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RESEARCH MEMORANDUM

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By James W. Useller, James L. Harp, Jr., and David B. Fenn

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

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SUMMARY

An experimental investigation was conducted in the altitude test chamber on an axial-flow turbojet engine operating at simulated transonic flight conditions at an altitude of 35,000 feet to determine the magnitude of the thrust augmentation and to ascertain the operational characteristics of a combined augmentation system. This system included the use of a high-performance afterburner (operating at approximately stoichiometric condition) and the introduction of liquid anhydrous ammonia into the compressor inlet (ammonia-air ratios from 0 to 0.055).

The maximum augmented net thrust ratio obtained, 2.13, resulted from a combination of a stoichiometric condition in the afterburner and the introduction of liquid ammonia into the compressor inlet to produce an ammonia-air ratio of 0.045. Maximum afterburner combustion efficiency and temperature occurred while the afterburner was operating stoichiometrically both with and without the ammonia injection. Combustion blowout of the engine and the afterburner was frequently encountered at ammonia-air ratios above 0.045. The combined use of ammonia injection and afterburning can be most favorably accomplished when high temperatures and relatively high combustion efficiencies prevail in the combustion zone, so as to reduce the influence of the ammonia on the combustion process. The aerodynamic drag of the ammonia-injection system caused a 1.3-percent total-pressure loss at the engine inlet. Stoichiometric operation of the afterburner was accompanied by an 11.7-percent total-pressure loss through the combustion chamber. The presence of ammonia in the combustion process had no effect on the afterburner pressure losses.

INTRODUCTION

The performance of a turbojet engine may be augmented by cooling the air either before or during the compression process and by adding heat in the exhaust stage of the basic thermodynamic cycle of operation. Although water and water-alcohol mixtures have been used extensively at sea level as the coolants for compressor cooling, reference 1 indicates that when such aqueous-base coolants were used in the engine under consideration in this investigation at the low ambient-air temperatures encountered with altitude flight, very limited thrust augmentation resulted. A search for a more effective coolant showed that liquid anhydrous ammonia would be adaptable to high-altitude flight conditions because of its relatively high heat of vaporization and low boiling point. An earlier investigation of this engine (ref. 2) reports that liquid ammonia as a compressor coolant at high-altitude flight conditions produces three to four times the thrust augmentation obtainable with a water-alcohol coolant. The combination of compressor coolant injection and afterburning at sea-level, zero-ram conditions produced a greater amount of thrust augmentation than could be provided by either method independently (ref. 3). However, the maximum performance of the combined system of reference 3 was restricted by the deleterious effect of the water vapor from the coolant on the combustion process in the afterburner. Thus, the use of an aqueous-base coolant presented a limitation to the range of operation of the afterburner. 2719

Because it was believed that ammonia, which in itself is combustible, might have less influence on the afterburner combustion process than the aqueous coolants and because its effectiveness as a coolant is good even at the low temperatures encountered at high altitudes, the present investigation was conducted to determine the magnitude of the thrust augmentation available and to ascertain the operational characteristics and limitations when using compressor-inlet ammonia injection and afterburning together. The engine was operated at a simulated transonic flight condition at an altitude of 35,000 feet and at several inlet-air temperatures to determine the effect of temperature on the augmentation. Liquid ammonia was introduced at the compressor inlet (ammonia-air ratios from 0 to 0.055) of an engine that employed a high-performance afterburner operating at approximately stoichiometric condition. The effect of the ammonia on the afterburner combustion temperature, efficiency, and stability is also reported.

APPARATUS AND INSTRUMENTATION

Engine. - The axial-flow turbojet engine used in this investigation (fig. 1) developed 3000 pounds thrust at sea-level, zero-ram conditions, with an average turbine-outlet gas temperature of 1625° R and a rated speed of 12,500 rpm. The primary-engine components included an 11-stage axial-flow compressor, a compressor-outlet mixer, a double-annular combustor of the through-flow type, a two-stage axial-flow turbine, a shrouded air-cooled afterburner, and a water-cooled variable-area exhaust nozzle. The compressor-outlet mixer produced a more favorable compressor-outlet velocity profile and consequently an improved turbine-outlet radial temperature distribution.

Compressor blade clearance indicators of NACA design were installed at the seventh to tenth stages of the compressor. These indicators warn of impending blade rubbing that may occur when the compressor casing is overcooled as a result of unvaporized coolants centrifuging to the outer casing. The engine was mounted on a movable thrust bed which was connected to a null-type thrust cell in the altitude test chamber as shown in figure 2. The fuel used in the primary-engine combustor was clear, unleaded gasoline (62-octane).

Afterburner. - The exhaust heat addition to the cycle of operation was provided by a high-performance afterburner based on a design evolved from the investigation reported in reference 4. Detailed dimensions of the afterburner used throughout this investigation are given on the cross-sectional sketch of figure 3. Most of the afterburner combustion-chamber length was air-cooled, while the exhaust nozzle and nozzle transition sections were water-cooled. Air for this afterburner cooling shroud was supplied from an outside source. The afterburner shell was constructed of 1/8-inch Inconel. Vortex generators were welded to the upstream end of the diffuser inner cone to prevent flow separation from the inner cone. The flame holder was of the three ring, V-gutter type, with a blocked area of 35 percent. A sketch of the flame holder is included in figure 3. The afterburner fuel-spray bars were located near the end of the inner cone and approximately 19 inches upstream of the flame holder. A total of 24 spray bars were equally spaced around the circumference of the afterburner, 12 long and 12 short tubes in alternate positions, as sketched in figure 3. The spray bars were constructed of 1/4-inch Inconel tubing flattened to a thickness of about 1/8 inch. Holes of 0.020-inch diameter were drilled in the flattened sides of the spray bars, thus ejecting fuel normal to the direction of gas flow. The orifices of the spray bars were spaced so as to be centered in equal annular areas.

Ignition of the afterburner was accomplished by the introduction of additional fuel at one location in the engine combustor so as to provide a hot streak through the turbine. The hot-streak method of afterburner ignition was used throughout this investigation. The fuel used in the afterburner was MIL-F-5624A, grade JP-4.

Ammonia-injection system. - One manifold (fig. 4(a)) containing 20 spray tubes was used to inject the liquid ammonia into the engine inlet-air stream approximately 68 inches upstream of the compressor inlet. This system of injection was designed to provide a more uniform distribution than that reported in the earlier work of reference 2, where a combination of spray bars and conical spray nozzles was used to introduce the ammonia into the air stream.

The spray tubes used in this investigation were constructed by flattening 1/4-inch-diameter circular stainless-steel tubing. Each spray tube contained 16 holes of 0.021-inch diameter that were radially spaced in pairs so as to introduce the ammonia at the centers of eight equal areas. In addition, each alternate tube had a 0.021-inch-diameter hole drilled in the end to permit penetration towards the center of the duct. A ring was welded to the outer ends of the tubes to provide additional support.

A schematic diagram of the ammonia-injection system is presented in figure 4(b). Liquid ammonia was supplied to the engine from a 5000-pound capacity storage tank, pressurized with high-pressure helium to approximately 400 pounds per square inch gage. Ammonia flow to the engine was controlled by a manually operated throttle valve. The ammonia used in this investigation was a commercial grade of liquid anhydrous ammonia.

Instrumentation. - Pressures and temperatures were measured throughout the engine at the stations designated in figure 1. The type of measurement made and the number of probes used at each station are tabulated in this figure. All probes were placed on centers of equal areas at each measuring station. Exhaust pressure in the altitude test chamber was measured in the plane of the exhaust-nozzle exit. Fuel flow was measured by means of direct reading rotameters and the ammonia flow by means of a remote indicating rotameter.

PROCEDURE

The flight condition simulated was an altitude of 35,000 feet and a flight Mach number of 1.0. Performance was investigated at inlet-air total temperatures of 13° (NACA standard for this flight condition), 80°, and 150° F in order to ascertain the effect of inlet-air temperature on the thrust augmentation. The engine was operated at rated conditions, a speed of 12,500 rpm, and an average turbine-outlet gas temperature of 1625° R. Turbine-outlet temperature was maintained at this value for all runs by adjustment of the variable-area exhaust nozzle. Engine performance was investigated over the following range of inlet-air temperatures, ammonia-air ratios, and over-all equivalence ratios (the actual fuel-mixture - air ratio compared with stoichiometric fuel-air ratio):

Inlet temperature, °F	Ammonia-air ratios	Nominal over-all equivalence ratios
13	0-0.050	0.8, 0.9, 1.0, 1.1, 1.2
80	0-0.054	1.0, 1.1
150	0-0.055	1.0, 1.1

The nomenclature used in the discussion is listed in appendix A; and the method of calculating thrust, afterburner temperatures, and efficiencies is described in appendix B.

RESULTS AND DISCUSSION

In the analysis of performance of the combined augmentation system, it was considered more enlightening to separate initially the consideration of the two basic methods of increasing engine performance. For this reason the performance of each system was investigated independently, and then the combined performance was studied with particular consideration given to the influence and limitations imposed by one system on the other. The engine performance data obtained are tabulated in table I.

Augmented Engine Performance

Effect of improved coolant-air mixing. - For the compressor coolant injection, which employed liquid anhydrous ammonia, it is shown in reference 2 that the cooling effectiveness obtained and the resultant thrust augmentation are directly related to the uniformity of mixing of the coolant and the air. The augmented thrust ratio obtained with the configuration of this investigation (B) and the configuration used in reference 2 (A) are compared in figure 5. Configuration A employed a combination of alternately positioned spray tubes and conical spray nozzles, while configuration B employed only spray tubes, as previously described. The improved distribution provided by configuration B produced an additional 5-percent increase in net thrust.

The uniformity of ammonia distribution achieved with configuration B is shown in figure 6(a), as recorded with a motion picture camera. Although the distribution appears to be good, further improvement is possible. Even with an ideally designed spray configuration, poor distribution of the coolant might result from either insufficient pressure in the supply manifold or from plugging of the spray orifices with foreign matter. A study of the ideal supply manifold pressure is beyond the scope of this investigation, but during an attempt to study the spray characteristics of configuration B by means of motion pictures, some plugging of the spray orifices was encountered. Figure 6(b) shows the poor distribution of the coolant that resulted from plugging of several orifices.

Afterburner performance. - The afterburner used in this investigation demonstrated satisfactory performance for a range of high altitudes and for a variation of fuel-air ratios at each altitude under consideration. The augmented net thrust ratio (defined as the ratio of the augmented net thrust to unaugmented net thrust) obtained with afterburning alone at an altitude of 35,000 feet and a flight Mach number of 1.0 is shown in figure 7. A maximum augmented net thrust ratio of 1.92 was obtained with an augmented liquid ratio (the total liquid consumption divided by the normal engine fuel flow) of 4.6.

Combined augmentation system. - The augmented performance obtained with the combined use of compressor ammonia injection and afterburning is shown in figure 8 for ammonia-air ratios from 0 to 0.045. Lines of constant ammonia-air ratio are shown for a range of afterburner fuel-air ratios. For each ammonia-air ratio, the augmented net thrust ratio increased with afterburner fuel-air ratio in a manner similar to the increase obtained with the afterburner operating alone. The maximum augmentation obtained with each ammonia-air ratio was reached at approximately an over-all equivalence ratio of 1.0 (stoichiometric mixture) in the afterburner, as is indicated on the curves. As the ammonia-air ratio was increased, the maintenance of a stoichiometric condition in the afterburner was accompanied by partial replacement of the hydrocarbon fuel with ammonia. Also, as the ammonia-air ratio was increased, the resultant specific liquid consumption increased rapidly, since ammonia is a less effective fuel than the hydrocarbon fuel, because of its lower heating value. The heating value of ammonia is approximately one-half that of the hydrocarbon fuel. The trend of the combined augmentation performance shown here is quite similar to that predicted in the analysis presented in reference 5, where the combination of afterburning and water-alcohol-mixture injection into the compressor is considered.

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Further increases in thrust augmentation by means of operation at higher afterburner fuel-air ratios were limited by the fact that increasing the fuel-air ratio in excess of stoichiometric produced a decrease in thrust augmentation because of a reduction in the combustion temperature. As an alternate means of achieving further increases in thrust augmentation, operation at higher ammonia-air ratios might be considered. Requirements of safe operation would have limited the use of ammonia to an ammonia-air ratio of 0.095, the combustible mixture ratio, if combustion blow-out had not presented a limitation at a lower ammonia-air ratio. Combustion blow-out occurred in both the engine and the afterburner when the ammonia-air ratio was increased to values between 0.050 and 0.060. At ammonia-air ratios of 0.040 and 0.045, combustion blow-out prevented operation of the afterburner in excess of a stoichiometric mixture. Reference 2 reported occasional blow-out of the engine combustor at ammonia-air ratios of 0.046, while operating at low engine inlet-air temperatures. Further discussion of the combustion blow-out encountered will be included in the section considering the influence of the coolant on afterburner combustion.

Variation of augmentation with engine inlet-air temperature. In order to illustrate the effect of engine inlet-air temperature variation on the combined augmentation system, data are presented in figure 9 for a range of ammonia-air ratios from 0 to 0.052 at inlet-air temperatures of 13° (NACA standard for this flight condition), 80°, and 150° F. The thrust ratio presented here is based on thrust produced by afterburning alone at stoichiometric condition in order to demonstrate the effect of increasing inlet-air temperature on the cooling portion of the combined system. The principal reason for the 13-percent increase in net thrust ratio shown in figure 9 for an increase in inlet-air temperature from 13° to 150° at an ammonia-air ratio of 0.045 was that, at higher temperatures and consequently lower corrected engine speeds, a given drop in temperature due to ammonia injection resulted in a greater percentage rise in the air flow and the engine pressure ratio. Secondly, the effectiveness of the coolant was increased somewhat when the ammonia, which was supplied at 80° F, was introduced into the 150° air stream, since the ammonia will produce some cooling because of this temperature difference without evaporation taking place. Increasing the engine inlet-air temperature increased the net thrust ratio substantially, and the data presented indicate that the ammonia coolant would become progressively more effective as the inlet temperature increased with flight Mach number.

Influence of Coolant on Afterburner Combustion

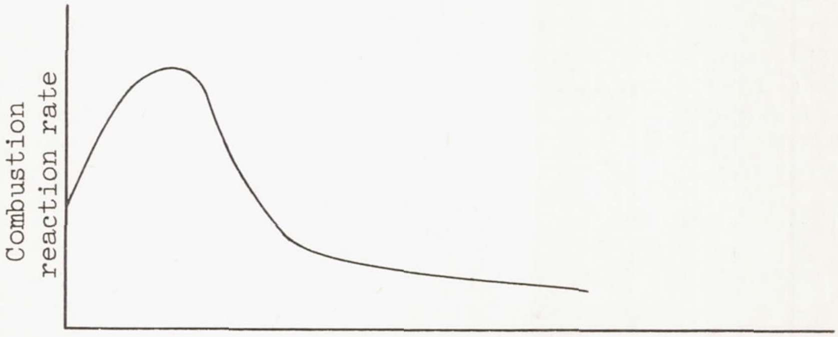
It may be expected that any coolant introduced into the compressor would exert some influence on the combustion performance of the afterburner. Reference 3 pointed out that with the addition of a water-alcohol-mixture coolant to air ratio of 0.074 into a stoichiometrically operating afterburner, the combustion temperature was lowered 700°, while the combustion efficiency was reduced from 90 to 58 percent. It has already been mentioned that the liquid ammonia used as the compressor coolant in this investigation provoked combustion blow-out of the afterburner and engine at high ammonia-air ratios; therefore, it may be suspected that the ammonia will also produce a change in the combustion temperature and efficiency achieved in the afterburner at lower ammonia-air ratios.

Combustion efficiency. - The variation of afterburner combustion efficiency as calculated for the range of equivalence ratios (0.8 to 1.2) investigated is shown in figure 10(a) for ammonia-air ratios from 0 to 0.045. For equivalence ratios between 0.7 and 0.9, the combustion efficiency decreased markedly as the concentration of ammonia was increased. The combustion efficiency as calculated herein was based on the ideal heat release of liquid ammonia. It is believed that the combustion temperatures associated with operation at equivalence ratios below 0.9 are not sufficient to dissociate the ammonia completely and would thus cause the calculated combustion efficiencies to become progressively lower as the quantity of undissociated ammonia was increased. In addition, the undissociated ammonia exerts a negative influence on the combustion process which also tends to decrease the combustion efficiency as the ammonia concentration is increased. The nature of this influence will be postulated in a discussion of the effect of ammonia on the combustion reaction rate. In the region of stoichiometric operation, combustion efficiency was unaffected by the ammonia. The maximum combustion efficiency obtained was 91 percent at an equivalence ratio of 1.02. At the high temperature associated with stoichiometric operation, the dissociation of ammonia into more readily combustible products reduced the influence of the ammonia on the combustion process. Above stoichiometric condition, the combustion efficiency was only slightly affected by an increase in ammonia concentration for the range of equivalence ratios investigated herein. With the reduction in combustion temperature encountered with rich operation (equivalence ratio greater than 1.0), some of the ammonia remained undissociated and thus had a detrimental effect on the flame propagation and combustion efficiency. 2719

Combustion temperature. - The combustion temperatures associated with the combustion efficiencies discussed previously are shown in figure 10(b). The combustion temperature peaked more sharply with the addition of ammonia, although the maximum temperature occurred in the region of stoichiometric operation for all ammonia-air ratios investigated. The reduction of the combustion temperature was the result of three distinct influences: As the engine inlet-air temperature is reduced with increasing ammonia concentration, it is necessary to reduce the afterburner heat release to maintain a constant over-all equivalence ratio, since the engine heat release must be increased if a constant turbine-outlet gas temperature is to be maintained. Secondly, the theoretical flame temperature of undissociated ammonia (3560° R, ref. 6) is somewhat below that of the hydrocarbon fuel (3920° R). The theoretical flame temperature obtainable from a mixture of ammonia and hydrocarbon fuel would be an average of the individual flame temperatures weighted according to the Le Chatelier mixture law. Increasing the ammonia concentration would thus lower the average gas temperature of the mixture. Finally, the reduced combustion efficiency caused by the influence of the undissociated ammonia on the combustion process would tend to lower the combustion temperature. With the complete dissociation of ammonia encountered with stoichiometric operation, the latter effect is minimized or eliminated. The decrease in combustion temperature above stoichiometric results from the quenching effect of the excess fuel. With this cooling, some of the ammonia is permitted to remain undissociated and thus can influence the temperature and the propagation of the flame.

Combustion reaction rate. - An examination of the kinetics of the combustion process will provide an insight into how the ammonia affects the combustion of the hydrocarbon fuel. Certain additives, because of their molecular structure, can exert a negative influence by reducing the flame propagation rate. Early combustion studies have demonstrated that water exhibits this faculty, and it is pointed out in reference 3 that water vapor reduced the combustion performance appreciably.

Evidence has been found (refs. 7 and 8) that with the addition of ammonia to the combustion process, the rate of flame propagation is remarkably dependent on the concentration of the ammonia present and is the result of an auto-catalytic reaction between the ammonia and the propagating agents. It has been demonstrated in reference 8 that for any given temperature there exists a concentration of ammonia for which ammonia radicals and oxidation products of the radicals assist the propagating agents in the combustion reaction, which results in a maximum reaction rate. At concentrations of ammonia less than that required for the maximum reaction rate, the ammonia tends to accelerate the chain-reaction propagation; while at concentrations greater than optimum, the ammonia acts so as to abate the chain reaction and thus reduce the propagation rate of the flame. That is, ammonia exhibits both a positive and a negative influence on the reaction rate, depending on the concentration present during combustion, as is shown in the following sketch:



It would thus appear that with respect to flame propagation there exists an optimum ammonia-air ratio for operation of the combined augmentation system. The early experimental data of references 7 and 8 are only qualitatively applicable, since they were obtained with fuels differing from those employed in the afterburner and were investigated over a different temperature range. It is believed that the optimum ammonia concentration with respect to combustion is much less than that which produced appreciable thrust augmentation. That is, all operation during this investigation was at ammonia concentrations greater than that for maximum combustion reaction rate.

Combustion stability. - With the marked effect of ammonia on the combustion temperature and efficiency, it is not surprising that the combustion stability would also be influenced. The combustion stability limits were characterized by combustion blow-out. The limit of stable operation included an increasingly narrower range of equivalence ratios as the ammonia-air ratio was increased, as may be seen in figure 11. Afterburner blow-out threshold data shown in figure 11 were obtained under conditions that were relatively unstable, but combustion continued long enough to permit collection of the test data. The combustion blow-out points shown were obtained at conditions that caused combustion cessation shortly after the indicated ammonia-air ratio was established. Stoichiometric operation (equivalence ratio equal to 1.0) permitted the use of the widest range of ammonia-air ratios, while only the lowest ammonia-air ratio allowed the afterburner to be operated over a wide range of equivalence ratios. Operation was not possible at ammonia-air ratios of 0.05 or above even at the stoichiometric condition. Experimental data obtained by means of high-speed motion pictures (ref. 9) indicate that combustion blow-out normally requires 0.025 second to take place. When the ammonia was present, the combustion blow-out differed from the normal type, in that the reaction was delayed considerably longer during most of the blow-outs experienced. That is, after the flame appeared to be stablized for a period of time, the turbine-outlet gas temperature would suddenly begin to increase without further changes in the fuel flow being supplied to the afterburner. The rise in turbine-outlet gas temperature continued for a period of time from 1/2 to 1 minute and culminated in combustion blow-out. Operation in the band indicated in figure 11 was considered unstable, since combustion blow-out was either imminent or actually occurring.

Performance Losses

The performance losses associated with any proposed system of thrust augmentation would influence the application of such a system. The losses associated with the combined augmentation system considered herein are not believed to be excessive. The ammonia-injection spray tubes, the supply manifold, and the surface friction of the 68 inches of inlet duct between the supply manifold and the compressor inlet caused a 1.8-percent loss in inlet total pressure. The injection spray system, including the manifold, was responsible for a total-pressure loss of 1.3 percent. The injection system used in this investigation was adapted from existing equipment and is not representative of a minimum pressure-loss design. The afterburner total-pressure loss due to friction encountered with flow in the components and to momentum pressure loss associated with heat addition to a flowing stream during combustion of the ammonia and hydrocarbon fuel is shown in figure 12 for the range of over-all equivalence ratios investigated. A total-pressure loss of 11.7 percent of the turbine-discharge pressure was measured for the afterburner operating at a stoichiometric condition. The drag losses of the afterburner components without afterburning accounted for a 5.8-percent loss in total pressure. No relation existed between the afterburner total-pressure loss and the ammonia concentration present.

Operational Problems

Automatic control of engine variables. - The complexity of operation of a high-speed airplane has necessitated the introduction of automatic control devices to regulate many of the variables of the engine system. For this reason, it is appropriate that some cognizance be given to the control problem involved in the application of the combined augmentation system discussed herein. The reduction of engine and afterburner fuel flows experienced with increasing ammonia-air ratio and with change in exhaust-nozzle area necessary to maintain rated turbine-outlet temperature for these conditions is shown in figure 13. The 31-percent reduction in engine fuel flow encountered with an increase of the ammonia-air ratio from 0 to 0.045 with the afterburner operating stoichiometrically is within the range of the engine fuel-flow reduction necessary for climb from sea level to high-altitude flight. Current control devices can easily accomplish the 5-percent reduction in exhaust-nozzle area required to maintain rated turbine-outlet temperature. The reduction of afterburner fuel flow with increasing ammonia flow necessary for maintenance of a stoichiometric condition in the afterburner cannot be achieved with the same degree of facility as the other two variables. In addition to the range of change necessary because of the ammonia, a variation in combustion efficiency with altitude would introduce a further complication. In spite of the latter limitation, it is believed that the variations required in engine fuel flow, afterburner fuel flow, and exhaust-nozzle area with the introduction of ammonia into the combustion process are within the realm of present automatic control devices.

Operating experience. - The engine and the afterburner were operated for approximately $11\frac{1}{2}$ hours with ammonia injection into the compressor inlet, with no apparent engine or afterburner component deterioration. During this period, approximately 16,000 pounds of liquid ammonia were injected into the compressor inlet. Following 140 hours of operation of the engine, two of the ammonia spray tubes failed from vibrational fatigue. One was lodged at the compressor inlet, while the other passed through the engine with only minor damage. A supporting spray-tube ring added to the original design to preclude further failure is shown in figure 4(a). Special precautions were taken to eliminate all copper and copper-alloy piping and equipment in the ammonia-injection system, and no corrosion by the ammonia was encountered. No modifications were made to standard engine components.

Although no increased sensitivity of control was noticed in the operation of the basic engine when the ammonia was introduced, the afterburner was found to be sensitive to the rate at which the ammonia flow was increased. Rapid increases in ammonia flow produced unstable operation of the afterburner and at times culminated in afterburner combustion blow-out. When high ammonia flow rates were being used, it was especially important that increases of the ammonia flow be accomplished slowly to prevent blow-out.

CONCLUDING REMARKS

Maximum thrust augmentation was achieved at a stoichiometric condition in the afterburner, combined with a compressor-inlet ammonia-air ratio of 0.045. The maximum augmented net thrust ratio obtained at a simulated altitude of 35,000 feet and Mach number of 1.0 was 2.13 as compared with 1.92 for afterburning alone. The peak afterburner combustion efficiency and combustion temperature were achieved near stoichiometric condition in the afterburner for all ammonia-air ratios investigated. At equivalence ratios below 0.9, the combustion temperature was reduced as the ammonia concentration was increased, because of the lower theoretical flame temperature of the ammonia. As the combustion temperature increased with equivalence ratios equal to 1.0, the ammonia was dissociated into more readily combustible products and the influence of the ammonia on the combustion performance was less. It is evident, therefore, that the ammonia is able to exert less influence on the combustion when high temperatures exist to cause dissociation of the ammonia.

Combustion blow-out of both the engine and the afterburner was frequently encountered at ammonia-air ratios above 0.045. Further increases in thrust augmentation by increasing the ammonia-air ratio appear unlikely without improvement of the combustion efficiency of the afterburner either by modification of the aerodynamic and fuel-spray configurations or the addition of some fuel additive or combustion catalyst that will minimize the negative influence of the ammonia on the flame propagation rate.

The aerodynamic drag of the ammonia-injection system caused a 1.3-percent total-pressure loss at the engine inlet, while stoichiometric operation of the afterburner was accompanied by an 11.7-percent total-pressure loss through the afterburner combustion chamber. The drag losses of the afterburner components without afterburning accounted for a 5.8-percent loss in total pressure. It is believed that the total-pressure loss caused by the ammonia-injection system as used in this investigation could be reduced, since the configuration used was adapted from existing equipment and is not representative of a minimum pressure-loss design.

Although the variation of engine fuel flow and exhaust-nozzle area necessary to maintain rated conditions during the use of the ammonia-injection system is not considered excessive, the variation required in the afterburner fuel flow due to the replacement of the hydrocarbon fuel by the ammonia could complicate the problem of automatic control of these variables. In spite of this difficulty it is believed that automatic regulation of these variables is within the realm of current devices.

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

APPENDIX A

NOMENCLATURE

The following symbols are used in this report:

A	cross-sectional area, sq ft
C_d	coefficient of discharge
C_t	coefficient of thermal expansion
C_V	effective velocity coefficient
d	ammonia-air ratio, W_{NH_3}/W_a
F_j	jet thrust
F_n	net thrust
g	acceleration due to gravity, 32.17 ft/sec^2
H^o	sensible enthalpy and chemical energy, Btu/lb
l	total liquid-air ratio
M	Mach number
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
R	gas constant, $\frac{1546 \text{ ft-lb}}{(\text{molecular weight})(\text{lb})(^{o}\text{R})}$
S	over-all equivalence ratio
T	total temperature, ^{o}R
V	velocity, ft/sec
W_a	air flow, lb/sec
W_f	fuel flow, lb/hr
W_g	gas flow, lb/sec

W_{NH_3} ammonia flow, lb/hr

γ ratio of specific heats

δ_{alt} engine-inlet total pressure/NACA standard total pressure associated with altitude and flight Mach number

η combustion efficiency

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Subscripts:

b afterburner

e engine

g gas

m fuel-manifold conditions

max maximum

n exhaust-nozzle throat

T total temperature

Numbered subscripts as indicated on figure 1.

APPENDIX B

METHODS OF CALCULATION

Flight Mach number. - The simulated flight Mach number was calculated from the following relation with complete ram recovery at the engine inlet assumed:

$$M_0 = \sqrt{\frac{2}{\gamma_1 - 1}\left[\left(\frac{P_1}{p_0}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]} \tag{1}$$

where γ for the inlet air was assumed to be 1.4.

Gas flow. - Engine inlet-air flow was calculated from the substitution of pressure and temperature measurements taken at station 1 into the following equation:

$$W_{a,1} = A_1 p_1 \sqrt{\frac{g}{R_1 T_1}} \sqrt{\frac{2\gamma_1}{\gamma_1 - 1}\left(\frac{P_1}{p_1}\right)^{\frac{\gamma-1}{\gamma}}\left[\left(\frac{P_1}{p_1}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]} \tag{2}$$

The gas flow at the exit of the afterburner was then determined from

$$W_{g,8} = W_{a,1} + \frac{W_{NH_3} + W_{f,e} + W_{f,b}}{3600} \tag{3}$$

Over-all equivalence ratio. - The term equivalence ratio is used to designate the fraction of stoichiometric combustion of a mixture of several fuels. Since each of the fuels might have a different fuel-air ratio for stoichiometric combustion, the commonly used fuel-air-ratio term is inadequate to describe the combustible mixture. A stoichiometric fuel-air mixture has an equivalence ratio of unity, while a rich fuel-air mixture has an equivalence ratio greater than unity. The over-all equivalence ratio of the combined cycle was calculated as follows:

$$S = 15.04 \frac{W_{f,e}}{3600\ W_{a,1}} + 14.81 \frac{W_{f,b}}{3600\ W_{a,1}} + 6.089 \frac{W_{NH_3}}{3600\ W_{a,1}} \tag{4}$$

where 15.04 is the stoichiometric air-fuel ratio of the engine fuel, 14.81 is the stoichiometric air-fuel ratio of the afterburner fuel, and 6.089 is the stoichiometric air-fuel ratio of the ammonia.

Afterburner combustion temperature. - The combustion temperature was determined from the total- and static-pressure measurements, the gas flow, and the flow area at station 8, with the following form of the continuity equation:

$$T_8 = \frac{2g}{R_8}\left(\frac{A_8 C_d C_t p_8}{W_{g,8}}\right)^2 \left(\frac{\gamma_8}{\gamma_8 - 1}\right)\left(\frac{P_8}{p_8}\right)^{\frac{\gamma_8 - 1}{\gamma_8}}\left[\left(\frac{P_8}{p_8}\right)^{\frac{\gamma_8 - 1}{\gamma_8}} - 1\right] \tag{5}$$

The water-cooled flow area at station 8 was assumed equal to the measurement obtained at room temperature. The flow coefficient $C_d C_t$ was assumed to be unity. The gas constant $R_{g,8}$ was determined from the composition of the exhaust gases and the hydrogen-carbon ratios of the fuels by the method of reference 10 from values based on reference 11. A water-gas reaction constant of 3.8 was assumed for mixtures richer than stoichiometric.

Afterburner combustion efficiency. - The combustion efficiency of the afterburner is defined as the ratio of the increase in energy in the afterburner to the ideal energy increase obtainable from the same quantity of fuel:

$$\eta_b = \frac{W_{g,8}\, H^o_{g,8} - W_{g,5}\, H^o_{g,5} - \frac{W_{f,b}}{3600}\, \lambda^o_{b,m}}{W_{g,8}\, H^o_{g,T_{max}} - W_{g,5}\, H^o_{g,5} - \frac{W_{f,b}}{3600}\, \lambda^o_{b,m}} \tag{6}$$

The term H^o is the sum of the sensible enthalpy and the chemical energy of the gas and was determined from the composition of the gas and the hydrogen-carbon ratios of the fuel by the method given in reference 10 from values based on reference 11, when no dissociation is assumed. A water-gas reaction constant of 3.8 was assumed for mixtures greater than stoichiometric. The value of T_{max} was determined from the ideal combustion temperature modified by a temperature difference to account for the increase in chemical energy in the products of combustion due to the effect of dissociation. The value of this temperature difference was determined as a function of afterburner fuel-air ratio based on data contained in reference 12. The term λ^o accounts for the difference between the H^o of the carbon dioxide and that of the water vapor in the burned mixture and the H^o of the oxygen removed from the air by their formation.

Thrust. - The jet thrust was determined from the gas flow, the combustion temperature, and the effective jet velocity by means of the following relations:

$$F_j = C_V \left[\frac{W_{g,8}}{g} V_n + A_n \ (p_n - p_0) \right] \tag{7}$$

$$F_j = C_V \ W_{g,8} \ \frac{V_{ef}}{\sqrt{gR_8T_8}} \ \sqrt{\frac{R_8}{g} T_8} \tag{8}$$

where

$$\frac{V_{ef}}{\sqrt{gR_8T_8}} = \frac{V_n}{\sqrt{gR_8T_8}} + \frac{p_nA_n}{\frac{W_{g,8}}{g} \sqrt{gR_8T_8}} - \frac{p_0A_n}{\frac{W_{g,8}}{g} \sqrt{gR_8T_8}} \tag{9}$$

The parameters of the effective velocity term obtained from reference 13 were used for ease of calculation.

The net thrust was calculated by deducting the inlet momentum from the jet thrust:

$$F_n = F_j - \frac{W_{g,1}}{g} V_0 \tag{10}$$

The net thrust was adjusted to standard flight conditions by means of the δ_{alt} factor to reduce the data scatter introduced by minor variations in setting test conditions. The augmented net thrust ratio was defined as the net thrust obtained from the combined augmentation cycle divided by the net thrust obtained from the unaugmented engine including the afterburner configuration, operating at rated engine speed of 12,500 rpm and limiting turbine-discharge temperature of 1625° R. The augmented liquid ratio was defined as the total liquid consumption (engine and afterburner fuels plus the ammonia) divided by the normal engine fuel flow.

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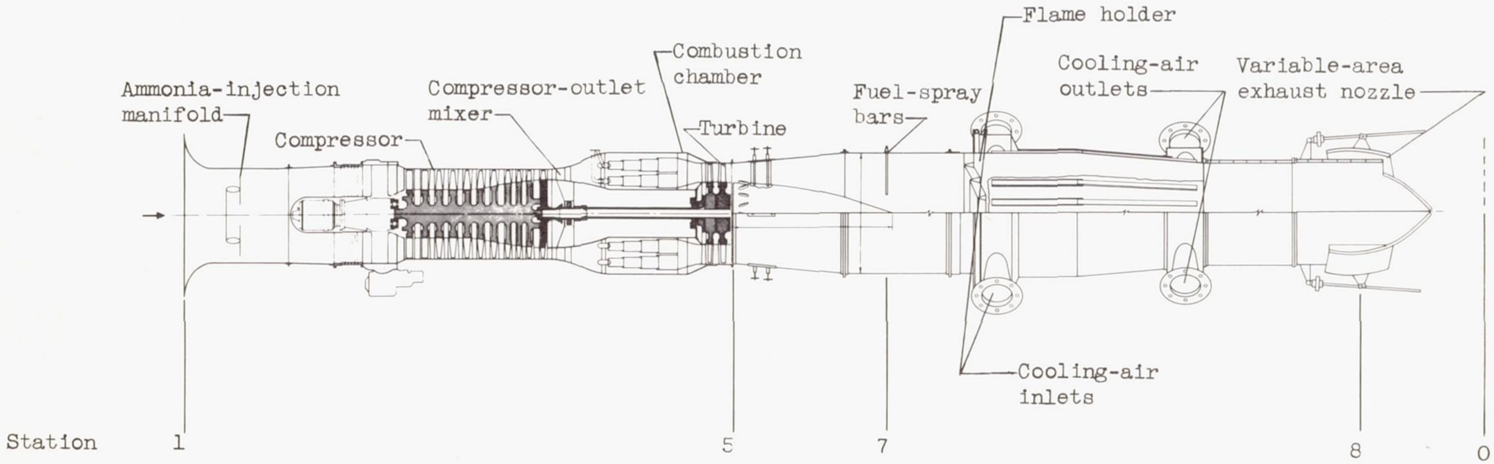
TABLE I. - COMBINED AUGMENTATION BY AMMONIA INJECTION AND AFTERBURNING

[Altitude, 35,000 feet; flight Mach number, 1.0; engine speed, 12,500 rpm; turbine-outlet gas temperature, 1625° R.]

Run	Engine-inlet total pressure, P_1, lb/sq ft	Engine-inlet total temperature, T_1, °R	Engine inlet-air flow, $W_{a,1}$, lb/sec	Ammonia-air ratio, d	Engine fuel flow, $W_{f,e}$, lb/hr	Afterburner fuel flow, $W_{f,b}$, lb/hr	Over-all equivalence ratio, S	Total liquid-air ratio, ι	Turbine-outlet pressure, P_5, lb/sq ft abs	Tail-pipe total pressure, P_8, lb/sq ft abs	Tail-pipe static pressure, p_8, lb/sq ft abs	Combustion temperature, T_8, °R	Afterburner combustion efficiency, η_b, percent	Jet thrust, F_j, lb	Altitude-adjusted net thrust, F_n/δ_{alt}, lb
1	939	473	27.61	0	1651	3720	0.80	0.054	1780	1590	1262	3232	0.804	3031	2203
2	939	472	28.08	.014	1448	3415	.80	.063	1844	1656	1328	3194	.806	3154	2312
3	942	473	27.87	.020	1398	3230	.81	.066	1863	1676	1356	3198	.811	3160	2317
4	943	474	28.28	.024	1345	3090	.80	.068	1876	1690	1373	3094	.773	3172	2313
5	932	475	28.08	.029	1270	2935	.80	.071	1884	1699	1385	3105	.785	3173	2348
6	937	471	28.69	.033	1253	2760	.78	.072	1930	1748	1447	3052	.805	3247	2393
7	937	466	28.84	.038	1200	2540	.77	.0740	1932	1752	1453	2990	.773	3247	2393
8	942	473	27.21	0	1651	4440	.93	0.062	1773	1567	1166	3642	0.909	3181	2357
9	936	474	27.52	.013	1441	4210	.93	.070	1811	1604	1203	3574	.897	3264	2447
10	940	473	28.48	.020	1385	3850	.88	.071	1906	1712	1337	3483	.891	3409	2552
11	943	477	27.76	0	1660	4460	.91	.061	1783	1584	1196	3491	.847	3185	2339
12	945	476	28.26	.015	1460	4100	.90	.069	1855	1649	1268	3417	.837	3296	2430
13	944	474	28.32	.020	1360	3700	.86	.070	1874	1672	1316	3297	.811	3267	2404
14	948	472	28.41	.024	1335	3750	.89	.074	1888	1688	1325	3322	.813	3318	2444
15	950	471	28.51	.029	1281	3600	.88	.077	1915	1716	1356	3322	.820	3359	2477
16	944	471	28.41	.034	1238	3420	.88	.079	1921	1716	1356	3313	.824	3366	2503
17	946	472	28.60	.039	1170	3280	.88	.082	1925	1712	1345	3265	.808	3377	2501
18	937	472	28.26	.044	1131	3150	.89	.086	1916	1702	1333	3297	.821	3376	2535
19	938	472	28.06	.050	1112	3000	.91	.091	1935	1721	1357	3315	.827	3377	2539
20	934	472	27.98	0.015	1441	4570	0.98	0.075	1837	1625	1202	3574	0.890	3356	2533
21	942	472	28.11	.020	1385	4480	.98	.078	1853	1631	1197	3596	.908	3394	2544
22	938	472	27.84	.025	1335	4435	1.01	.083	1857	1632	1190	3655	.942	3418	2586
23	946	472	28.37	.030	1270	4270	.98	.084	1873	1641	1207	3494	.876	3411	2543
24	939	472	28.08	.034	1237	4100	.99	.087	1906	1681	1264	3566	.919	3465	2624
25	934	472	28.20	.040	1175	3970	1.00	.091	1882	1669	1236	3530	.910	3481	2652
26	937	472	28.65	.044	1142	3720	.97	.092	1923	1698	1295	3343	.827	3464	2611
27	956	473	28.11	0	1640	5050	0.98	0.066	1804	1596	1153	3678	0.909	3320	2434
28	942	474	28.07	.015	1425	5020	1.04	.079	1838	1610	1157	3601	.896	3385	2535
29	948	474	28.13	.020	1390	4560	1.00	.079	1882	1659	1215	3718	.962	3469	2601
30	942	472	28.30	.024	1318	4180	.95	.078	1866	1656	1228	3588	.918	3435	2579
31	945	472	28.59	.030	1265	4210	.98	.083	1884	1675	1240	3551	.904	3486	2614
32	956	472	28.77	.034	1265	4050	.97	.086	1967	1742	1328	3565	.919	3540	2631
33	949	473	28.60	.038	1208	3930	.97	.088	1936	1745	1346	3505	.897	3511	2626
34	945	473	28.76	.0431	1150	3780	.97	.091	1929	1755	1352	3482	.892	3550	2672
35	948	475	27.48	0	1640	4845	.97	.066	1777	1568	1149	3637	.892	3216	2368
36	949	475	28.15	0.015	1448	5365	1.09	0.082	1853	1611	1150	3573	0.884	3397	2525
37	959	475	28.30	.020	1361	5265	1.09	.085	1877	1643	1210	3482	.854	3396	2493
38	935	476	28.51	.025	1317	4990	1.06	.086	1871	1646	1220	3399	.827	3414	2569
39	944	475	28.13	.030	1275	4910	1.09	.091	1892	1668	1242	3499	.870	3452	2596
40	939	473	27.82	.035	1220	4785	1.11	.095	1896	1672	1251	3503	.881	3455	2621
41	945	474	28.51	.040	1197	4560	1.08	.096	1956	1739	1313	3549	.901	3578	2706
42	945	473	28.73	.044	1150	4460	1.07	.098	1979	1757	1330	3527	.896	3620	2741
43	945	473	28.73	.048	1122	4250	1.06	.100	1992	1768	1346	3516	.899	3631	2752
44	938	478	27.42	0	1725	6470	1.23	0.083	1779	1555	1104	3488	0.886	3264	2439
45	940	469	28.17	.015	1460	6020	1.19	.089	1863	1623	1196	3364	.843	3362	2518
46	940	475	28.00	.020	1390	5810	1.18	.091	1843	1615	1186	3359	.842	3348	2504
47	936	473	28.00	.024	1332	5710	1.19	.094	1882	1664	1255	3371	.852	3412	2583
48	942	473		0	1640	0	0.25	0.016	----	----	----	----	-----	----	1285
49	940	544	24.52	0	1379	4632	1.01	.068	1580	1394	1010	3651	-----	2770	1978
50	942	542	24.79	0	1368	4640	1.00	.067	1556	1357	922	3651	-----	2765	1962
51	945	543	25.88	.020	1170	4350	1.00	.079	1669	1447	1027	3491	-----	2951	2105
52	940	542	26.01	.028	1118	3909	.97	.082	1743	1549	1175	3525	-----	3080	2242
53	940	542	26.41	.036	1071	3815	.98	.087	1776	1576	1201	3446	Not computed	3138	2287
54	945	541	26.71	.043	1030	3470	.96	.090	1812	1607	1236	3397	-----	3184	2313
55	940	540	26.46	.047	1012	3365	.97	.093	1836	1634	1276	3434	-----	3209	2358
56	945	539	26.75	.048	1031	3360	.97	.093	1821	1598	1214	3401	-----	3202	2331
57	945	541	26.71	.051	1012	3200	.96	.095	1835	1617	1244	3387	-----	3211	2340
58	942	542	26.63	.054	1015	3030	.96	.096	1791	1555	1171	3267	-----	3116	2253
59	943	542	24.83	0	1390	5260	1.10	0.074	1575	1375	946	3576	-----	2798	1992
60	954	539	25.45	0	1425	5220	1.08	.073	1612	1398	946	3602	-----	2870	2022
61	946	540	25.73	.020	1175	4940	1.10	.086	1712	1489	1074	3553	-----	3035	2194
62	946	540	25.95	.028	1138	4560	1.07	.089	1750	1528	1123	3555	-----	3103	2255
63	942	539	26.10	.036	1051	4220	1.05	.092	1785	1564	1154	3613	-----	3195	2352
64	956	539	26.72	.043	1003	3960	1.03	.095	1822	1605	1215	3453	-----	3221	2324
65	939	537	25.91	.044	1026	3950	1.06	.097	1717	1489	1077	3369	-----	3038	2210
66	934	541	25.14	0	1361	0	0.23	0.015	1554	1467	1363	----	-----	1810	1005
67	938	544	24.71	0	1368	0	.23	.015	1551	1458	1353	----	-----	1767	984
68	953	612	22.22	0	778	4220	1.00	0.067	1391	1201	806	3606	-----	2316	1533
69	942	616	22.22	0	1138	4280	1.01	.068	1388	1215	794	3753	-----	2392	1625
70	942	616	22.63	.016	1012	3975	1.01	.077	1468	1241	890	3777	-----	2556	1774
71	947	616	23.32	.024	958	3710	.97	.079	1529	1355	984	3643	-----	2647	1832
72	946	616	23.51	.033	900	3525	.98	.085	1568	1390	1021	3635	-----	2724	1904
73	938	611	24.08	.041	900	3230	.96	.089	1641	1453	1100	3542	-----	2832	2012
74	938	614	24.02	.046	875	2965	.94	.091	1643	1456	1108	3512	-----	2825	2005
75	946	614	24.13	.050	840	2910	.95	.094	1647	1460	1117	3428	-----	2813	1973
76	956	613	24.09	.050	841	2910	.95	.094	1654	1449	1122	3308	-----	2738	1880
77	952	613	23.85	.055	824	2840	.97	.098	1646	1437	1106	3324	-----	2728	1887
78	942	620	21.49	0	1150	4735	1.13	0.076	1647	1198	793	3721	-----	2331	1586
79	944	613	22.08	0	1152	4740	1.10	.074	1405	1216	817	3618	-----	2366	1603
80	946	613	22.75	.014	1028	4415	1.08	.081	1462	1264	862	3558	-----	2485	1694
81	942	614	22.89	.024	967	4120	1.07	.086	1489	1288	905	3479	-----	2524	1735
82	941	612	22.91	.020	1015	4290	1.08	.084	1477	1279	879	3525	-----	2528	1741
83	941	612	23.32	.029	956	3949	1.04	.087	1529	1325	944	3483	-----	2606	1805
84	942	615	21.81	0	1138	0	0.22	0.014	1378	1298	1197	----	-----	1501	748
85	935	615	21.93	0	1128	0	.22	.014	1357	1275	1182	----	-----	980	751



Station	1	5	7	8	0
Total-pressure probes	22	20	8	16	-
Static-pressure probes	9	4	10	8	5
Thermocouples	21	48	24	-	-

CD-2765

Figure 1. - Sectional view of engine showing instrumentation stations.

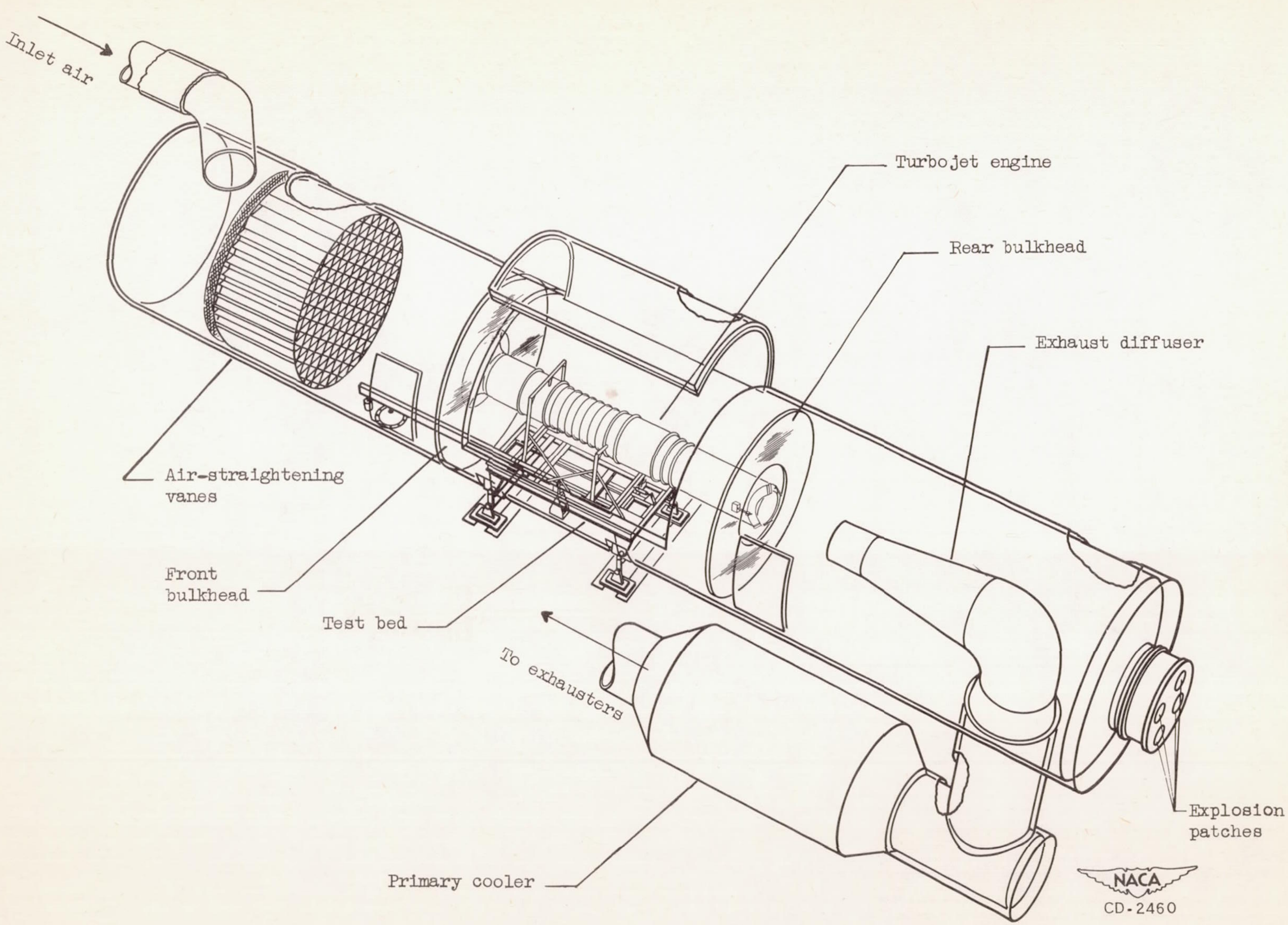


Figure 2.-Sketch of altitude test chamber showing turbojet engine installed on test bed.

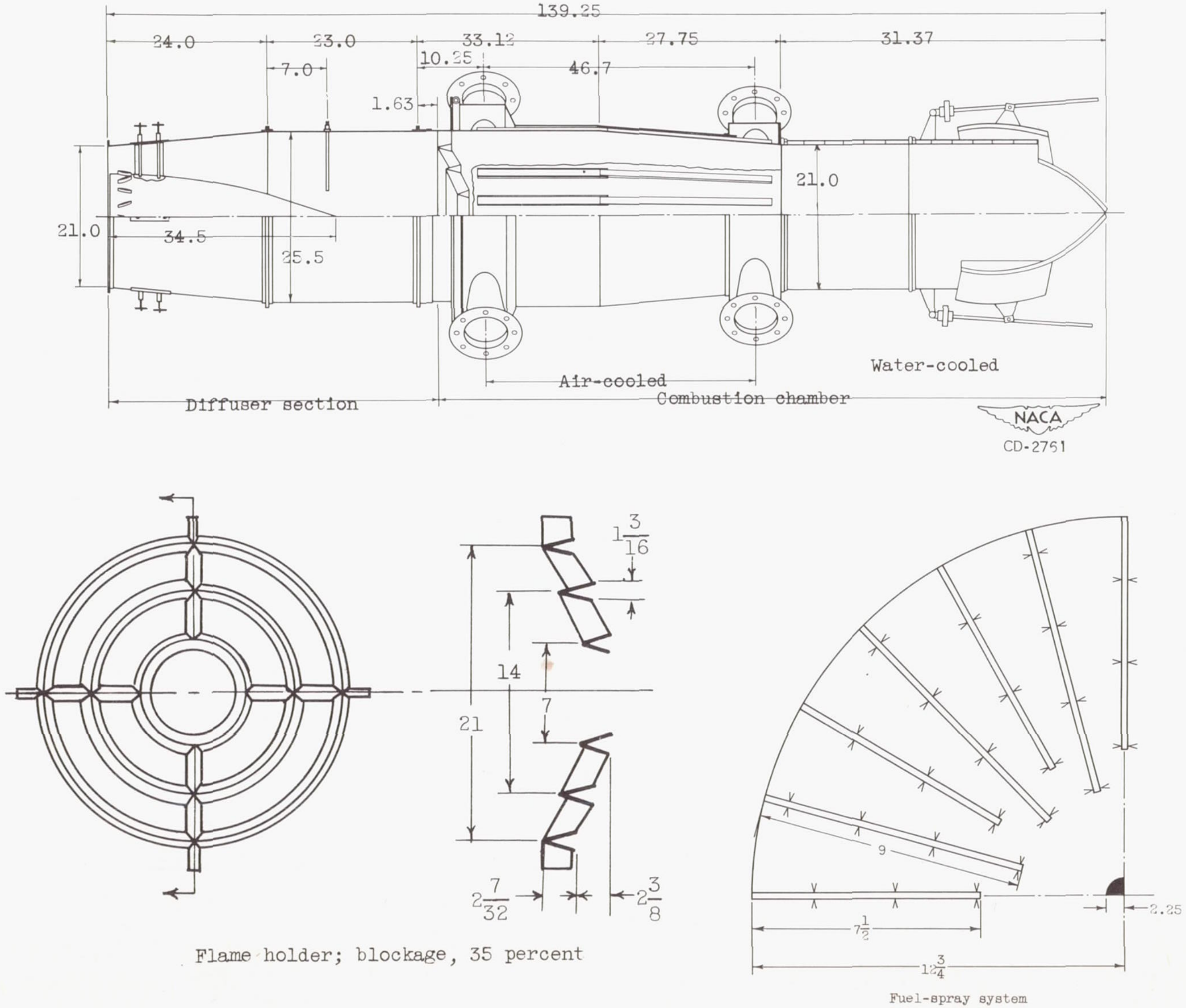
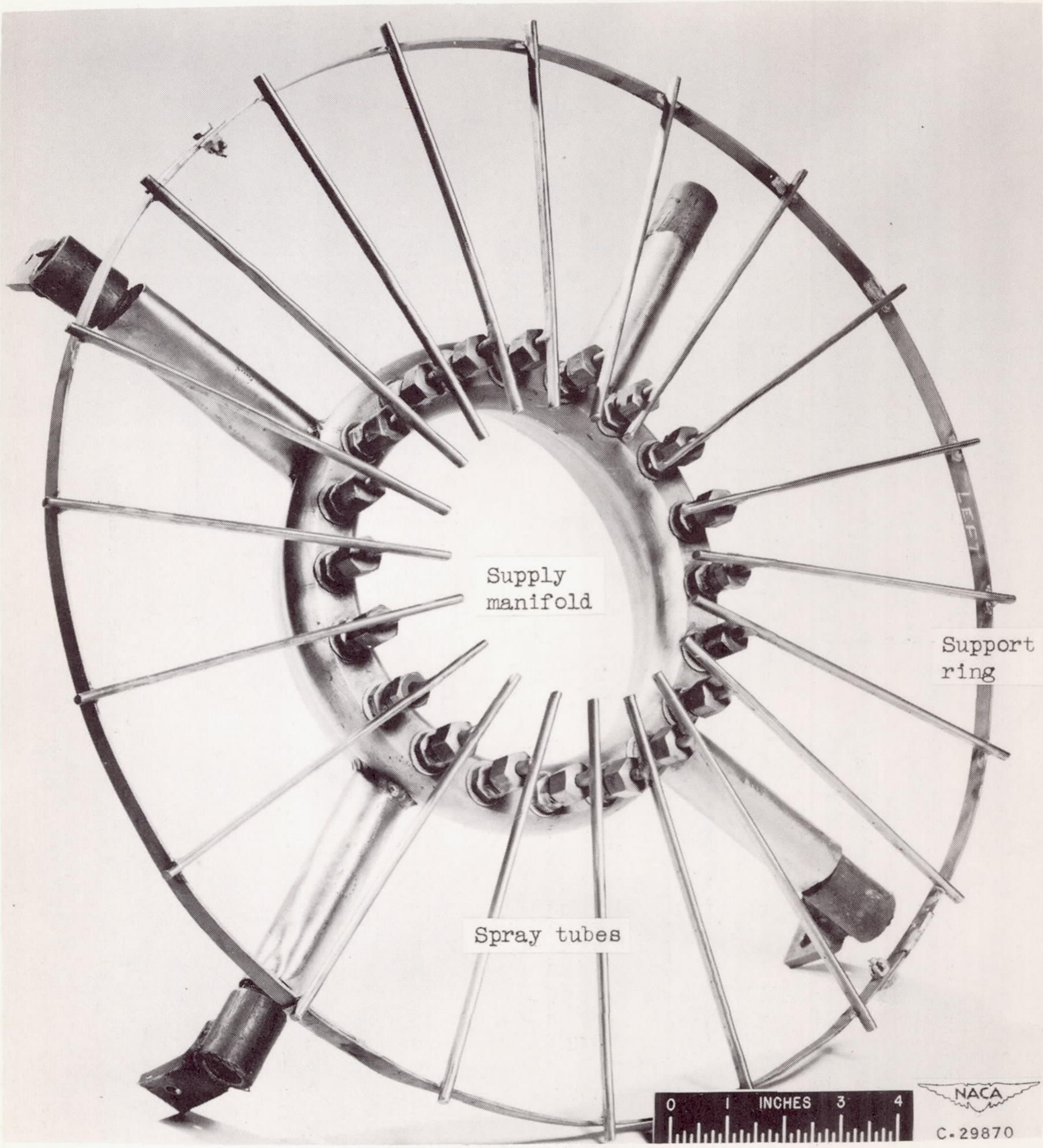
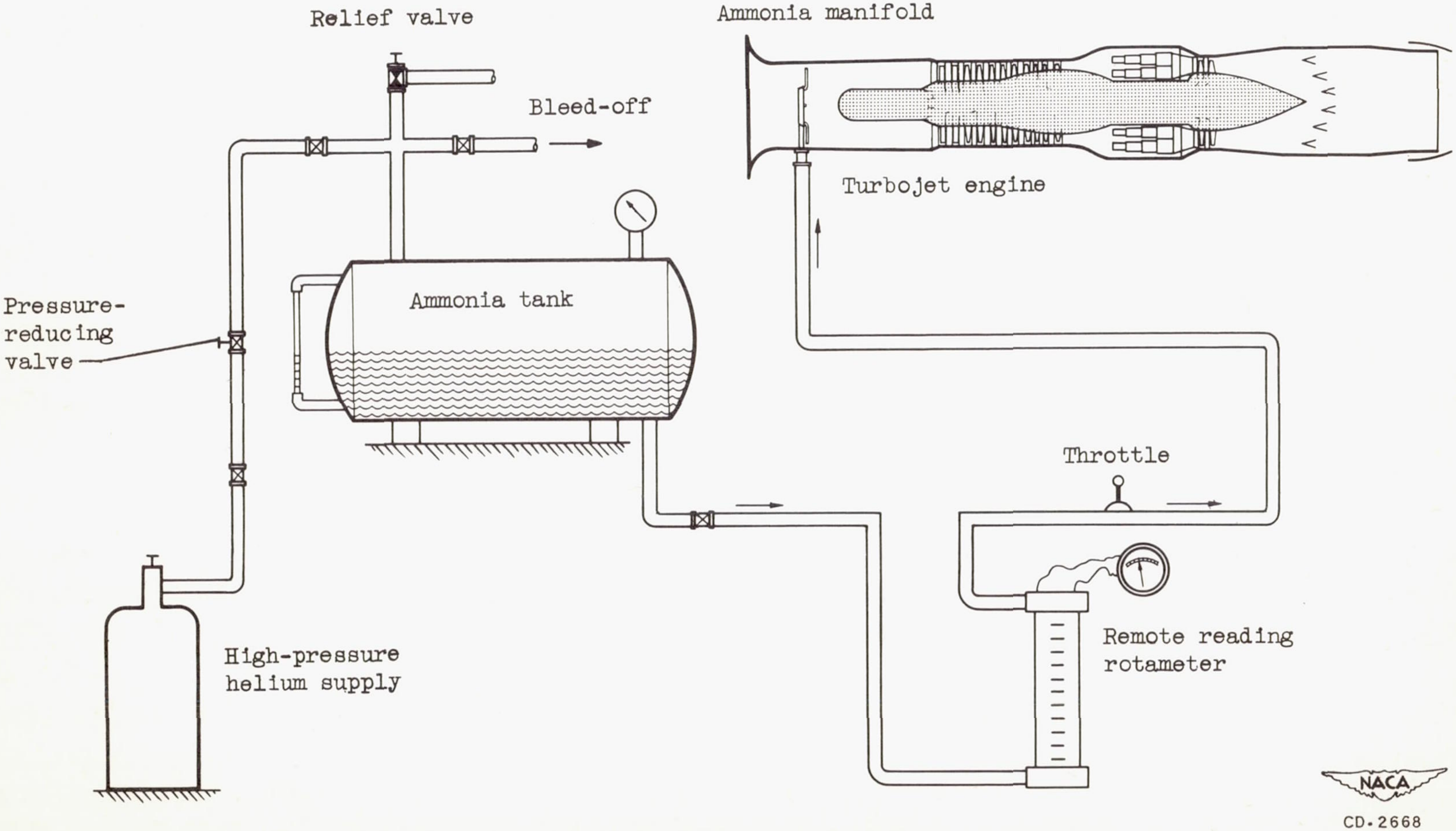


Figure 3. - Sectional view of afterburner. (All dimensions in inches.)



(a) Spray manifold and tubes.

Figure 4. - Ammonia-injection system.



(b) Schematic diagram.

Figure 4. - Concluded. Ammonia-injection system.

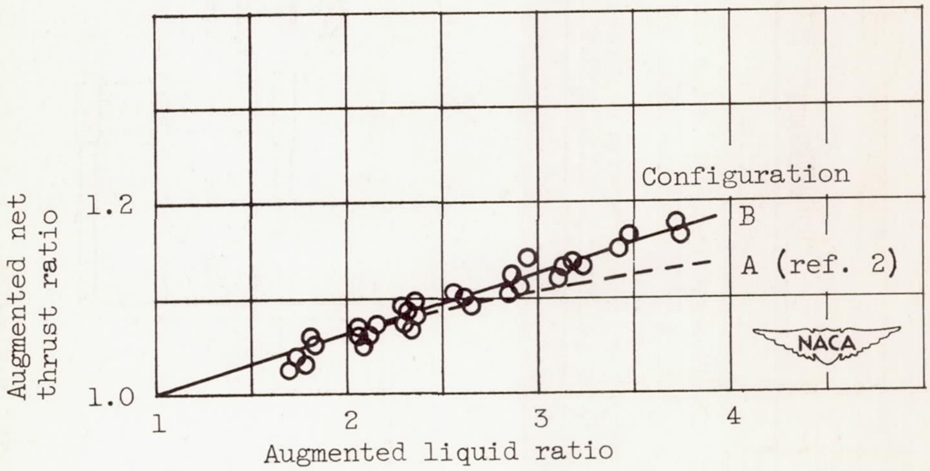
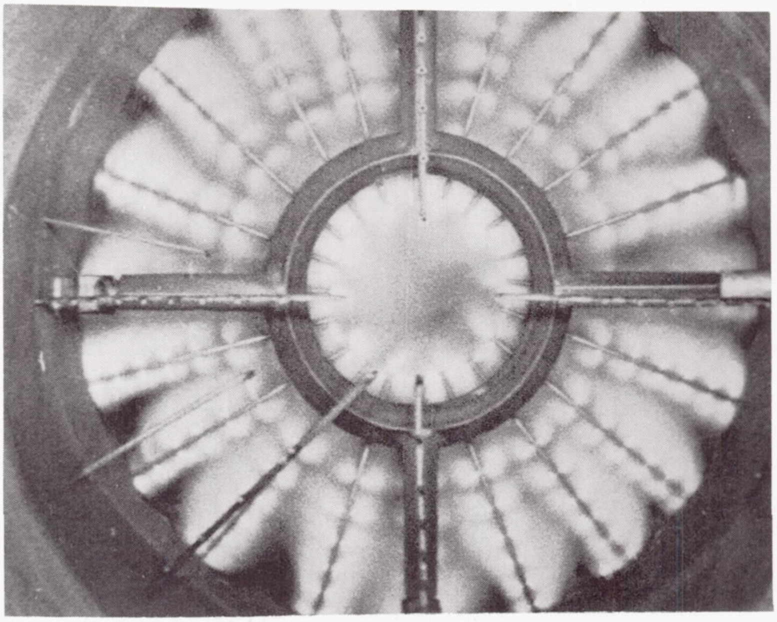
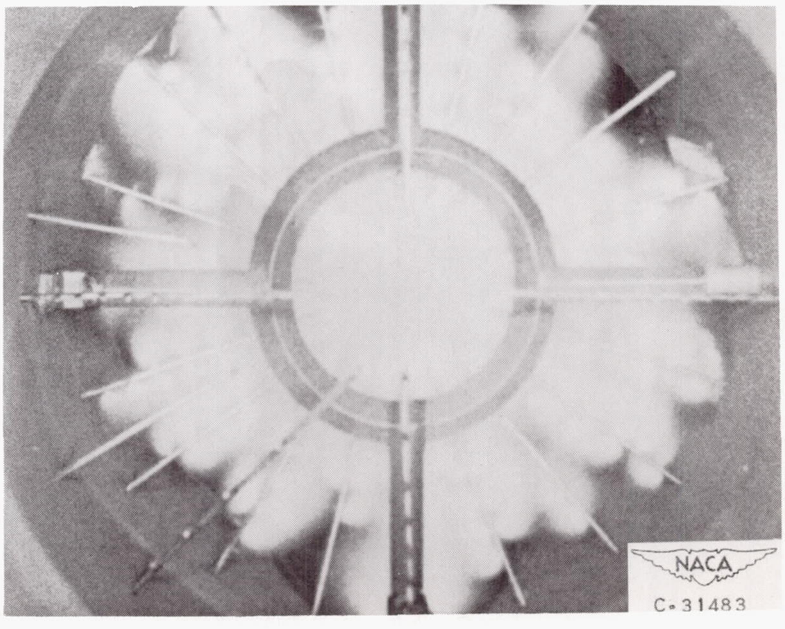


Figure 5. - Additional thrust augmentation achieved with improved distribution of ammonia and air at engine inlet. Engine operation with ammonia injection alone. Altitude, 35,000 feet; flight Mach number, 1.0; engine inlet-air temperature, 13° F.



(a) Optimum ammonia distribution. Flow, 1.12 pounds per second; manifold pressure, 70 pounds per square inch gage.



(b) Poor ammonia distribution due to plugging of orifices by foreign matter. Flow, 1.0 pounds per second; manifold pressure, 75 pounds per square inch gage.

Figure 6. - Ammonia-spray distribution achieved with configuration B.

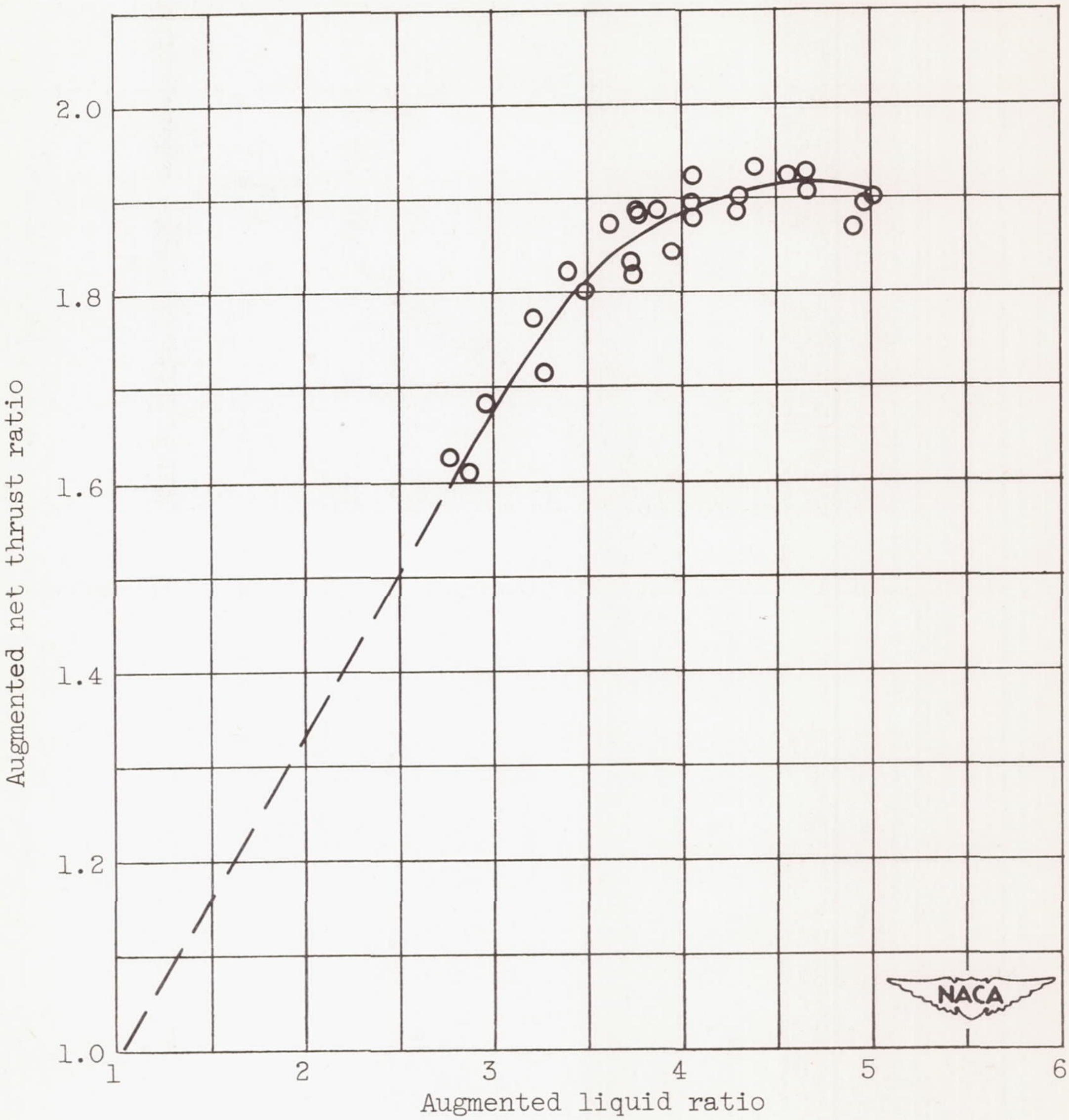


Figure 7. - Thrust augmentation achieved with high-performance afterburner. Altitude, 35,000 feet; flight Mach number, 1.0; engine inlet-air temperature, 13° F.

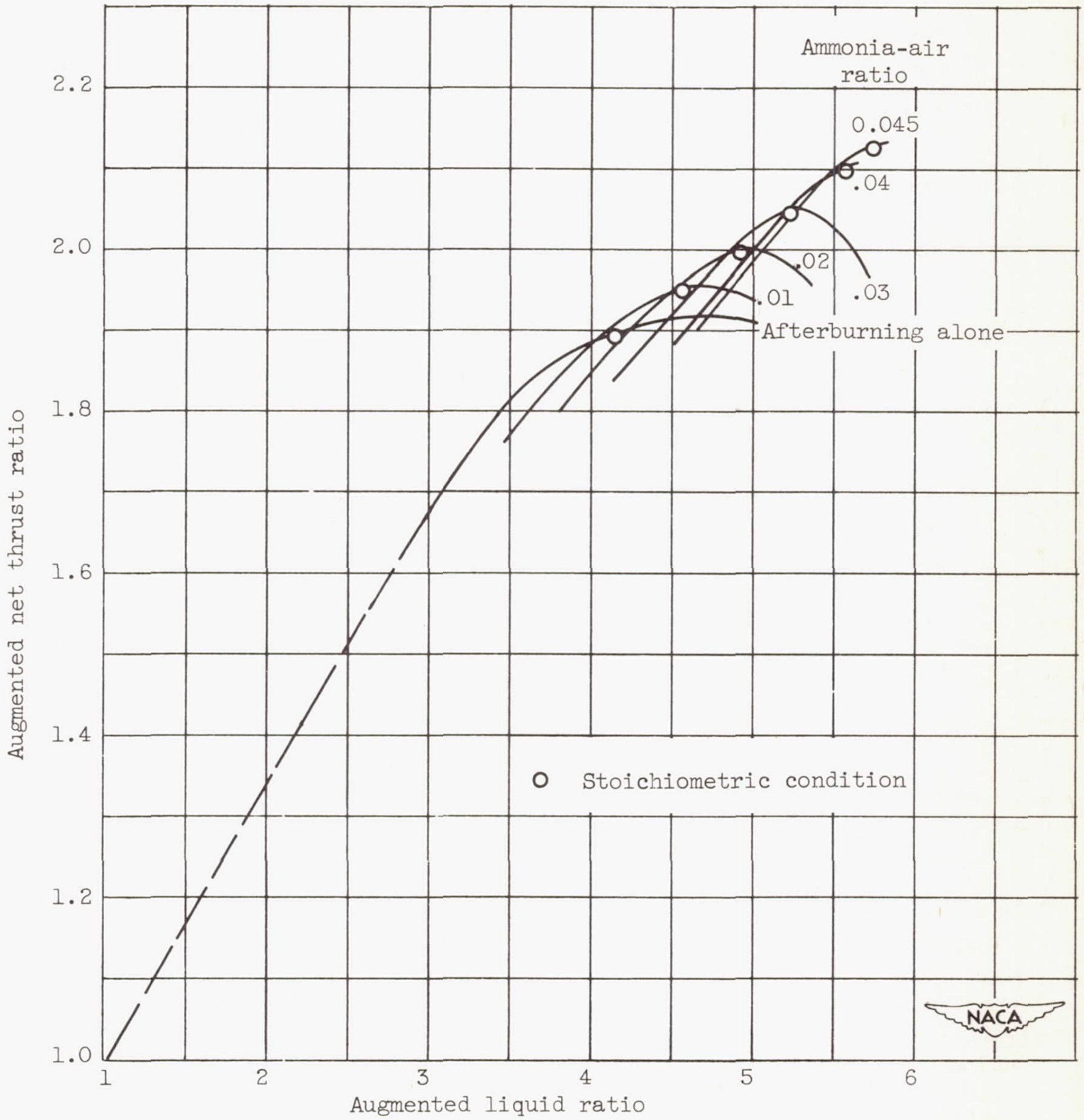


Figure 8. - Variation of augmented net thrust ratio with ammonia-air ratio for turbojet engine operating with combined compressor-inlet ammonia injection and afterburning. Altitude, 35,000 feet; flight Mach number, 1.0; engine inlet-air temperature, 13° F.

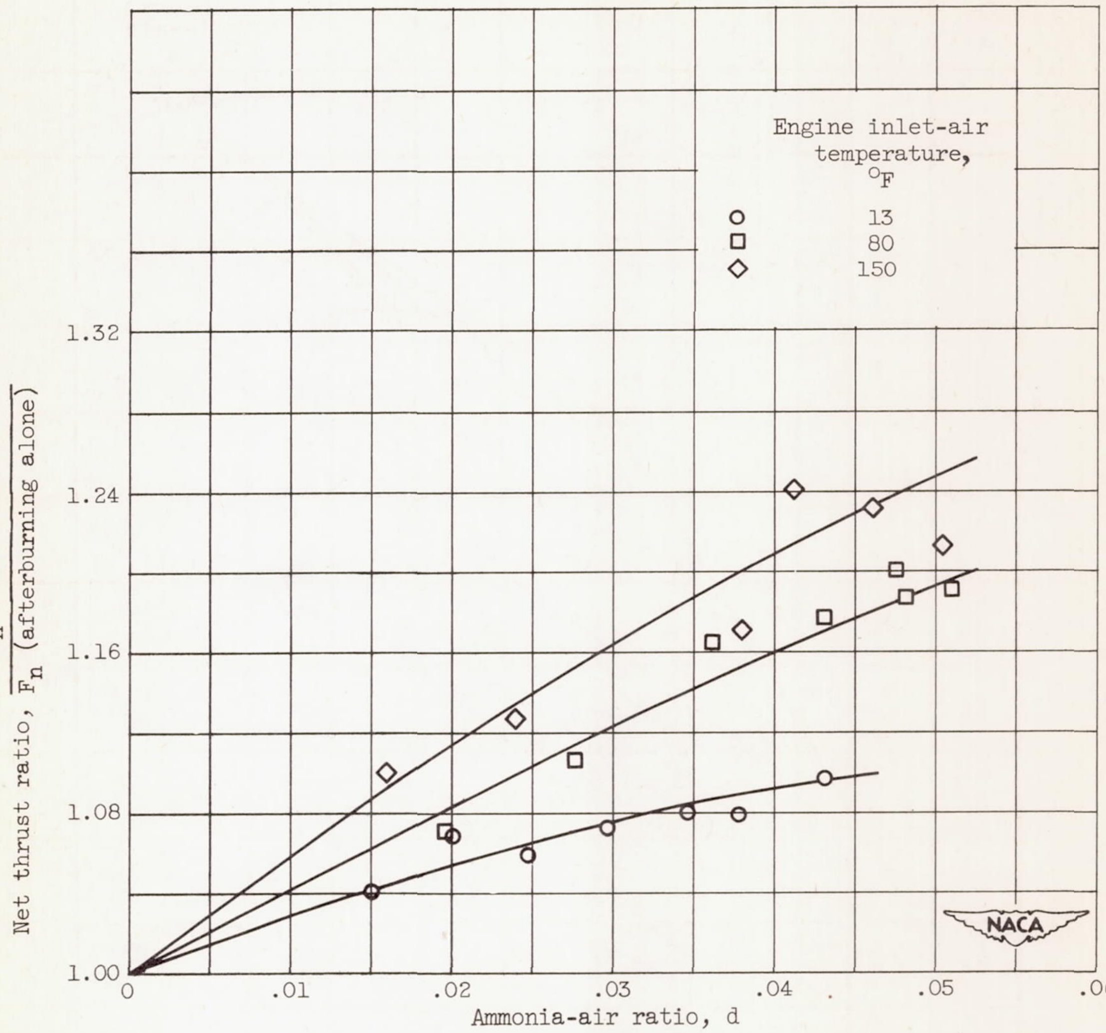
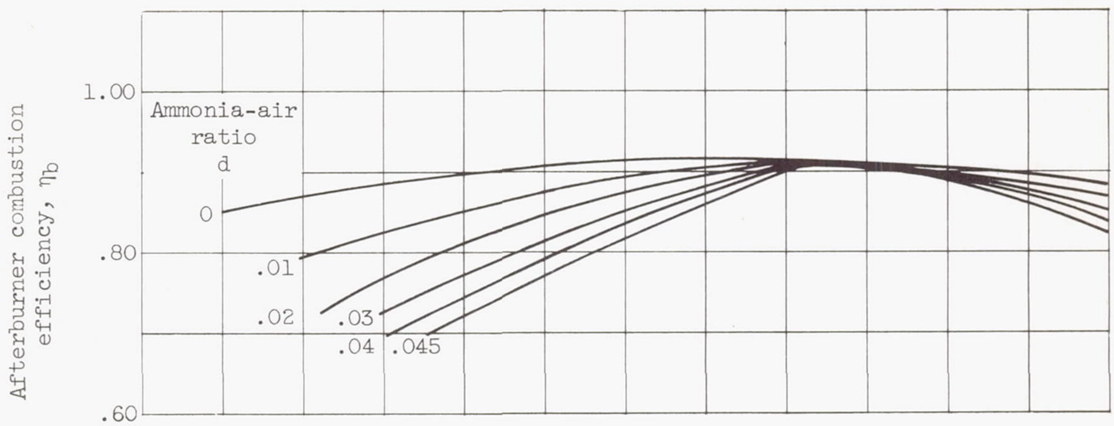
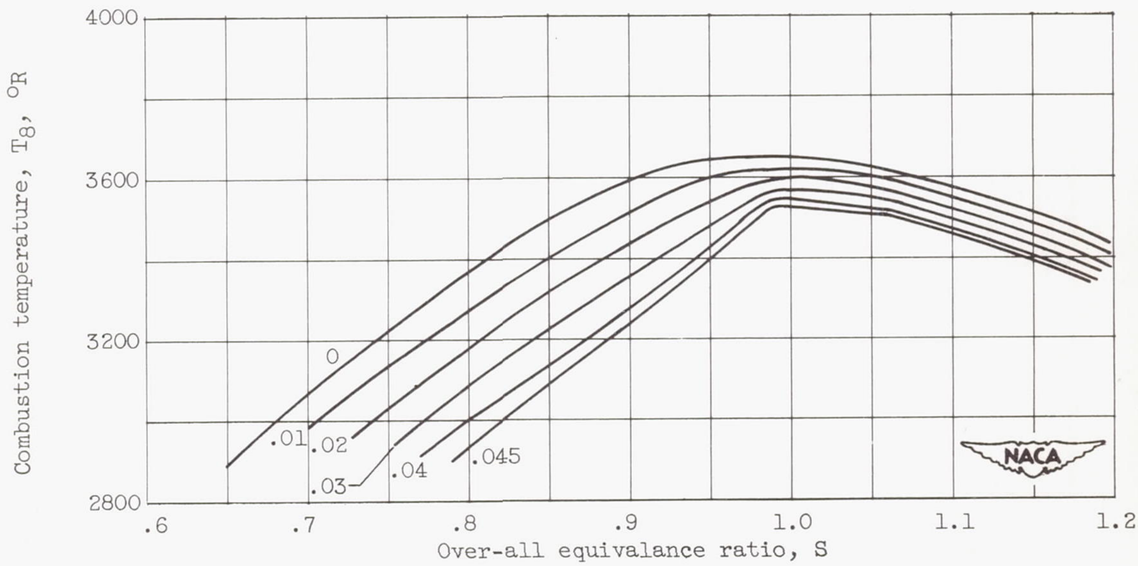


Figure 9. - Variation of net thrust ratio (ratio of combined ammonia injection and afterburning to afterburning alone) for range of ammonia-air ratios and engine inlet-air temperatures. Altitude, 35,000 feet; flight Mach number, 1.0; over-all equivalance ratio, 1.0.



(a) Afterburner combustion efficiency.



(b) Afterburner combustion temperature.

Figure 10. - Effect of ammonia-air ratio on afterburner combustion efficiency and temperature for range of over-all equivalance ratios. Altitude, 35,000 feet; flight Mach number, 1.0; engine inlet-air temperature, 13° F; afterburner-inlet gas temperature, 1165° F; velocity, 391 feet per second; total pressure, 1780 pounds per square foot.

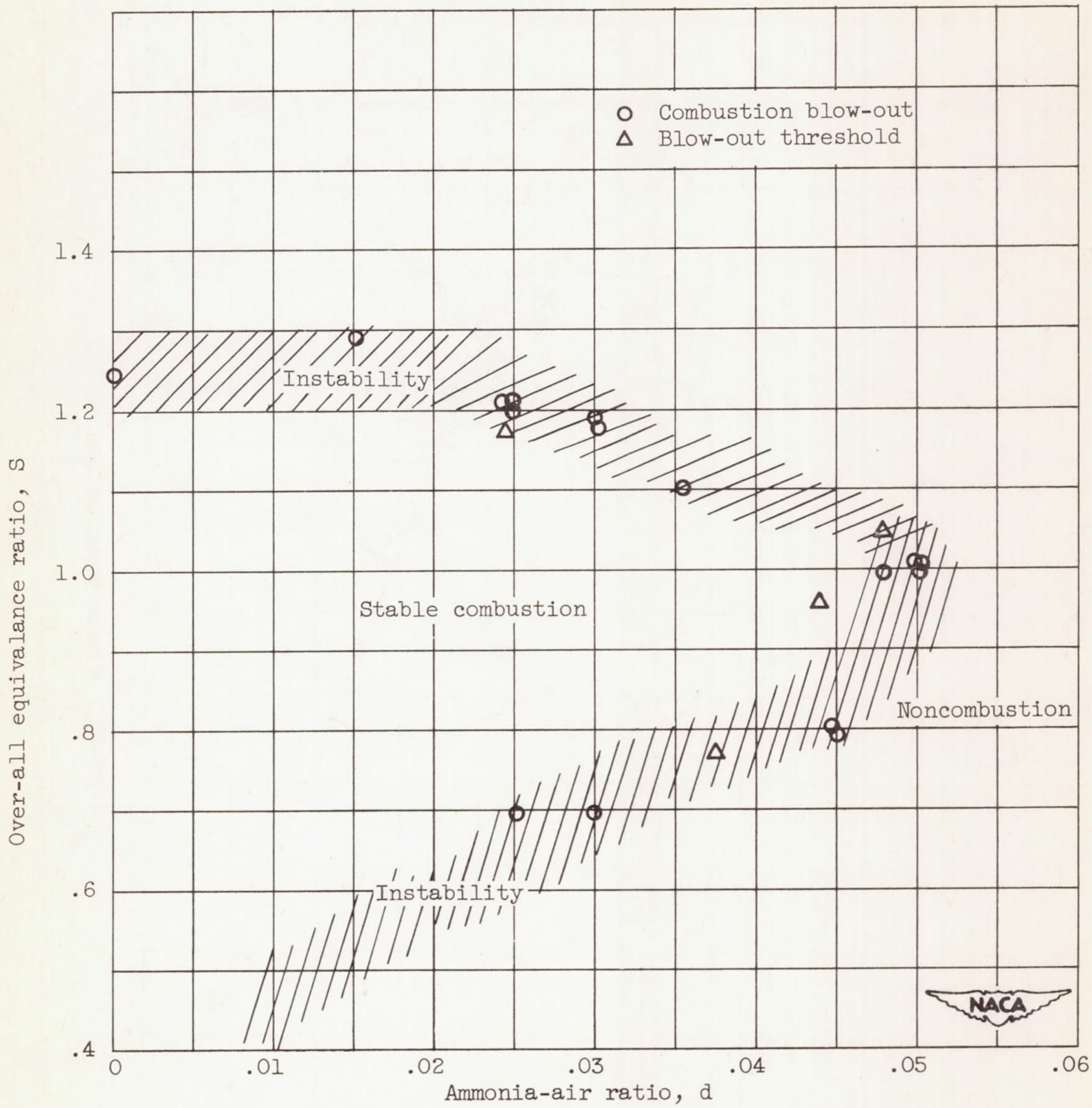


Figure 11. - Region of stable combustion in afterburner with varying ammonia-air ratio. Altitude, 35,000 feet; flight Mach number, 1.0; engine inlet-air temperature, 13° F; afterburner-inlet gas temperature, 1165° F; velocity, 391 feet per second; total pressure, 1780 pounds per square foot.

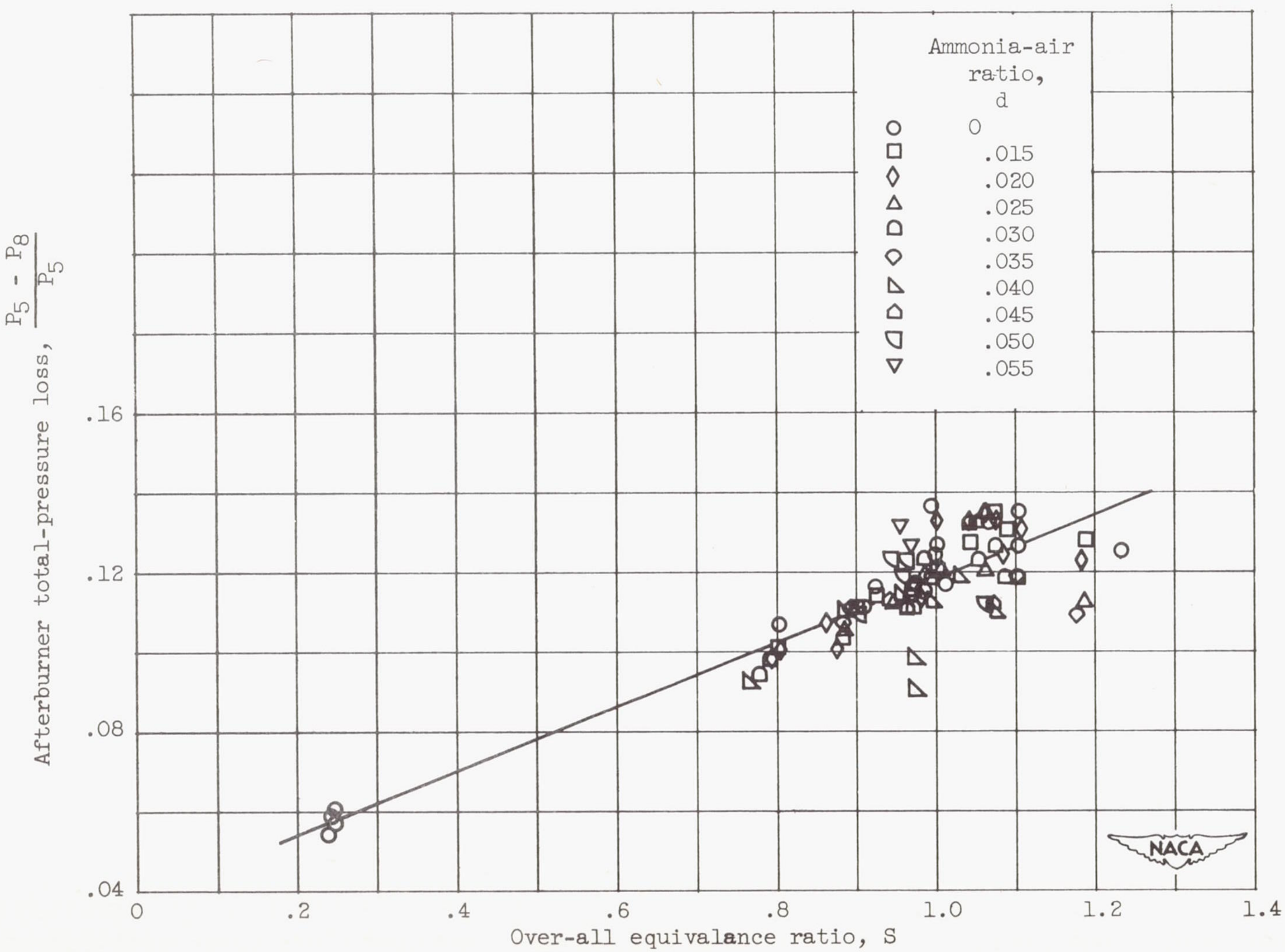


Figure 12. - Afterburner total-pressure loss due to drag and momentum change during combustion of ammonia and hydrocarbon fuel. Altitude, 35,000 feet; flight Mach number, 1.0.

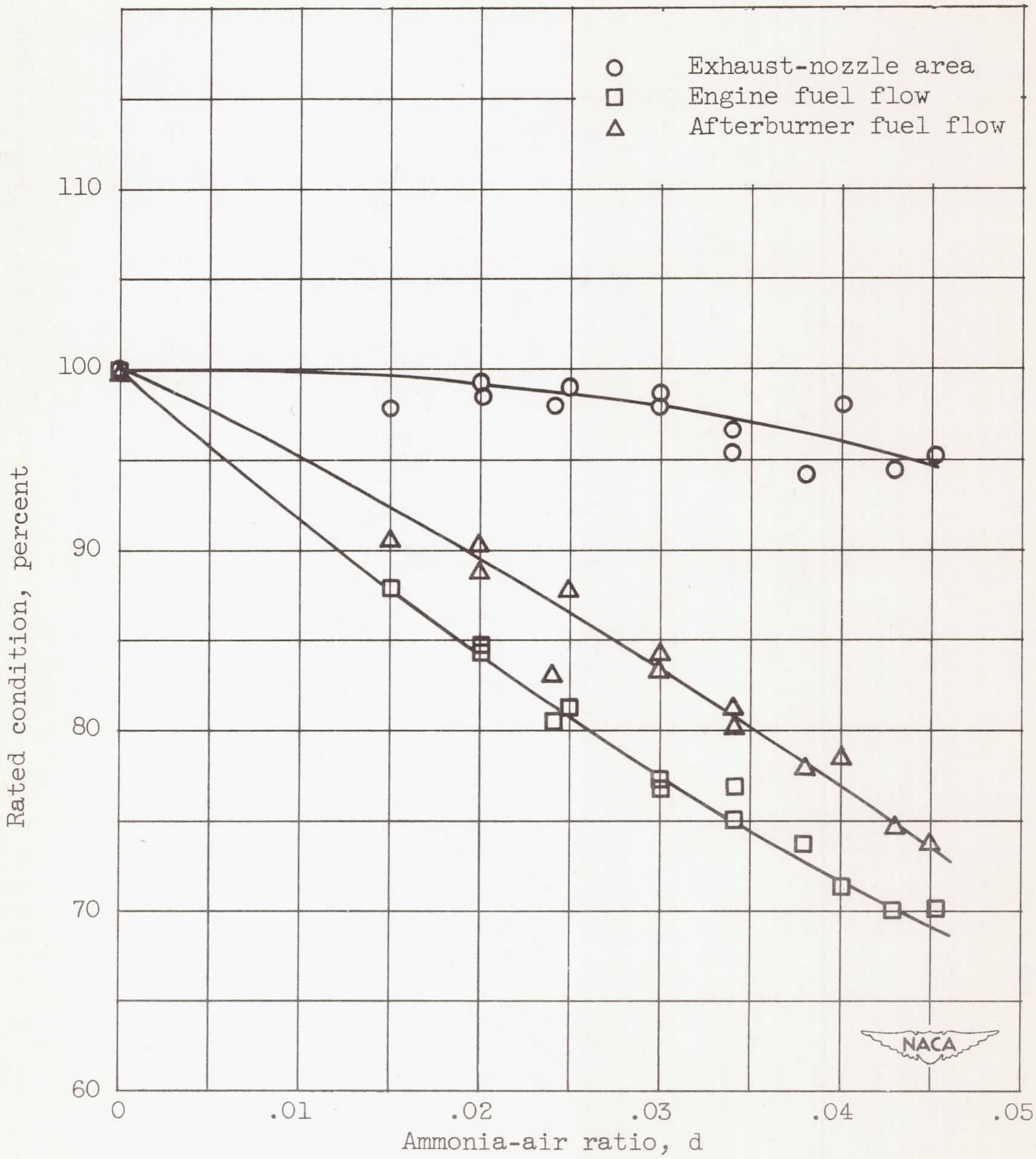


Figure 13. - Variation of major variables of combined augmentation system with increase of ammonia-air ratio in compressor. Stoichiometric operation of afterburner; altitude, 35,000 feet; flight Mach number, 1.0.