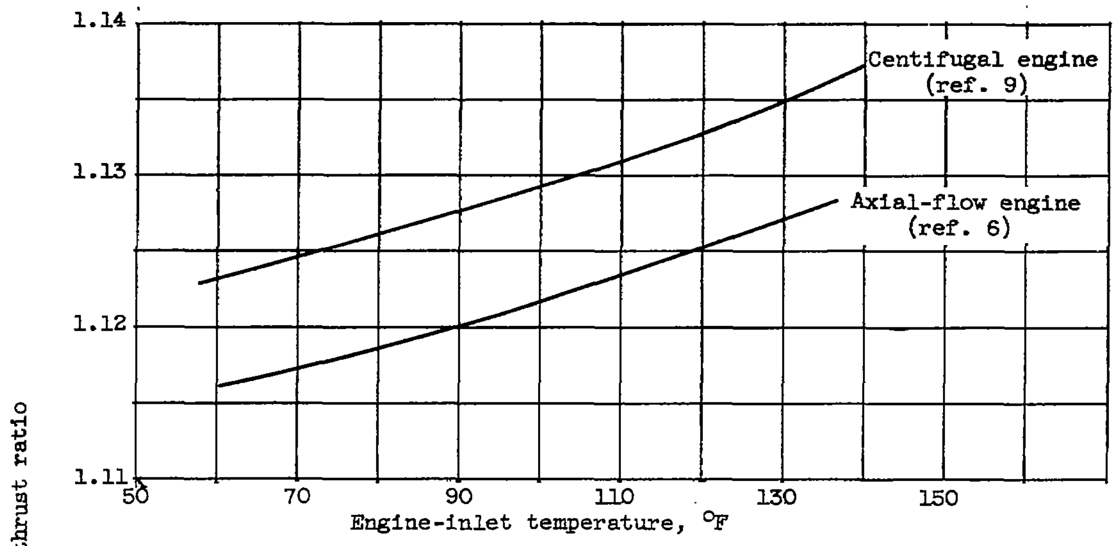
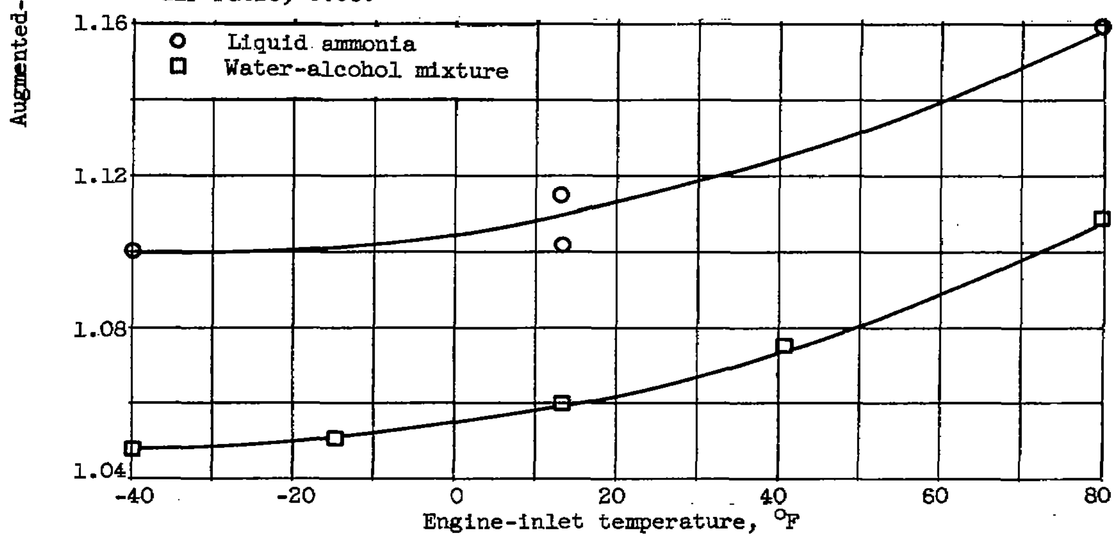


(a) Sea-level, zero-ram conditions. Water-alcohol mixture. Coolant-air ratio, 0.06.



(b) Simulated flight Mach number of 1.0 at altitude of 35,000 feet. Coolant-air ratio, 0.04

Figure 6. - Effect of engine-inlet temperature on thrust augmentation by compressor injection of coolant at sea-level take-off and altitude flight conditions.

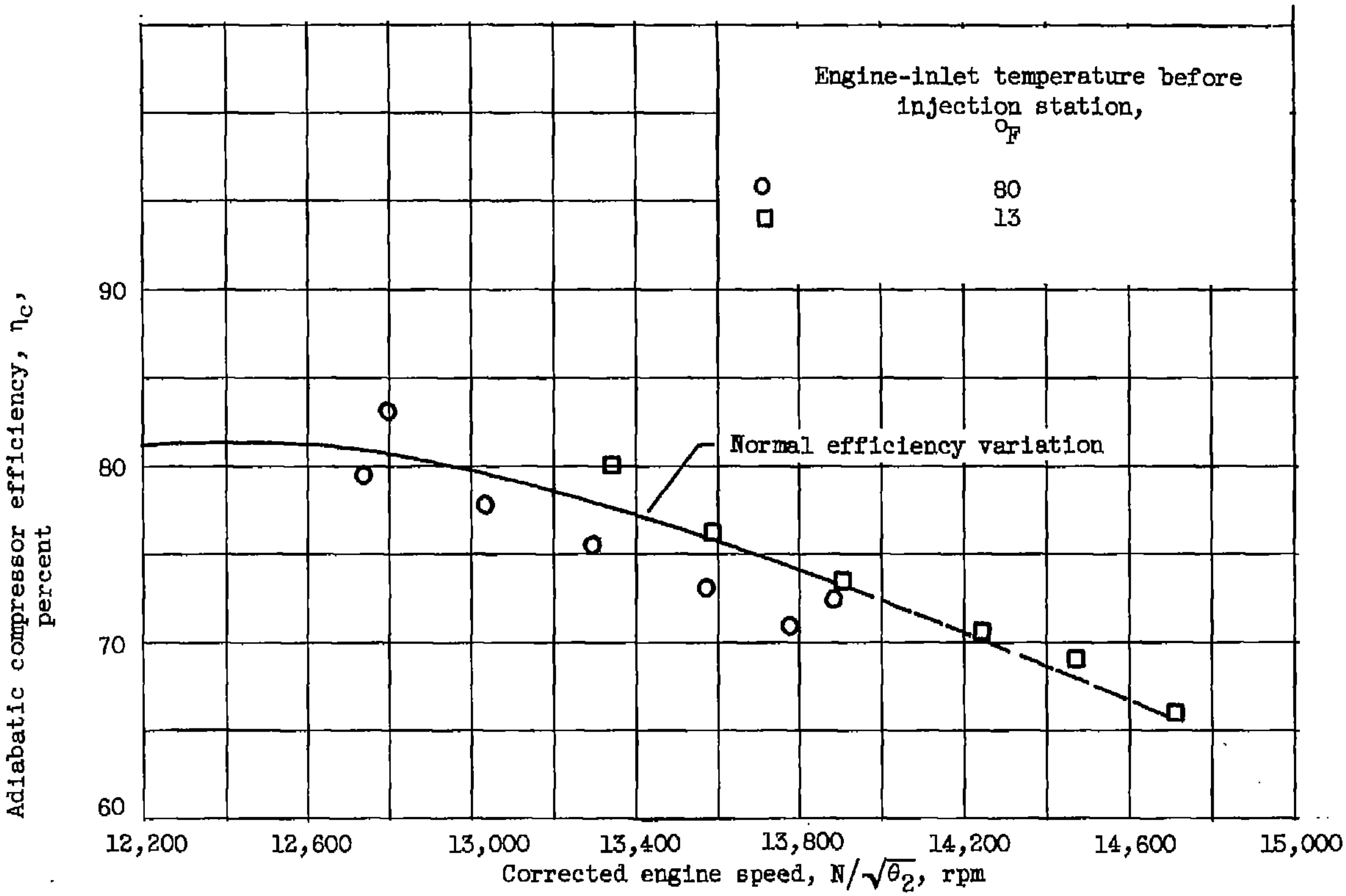


Figure 7. - Variation of adiabatic compressor efficiency with corrected engine speed compared with compressor efficiency calculated during injection of liquid ammonia at compressor inlet. Simulated flight Mach number of 1.0 at altitude of 35,000 feet.

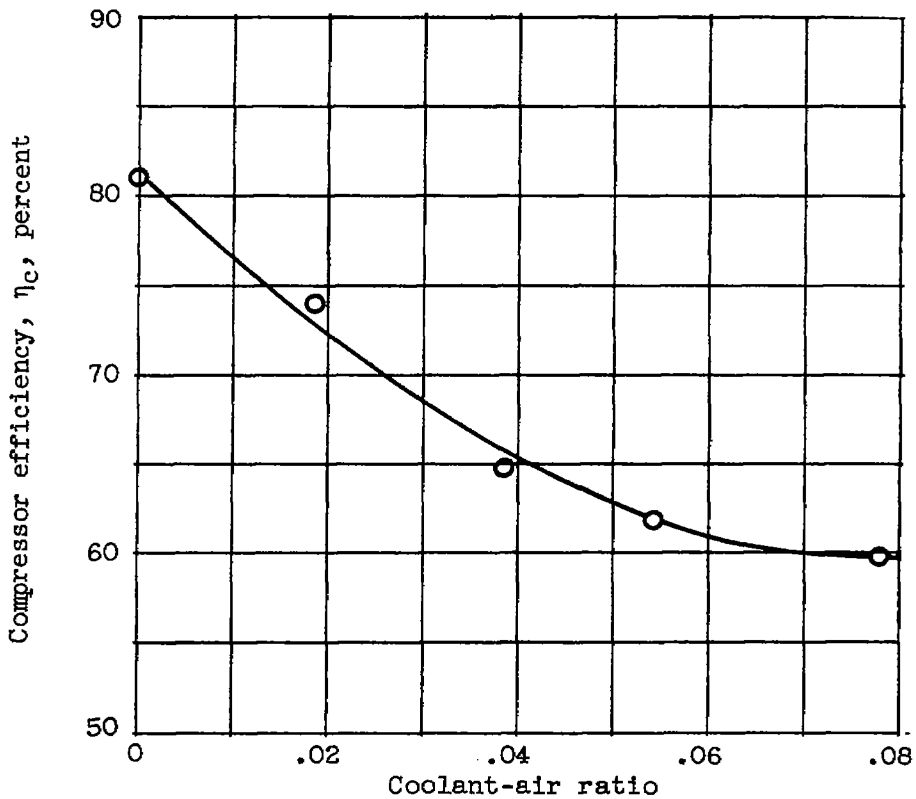
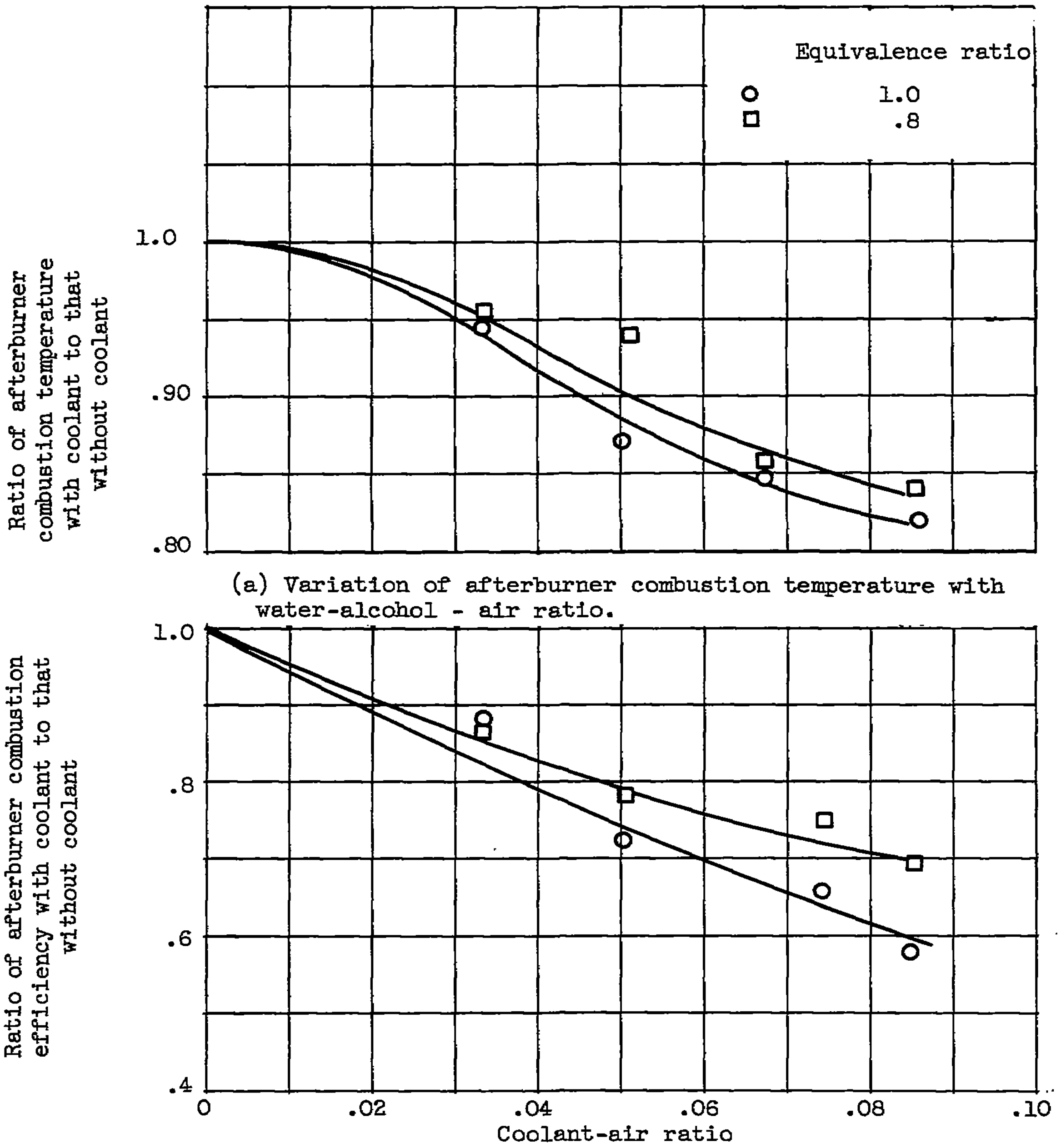
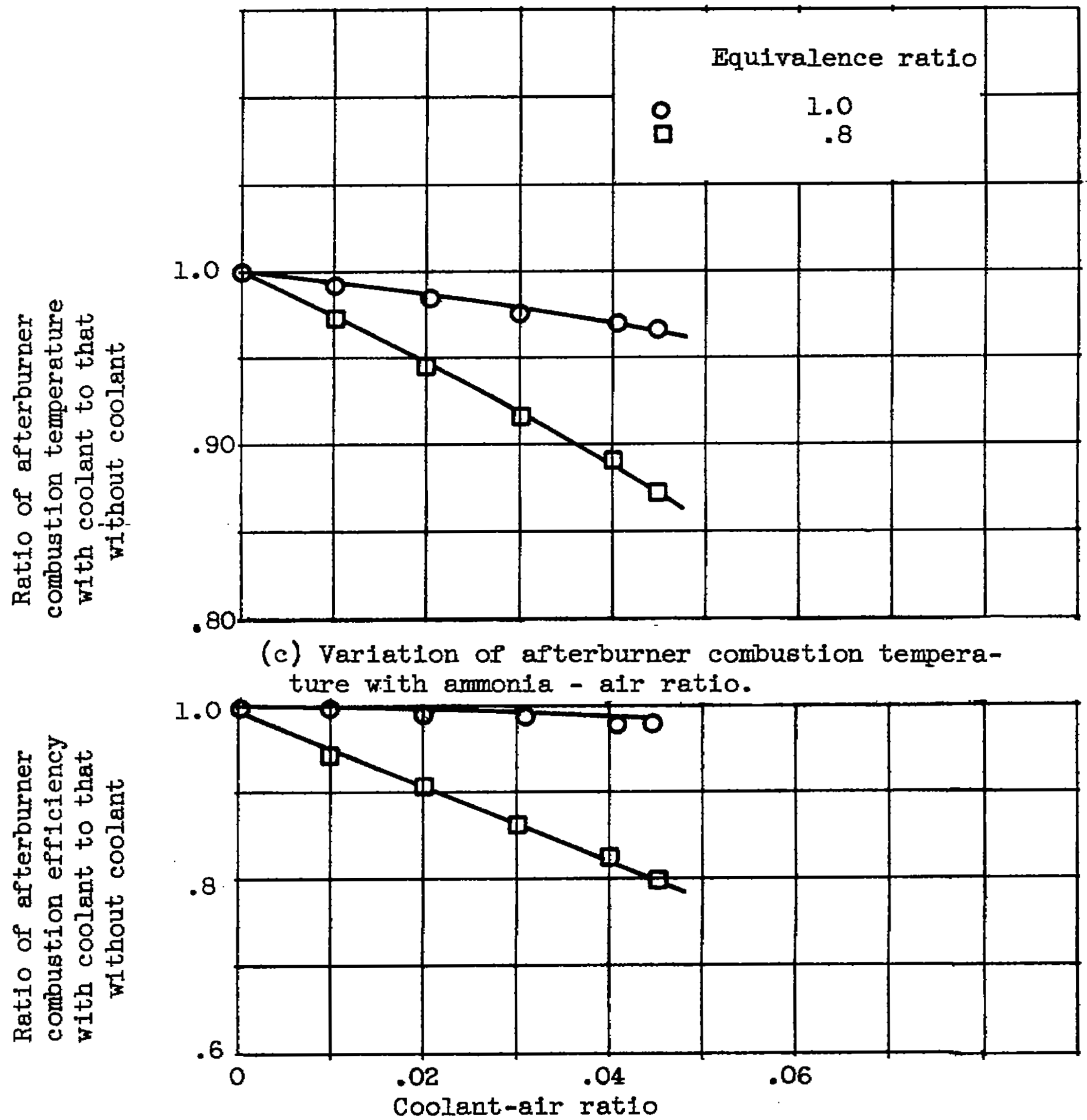


Figure 8. - Effect of coolant-air ratio on compressor efficiency during injection of 70 percent water - 30 percent alcohol mixture into compressor inlet during simulated flight at Mach number of 1.0 and altitude of 35,000 feet.



(b) Variation of afterburner combustion efficiency with water-alcohol - air ratio.

Figure 9. - Influence of coolant on afterburner performance during combined use of compressor coolant injection and afterburning.



(d) Variation of afterburner combustion efficiency with ammonia - air ratio.

Figure 9. - Concluded. Influence of coolant on afterburner performance during combined use of compressor coolant injection and afterburning.

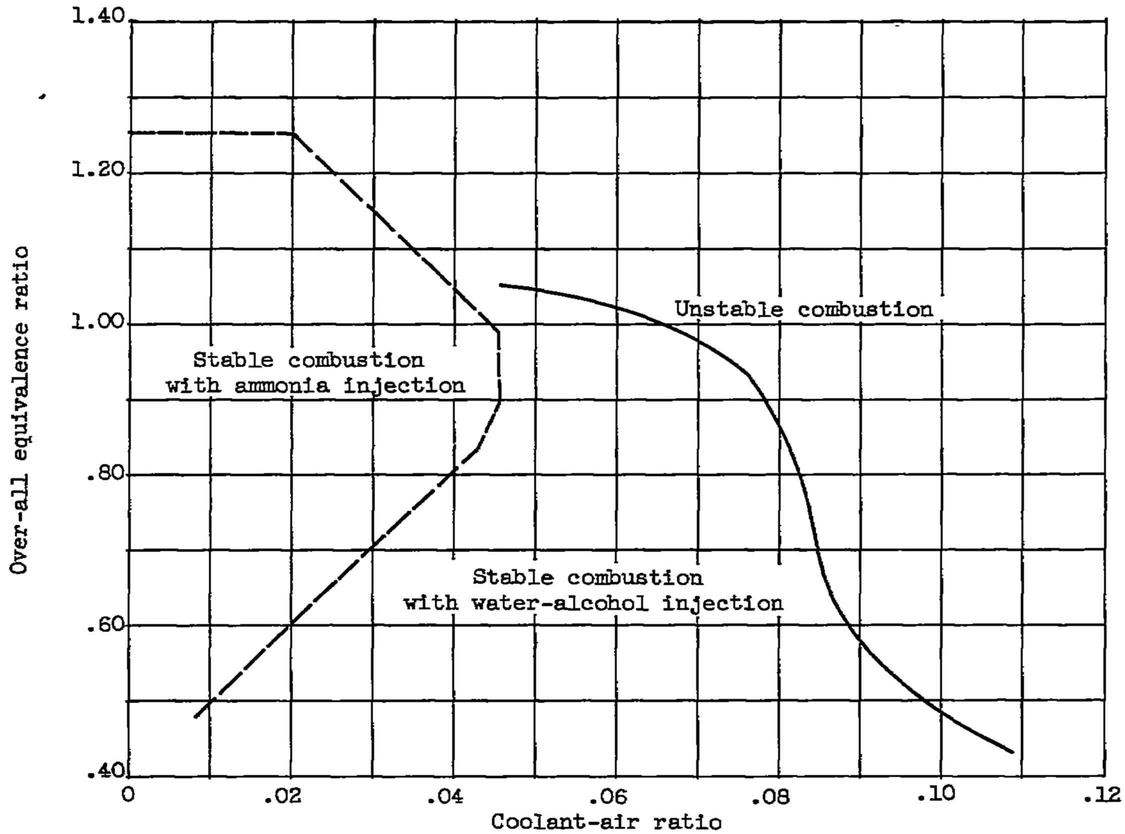


Figure 10. - Effect of compressor injection coolants on combustion stability of afterburner during combined use of compressor injection and afterburning. Water-alcohol injection at sea-level, zero ram; ammonia injection at 35,000 feet, flight Mach number 1.0.

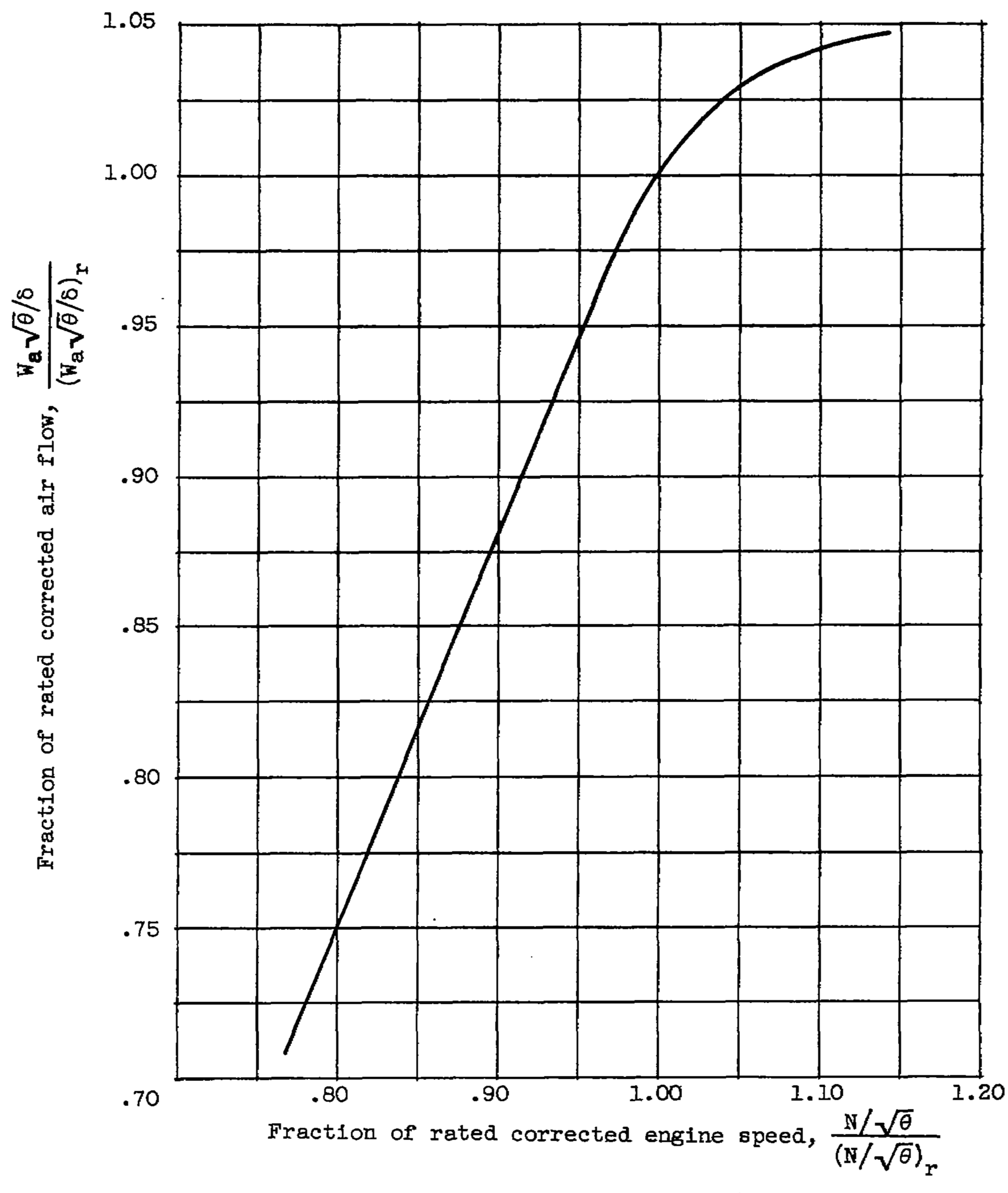


Figure 11. - Air-flow characteristics of hypothetical turbojet engine.

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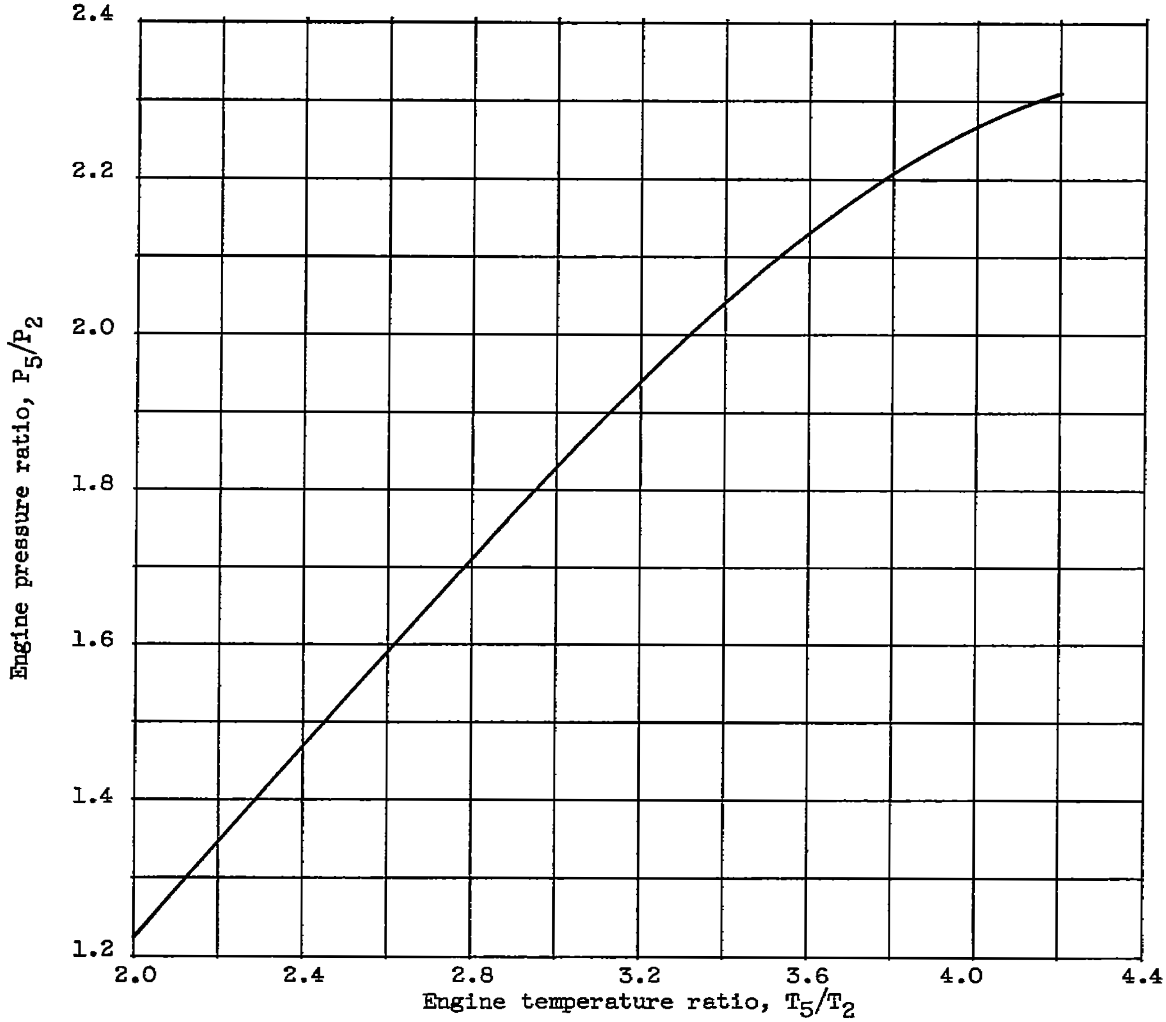

Figure 12. - Engine pumping characteristics of hypothetical turbojet engine operating at altitude of 35,000 feet.

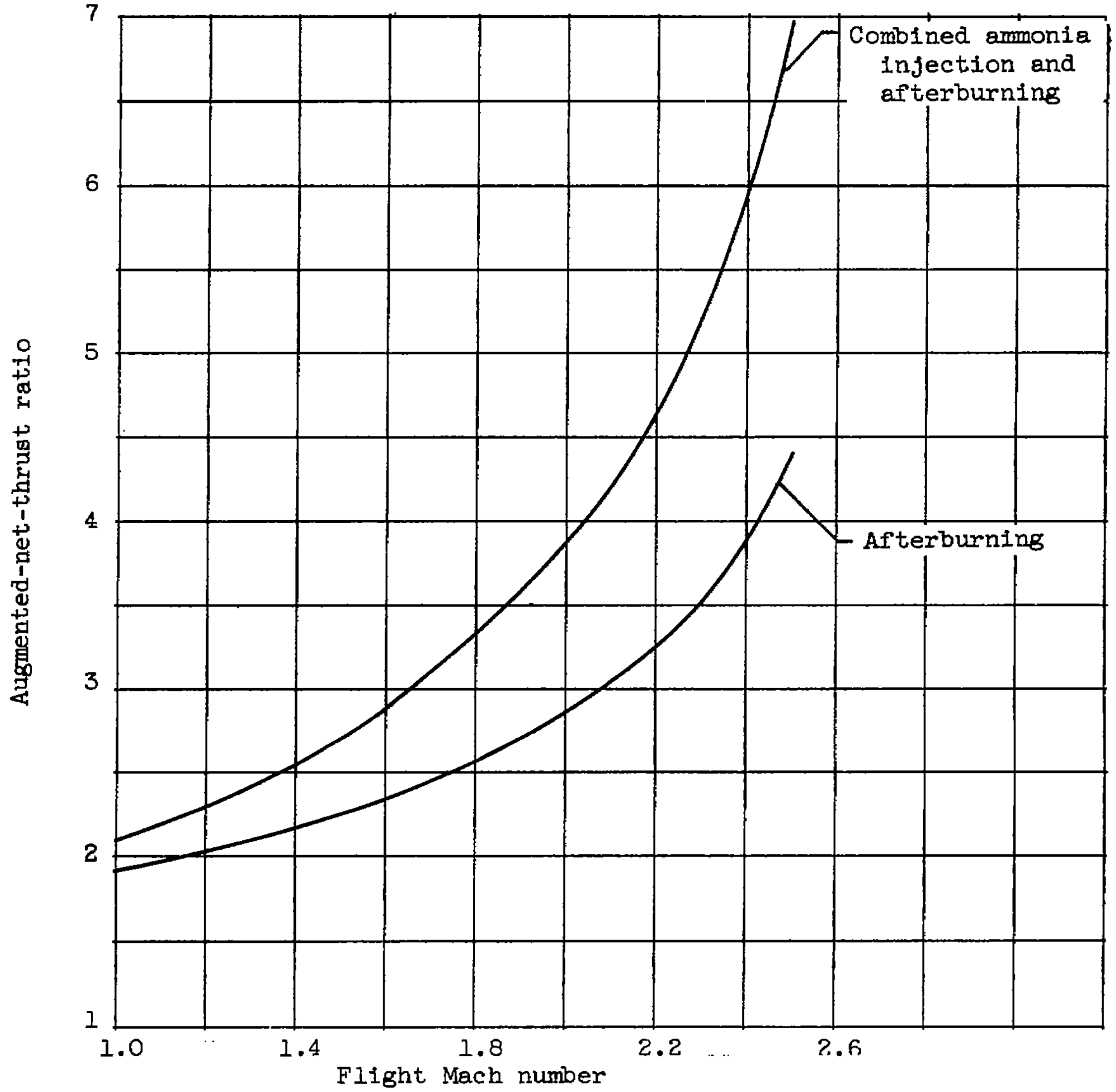

Figure 13. - Augmented net-thrust ratio calculated for hypothetical turbojet engine operating supersonically at altitude of 35,000 feet using compressor-inlet injection ammonia-air ratio of 0.04 and stoichiometric afterburning. Engine operating at rated speed and rated turbine-discharge temperature.

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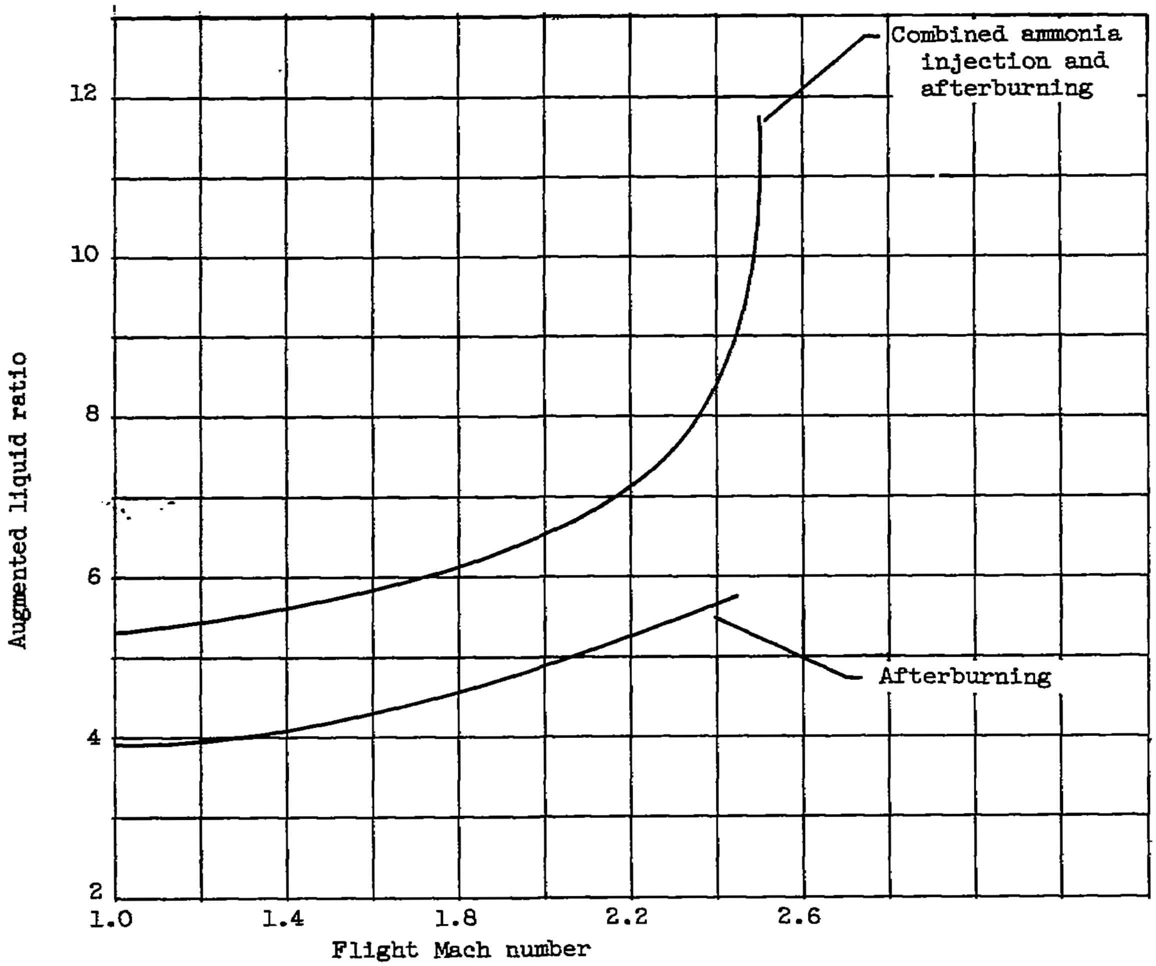
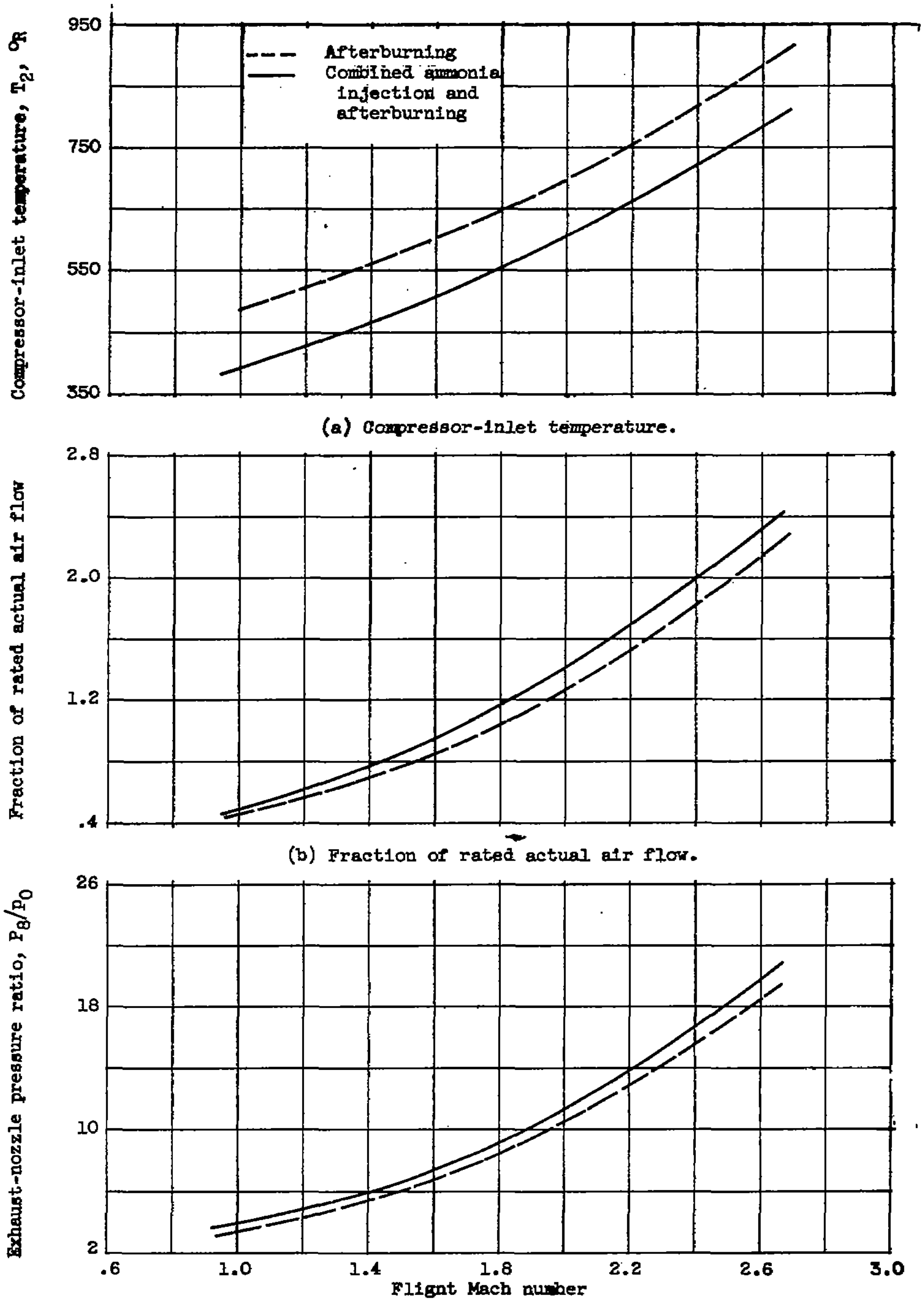


Figure 14. - Augmented liquid ratio calculated for hypothetical turbojet engine operating supersonically at altitude of 35,000 feet using compressor-inlet injection ammonia-air ratio of 0.04 and stoichiometric afterburning. Augmented liquid ratios shown are associated with augmented-thrust ratio data of figure 13.



(c) Exhaust-nozzle pressure ratio.

Figure 15. - Comparison of engine performance variables with use of compressor-inlet ammonia injection (ammonia-air ratio of 0.04) and stoichiometric afterburning for transonic and supersonic flight at altitude of 35,000 feet. Calculated for hypothetical turbojet engine.