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RESEARCH MEMORANDUM

INVESTIGATION OF INTERNAL FILM COOLING OF EXHAUST NOZZLE

OF A 1000-POUND-THRUST LIQUID-AMMONIA LIQUID-OXYGEN

ROCKET

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NATIONAL ADVISORY COMMITTEE
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INVESTIGATION OF INTERNAL FILM COOLING OF EXHAUST NOZZLE OF A 1000-POUND-THRUST LIQUID-AMMONIA LIQUID-OXYGEN ROCKET

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SUMMARY

An investigation of internal film cooling of the exhaust nozzle of a 1000-pound-thrust liquid-ammonia liquid-oxygen rocket engine was undertaken. The experiments covered a range of oxidant-fuel ratios with the combustion-chamber pressure varying from 220 to 275 pounds per square inch absolute. Water and anhydrous liquid ammonia were used as coolants with the coolant flow constituting from about 12 to 25 percent of the total flow. The coolant was injected uniformly about the circumference of the nozzle entrance.

With water as a coolant, approximately 16 percent of the total propellant and coolant flow was required to film-cool the entire nozzle and with anhydrous liquid ammonia, approximately 19 percent of the total flow was required.

The maximum specific impulse obtained from an uncooled rocket engine was 207 pound-seconds per pound or 83 percent of the theoretical equilibrium specific impulse. When the entire nozzle was film-cooled with ammonia, the maximum specific impulse obtained was 162 pound-seconds per pound or only 78 percent of the specific impulse obtained with a corresponding uncooled engine. Although the performance of the uncooled rocket engine was low, this reduction in performance was of sufficient magnitude to make film cooling of the entire nozzle by annular injection of ammonia at the nozzle entrance appear undesirable.

INTRODUCTION

In rocket engines, in which high temperatures and accompanying high heat-flux densities are encountered, one of the major problems is that of providing a suitable combustion chamber and nozzle capable of withstanding the adverse temperature effects. The methods of cooling these components of the rocket engine which have received the most emphasis are transpiration cooling (references 1 to 6), regenerative cooling, and film cooling.

Internal film cooling is a promising method of cooling and appears to offer some advantages over regenerative cooling, such as simplicity of design and low weight of thrust chamber. In general, if a rocket propellant is to be a satisfactory coolant, it must have the required heat-absorbing capacity. For regenerative cooling, the heat-absorbing capacity of a propellant is determined by the amount of heat it can absorb in the liquid state; for film cooling, however, the heat-absorbing capacity includes not only the heat that can be absorbed in the liquid state but also the latent heat of vaporization. Whereas the heat-transfer rate to a film coolant is higher than that to a regenerative coolant, the larger heat-absorbing capacity of a film coolant, in general, more than compensates for the increased heat-transfer rate. Hence some propellants not suitable as regenerative coolants may be used as film coolants. For a given application, however, the merits of each method of cooling are to be considered and in some rocket engines, a combination of both regenerative and film cooling is used.

Investigations relating to film cooling under idealized conditions are given in references 7 to 10. A visual study of the flow characteristics of liquid films over a range of gas-stream conditions is given in reference 7. The stability of liquid films established by a slot-type liquid injector oriented at different angles to the wall is reported in reference 8 for various gas-stream conditions. Film cooling of a hydrogen-oxygen flame tube with water is reported in reference 9 and a preliminary correlation of film-cooling heat-transfer results obtained at air temperatures to 2000° F in a 4-inch-diameter duct with water as a coolant is given in reference 10. Application of film cooling to rocket engines is reported in references 11 to 13. A considerable reduction of the heat load to the regenerative coolant by use of film cooling with water in an acid-aniline rocket engine is reported in reference 11. Film cooling the combustion chamber of an ammonia-oxygen rocket with water, ammonia, and ethyl alcohol was reported in reference 12. The performance of the rocket with the combustion chamber film-cooled with ammonia was as good as the performance of a similar uncooled rocket engine. Film cooling the nozzle of an acid-aniline rocket by injecting water through several individual injectors located at various stations along the rocket nozzle resulted in a considerable reduction in the nozzle wall temperature (reference 13).

Previous investigations have not determined the feasibility of completely film cooling the nozzle. The investigation reported herein was conducted at the NACA Lewis laboratory in order to determine this possibility. A 1000-pound-thrust liquid-ammonia liquid-oxygen rocket employing annular injection of the coolant at the nozzle entrance was used. Water and liquid ammonia were used as coolants. The nozzle wall temperature was determined by skin thermocouples spotted on the outside surface of the nozzle at the desired positions.

SYMBOLS

The following symbols are used in this report:

F engine thrust, lb force

I specific impulse of rocket engine, $\dfrac{\text{thrust (lb force)}}{\text{total liquid flow rate (lb/sec)}}$

L^* ratio of combustion-chamber volume to nozzle throat area, in.

P_c combustion-chamber pressure, lb/sq in. abs

W_f fuel flow, lb/sec

W_l liquid-coolant flow, lb/sec

W_o oxidant flow, lb/sec

APPARATUS

The rocket engine used for this investigation (fig. 1) consisted of a propellant injector incorporating a gunpowder squib ignitor, an uncooled combustion chamber, a coolant injector, and a thin-walled nozzle. A gas-pressurized propellant and coolant feed system was used. The characteristic length L^* of the engine was 42 inches.

Propellant injector. - An impinging-jet propellant injector as shown on figure 2 was used. It consisted of 24 pairs of jets of one-on-one impingement approximately 0.375 inch from the injector face.

Ignition system. - The propellant injector was designed to incorporate a gunpowder squib ignitor at the center of the injector. The squib was ignited by electric power.

Combustion chamber. - The combustion chamber was made of mild steel having a 4-inch inside diameter with walls 1/2 inch thick and a length of 6.70 inches. A flange from the combustion chamber mounted the rocket motor to the thrust stand.

Coolant injector. - The coolant injector was mounted between the combustion chamber and the exhaust nozzle. The injector provided a supply annulus for the coolant from which the coolant flowed through an annular slot to the inside surface of the nozzle, as shown on figure 3. The slot was 0.010 inch wide and inclined 30° to the center line of the rocket and thus directed the coolant along the nozzle surface. It was necessary to provide a lip on the coolant injector to cover the joint of coolant injector and nozzle, as slight misalinement of the coolant injector and nozzle encountered during assembly would otherwise result in nonuniform coolant distribution.

Exhaust nozzle. - The convergent-divergent exhaust nozzle, made of stainless steel, is shown on figure 4. The nozzle had the following dimensions: wall thickness, 0.095 inch; throat diameter, 1.85 inches; exit diameter, 3.69 inches; exit-throat area ratio, 3.96; convergent half-angle, 30°; and divergent half-angle, 15°. The design expansion ratio was 20.4. On the nozzle were spotted 34 chromel-alumel skin thermocouples arranged in four longitudinal rows at the positions given in the following table:

Distance downstream of nozzle entrance (in.)	Number of thermocouples	Circumferential spacing of thermocouples (deg)
0.5	1	360
.9	4	90
1.4	4	90
1.9	4	90
[a]2.35	4	90
2.9	4	90
3.3	1	360
4.3	4	90
5.3	4	90
6.1	4	90

[a]Location of nozzle throat

Propellants and coolants. - Liquid oxygen and anhydrous liquid ammonia were used as propellants. Anhydrous liquid ammonia and filtered water from the city mains were used as coolants.

Instrumentation. - The propellant flow was measured with an accuracy of 1 percent by the use of a strain gage attached to a counter-balanced weighing beam which supported the propellant tanks. The output of the strain gage was recorded continuously as a function of time on a self-balancing potentiometer. Coolant flow was measured with an accuracy of 0.02 pound per second by an area-type flowmeter.

The thrust of the rocket engine was measured with an accuracy of 10 pounds by means of a strain gage attached to a parallelogram thrust stand; the output of the strain gage was recorded by a self-balancing potentiometer.

The outputs of 12 chromel-alumel thermocouples were measured with an accuracy of 0.125 millivolt in the 10-millivolt range by a recording self-balancing potentiometer. The outputs of 16 of the remaining thermocouples were recorded by means of a single-channel oscillograph in conjunction with a selector switch making 10 contacts a second. Accuracy of 0.2 millivolt in a 22-millivolt range was obtained.

The combustion-chamber pressure was measured with an accuracy of 5 pounds per square inch by means of a Bourdon-tube type strip-chart recorder.

PROCEDURE

In operating the rocket, the squib was first ignited and then propellant and coolant flows were started simultaneously. If the nozzle remained completely cooled, the rocket was operated for approximately 15 seconds so as to obtain reliable data. If the nozzle became overheated because of insufficient coolant flow or improper coolant distribution, the run was terminated within 5 seconds to prevent burnout of the nozzle.

Reduction of performance data. - All performance data were corrected to a common combustion-chamber pressure of 250 pounds per square inch absolute for purposes of comparison. A plot of specific impulse against combustion-chamber pressure obtained from reference 14 showed that the change in theoretical specific impulse for a change in combustion-chamber pressure from $P_{c,1}$ to $P_{c,2}$ could be approximated by the equation

$$\Delta I = 79.5 \log \frac{P_{c,1}}{P_{c,2}}$$

for the range of combustion-chamber pressures and oxidant-fuel ratios encountered in this investigation. The oxidant-fuel ratio was calculated by neglecting the coolant flow. The coolant flow was included in the determination of the specific impulse and thus gave the thrust per pound of combined propellant and coolant flow.

Uncooled rocket performance. - The performance of an uncooled rocket engine similar to the engine used for the film-cooling runs was determined for purposes of comparison. This uncooled rocket engine consisted of the same combustion chamber and propellant injector that were used for the film-cooling runs. The coolant injector and the thin-walled nozzle were replaced with a solid-copper nozzle. The characteristic length of the uncooled engine was 35 inches as compared with 42 inches for the film-cooled engine.

RESULTS AND DISCUSSION

The pertinent data obtained from the runs are tabulated in table I. The oxidant-fuel ratio by weight varied between 1.35 and 3.98. The combustion-chamber pressure varied between 220 and 275 pounds per square inch absolute and the coolant flow constituted from about 12 to 25 percent of the total flow.

Coolant Results

Nozzle wall temperature. - The wall temperature profile of the nozzle when completely film-cooled with water is shown on figure 5(a). Also shown is the boiling point of water corresponding to the pressure variation along the nozzle. The wall temperature rises rapidly downstream of the coolant injector to a value slightly below the boiling point of water and remains slightly below the boiling point for the remaining length of the nozzle. The nozzle wall temperature profile obtained with ammonia as a coolant is given on figure 5(b). The wall temperature is approximately 100° F along the entire nozzle, whereas the boiling point of ammonia varies from 115° F at the nozzle entrance to -35° F at the exit. Wall temperatures exceeding the boiling point of ammonia were also encountered when ammonia was used to film-cool the combustion chamber of a rocket engine (reference 12). A possible explanation for this phenomenon was that a fuel-rich low-temperature region exists near the wall and thus gives a region having a low diffusion rate which would tend to allow superheating of the liquid. Another possible explanation is that, as the exhaust gases contain a high percentage of water vapor, the liquid ammonia along the wall could absorb some water vapor and thus change to ammonium hydroxide, which has a higher boiling point. Still another possibility is that the liquid ammonia does not cover the nozzle surface in a continuous liquid sheet; that is, the film consists of a mixture of vapor and droplets.

Coolant flow required to film-cool the entire nozzle. - For the rocket configuration used in this investigation, the minimum coolant flow required for film cooling the entire nozzle with water was approximately 16 percent of the total flow and with ammonia, approximately 19 percent of the total flow. For lower coolant flows, the liquid film terminates upstream of the nozzle exit and the portion of the nozzle which is not protected by a liquid film becomes overheated.

Performance Results

The performance obtained for the various experimental conditions is shown on figure 6 in which specific impulse is plotted against oxidant-fuel ratio. The experimental specific impulse obtained for the uncooled rocket engine is low, approximately 83 percent of the theoretical performance. The theoretical performance is for complete combustion at 250 pounds per square inch absolute followed by shifting equilibrium isentropic expansion to 1 atmosphere absolute pressure. The experimental specific impulse obtained when the entire exhaust nozzle was film-cooled was considerably lower than that obtained for the uncooled engine. The use of ammonia as a coolant resulted in a slightly higher specific impulse than was obtained with water as a coolant.

A maximum specific impulse of 207 pound-seconds per pound was obtained for the uncooled engine, whereas only 78 percent of this value or 162 pound-seconds per pound was obtained when the entire exhaust nozzle was film-cooled with ammonia. Although the performance of the uncooled rocket engine was low, this reduction in performance was of sufficient magnitude to make film cooling of the entire nozzle by annular injection of ammonia at the nozzle entrance appear undesirable.

CONCLUDING REMARKS

Inasmuch as film cooling of the entire exhaust nozzle by injection of the coolant at the nozzle entrance results in an intolerable loss in performance, other cooling aids or methods for the nozzle, such as regenerative cooling or ceramic linings or coatings, may be sought. Also, there is a possibility of film cooling a portion of the convergent section of the nozzle with a reactive coolant without adversely affecting the performance of the rocket. From consideration of the residence time of a particle in the exhaust nozzle (fig. 7), it is evident that if a given time is required for the vaporized coolant to diffuse and burn, then only that which vaporizes upstream of a given station along the nozzle will effectively burn in the nozzle and enhance the performance of the rocket. Further, heat that is released at low pressure in the nozzle does not have as great an expansion ratio and cannot contribute as much to performance as would be possible if it were released in the combustion chamber. The fact that the performance obtained when a reactive coolant was used, however, exceeded that obtained from an inert coolant (fig. 6) indicates that performance gain was obtained from some of the reactive coolant.

SUMMARY OF RESULTS

Internal film cooling of the nozzle of a 1000-pound-thrust liquid-ammonia liquid-oxygen rocket engine with a 42-inch characteristic length was investigated with combustion-chamber pressures from 220 to 275 pounds per square inch absolute over a range of oxidant-fuel ratios from 1.35 to 3.98. Water and liquid ammonia were used as coolants with the coolant flow constituting from about 12 to 25 percent of the total flow.

The results of the investigation can be summarized as follows:

1. The entire nozzle of the rocket engine was film-cooled with water and with liquid ammonia by uniform annular injection of the coolant at the nozzle entrance. When water was used as a coolant, 16 percent of the total flow was required to cool the entire nozzle. With liquid ammonia as a coolant, 19 percent of the total flow was required to cool the entire nozzle.

2. The maximum specific impulse obtained when the entire nozzle was film-cooled with liquid ammonia was 162 pound-seconds per pound or 78 percent of the maximum specific impulse obtained from a corresponding uncooled engine. This reduction in performance is of sufficient magnitude to make film cooling of the entire nozzle by annular injection of ammonia at the nozzle entrance appear undesirable.

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TABLE I - PERFORMANCE DATA FROM ROCKET-NOZZLE FILM-COOLING EXPERIMENTS

Run	Oxidant flow W_o (lb/sec)	Fuel flow W_f (lb/sec)	Coolant flow W_l (lb/sec)	Thrust F (lb)	Combustion-chamber pressure P_c (lb/sq in. abs)	Oxidant-fuel ratio neglecting coolant W_o/W_f	Coolant flow $100W_l/(W_o+W_f+W_l)$ (percent)	Specific impulse including coolant I ((lb)(sec)/lb) (a)
Coolant, water								
1	2.58	1.87	----	826	233	1.38	----	---
2	2.60	1.92	1.34	829	232	1.35	22.8	144
3	2.85	1.55	1.35	810	232	1.84	23.6	144
4	2.51	1.67	1.29	816	230	1.50	23.6	152
5	3.00	1.71	1.13	894	245	1.75	19.3	155
6	2.90	1.89	.70	905	250	1.53	12.7	165
Coolant, anhydrous liquid ammonia								
7	2.53	1.70	1.25	868	240	1.49	22.8	159
8	2.91	1.51	1.11	875	245	1.93	20.0	160
9	3.10	1.34	1.07	890	250	2.32	19.4	161
10	2.95	1.40	1.10	858	245	2.10	20.2	159
11	2.82	1.51	1.11	851	240	1.86	20.4	157
12	2.73	1.40	1.17	855	248	1.95	22.1	161
13	2.90	1.29	1.02	780	225	2.25	19.6	154
14	2.98	1.18	.85	787	230	2.53	16.9	160
15	3.12	1.06	1.04	792	225	2.95	20.0	156
16	3.53	.89	1.43	736	220	3.98	24.4	130
17	3.26	.93	1.28	763	220	3.53	23.4	144
18	2.81	1.93	1.17	986	272	1.45	19.8	164
19	3.00	1.76	.91	1004	275	1.71	16.0	174

[a]Corrected to a combustion-chamber pressure of 250 lb/sq in. abs.

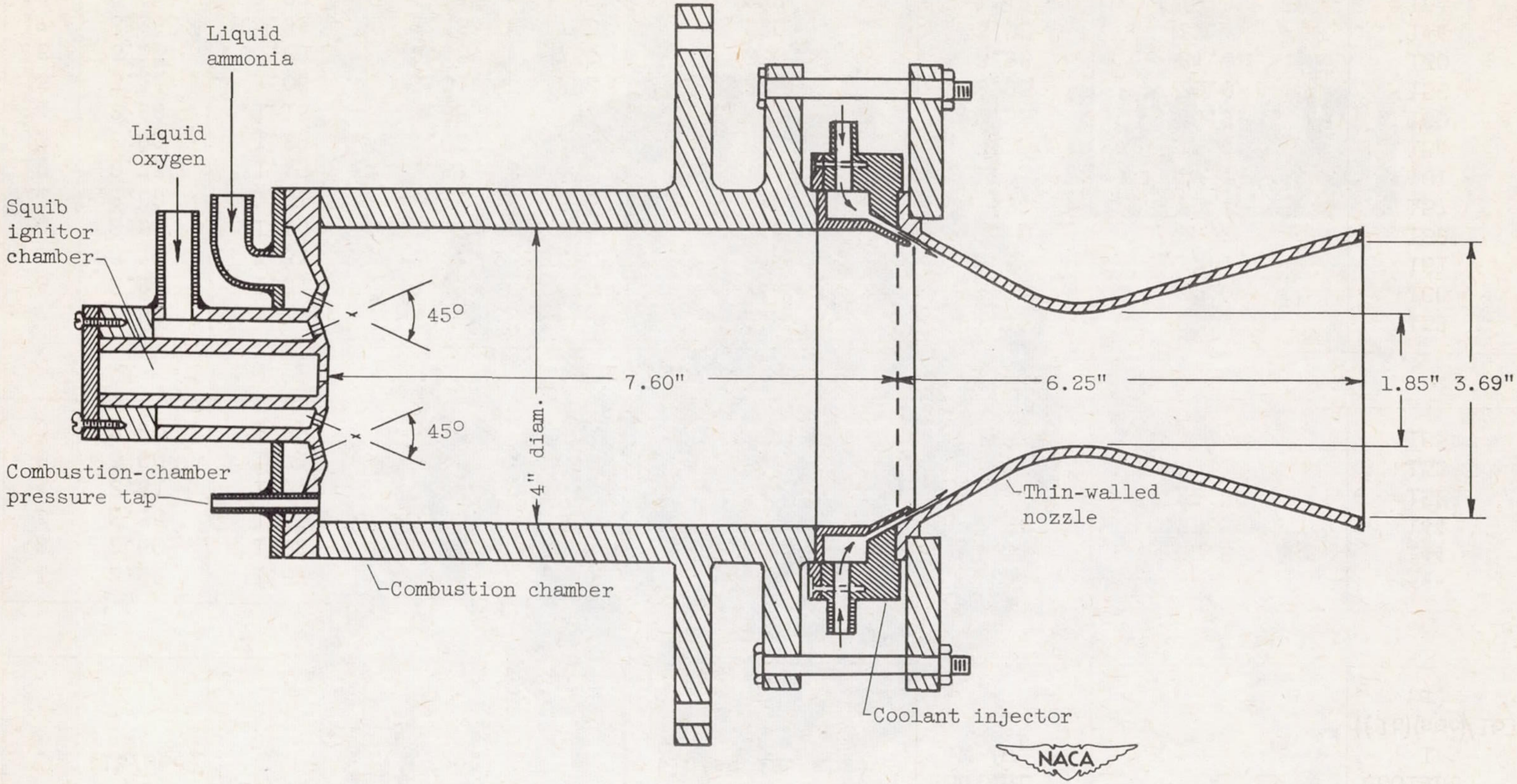


Figure 1. - Rocket engine used for investigation of nozzle film cooling.

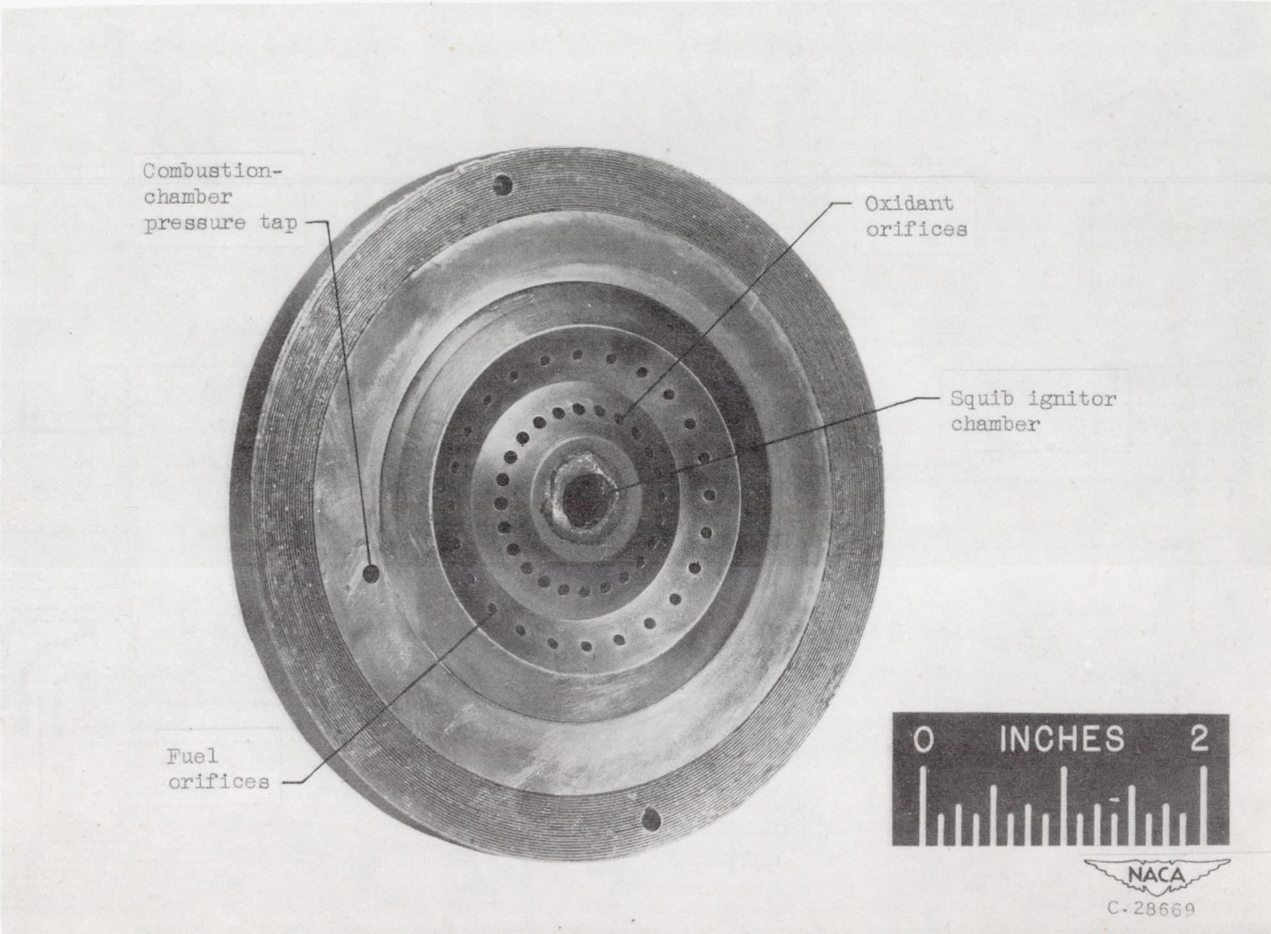


Figure 2. - Propellant injector.

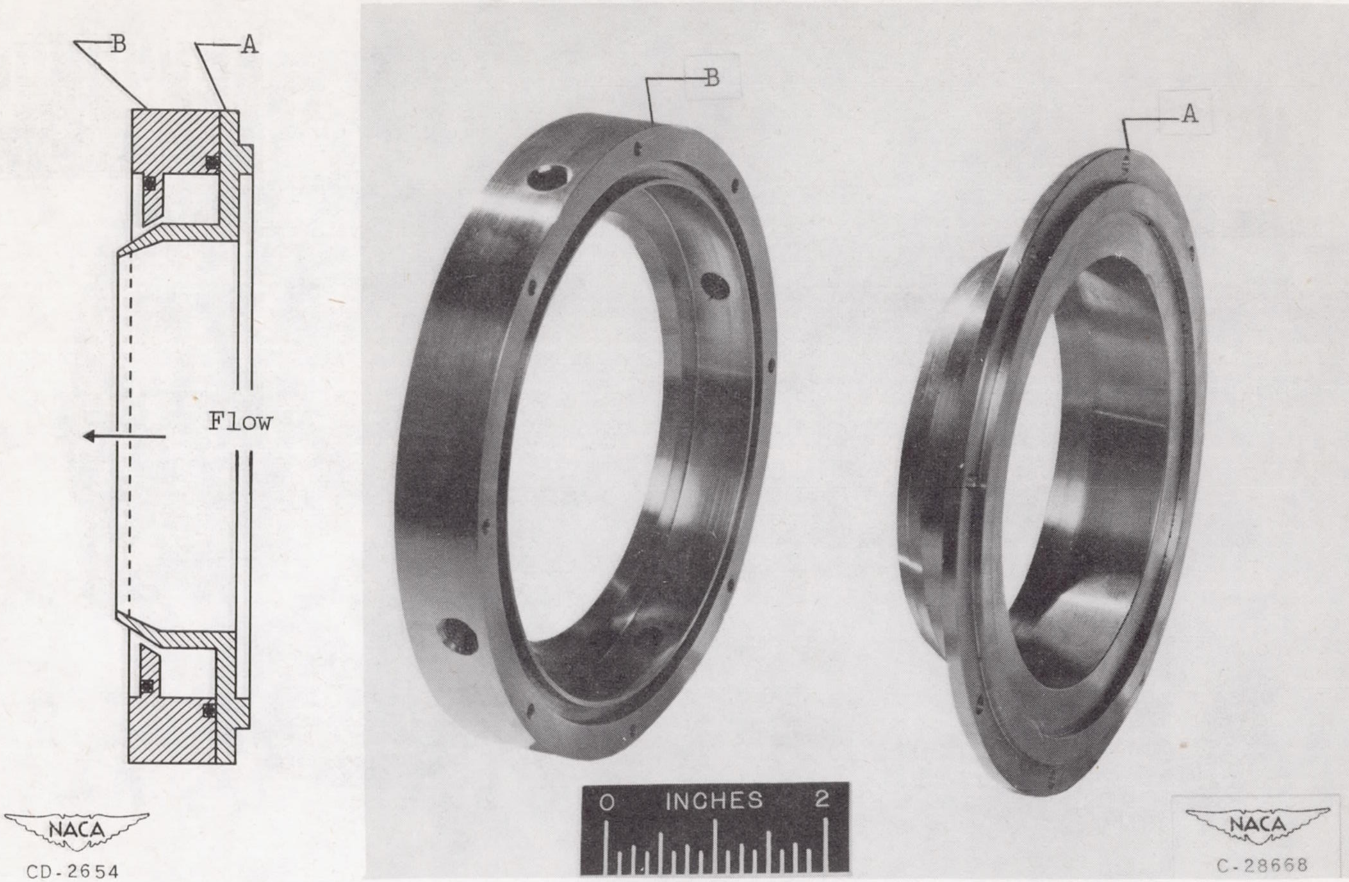


Figure 3. - Component parts of coolant injector.

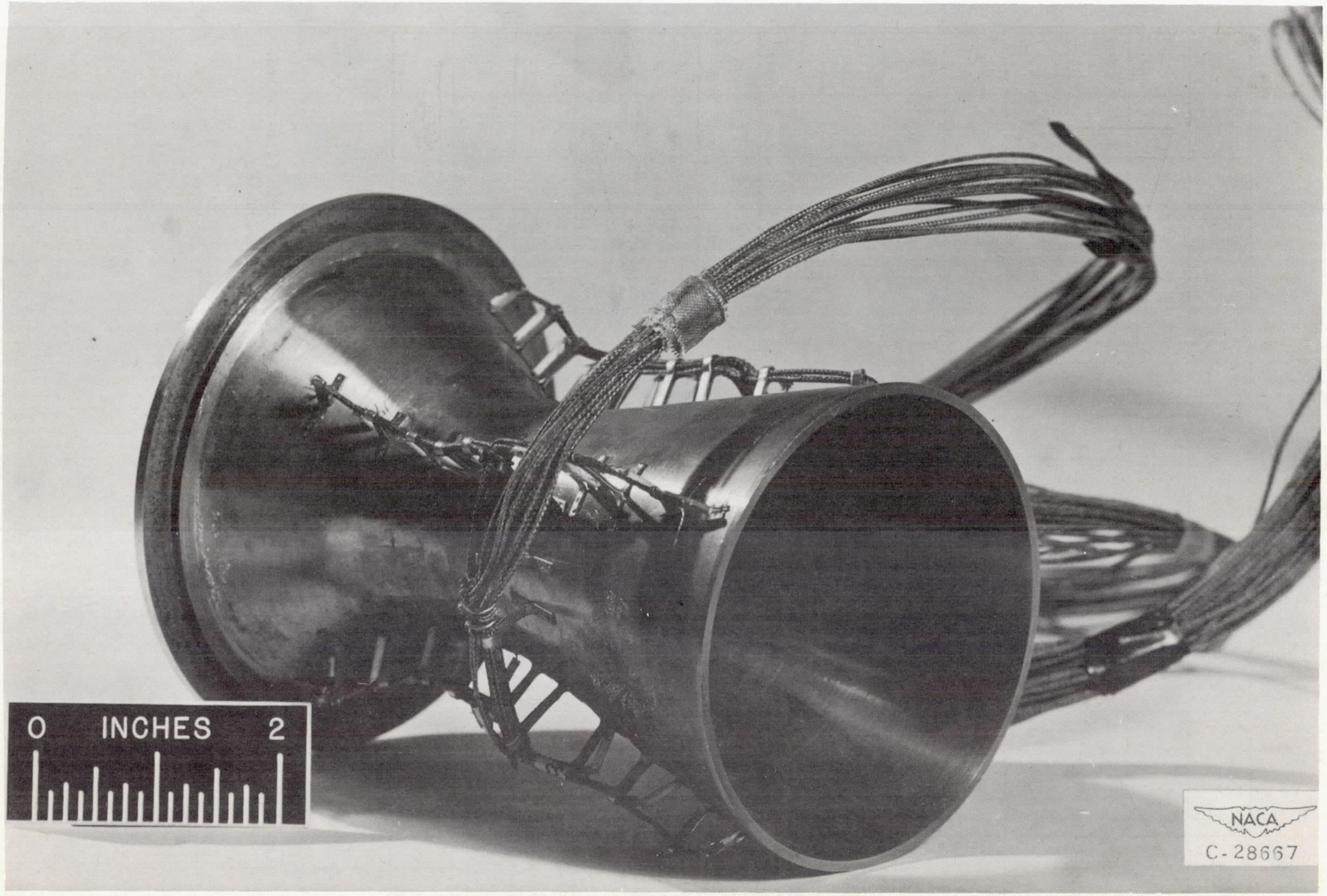
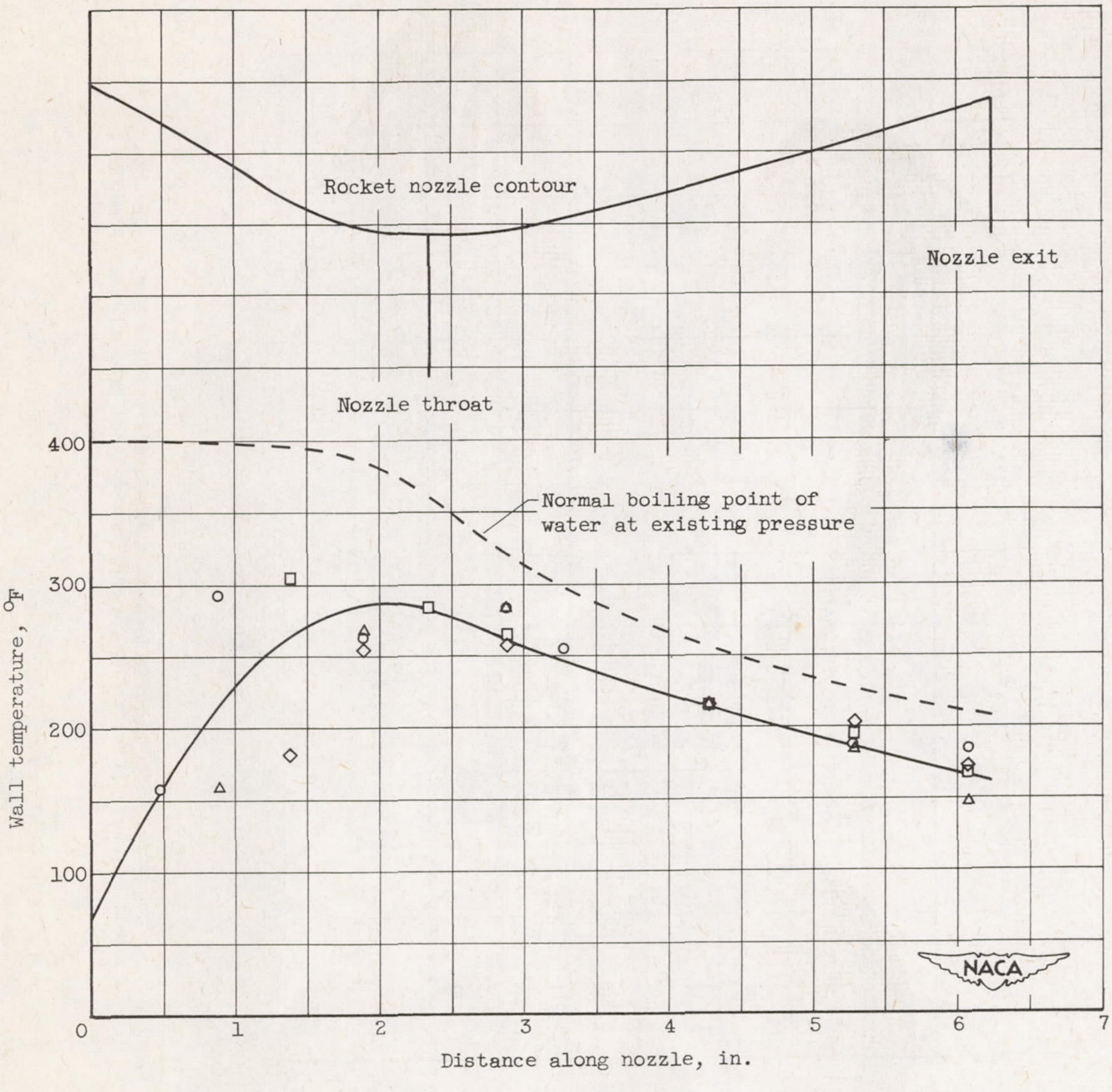
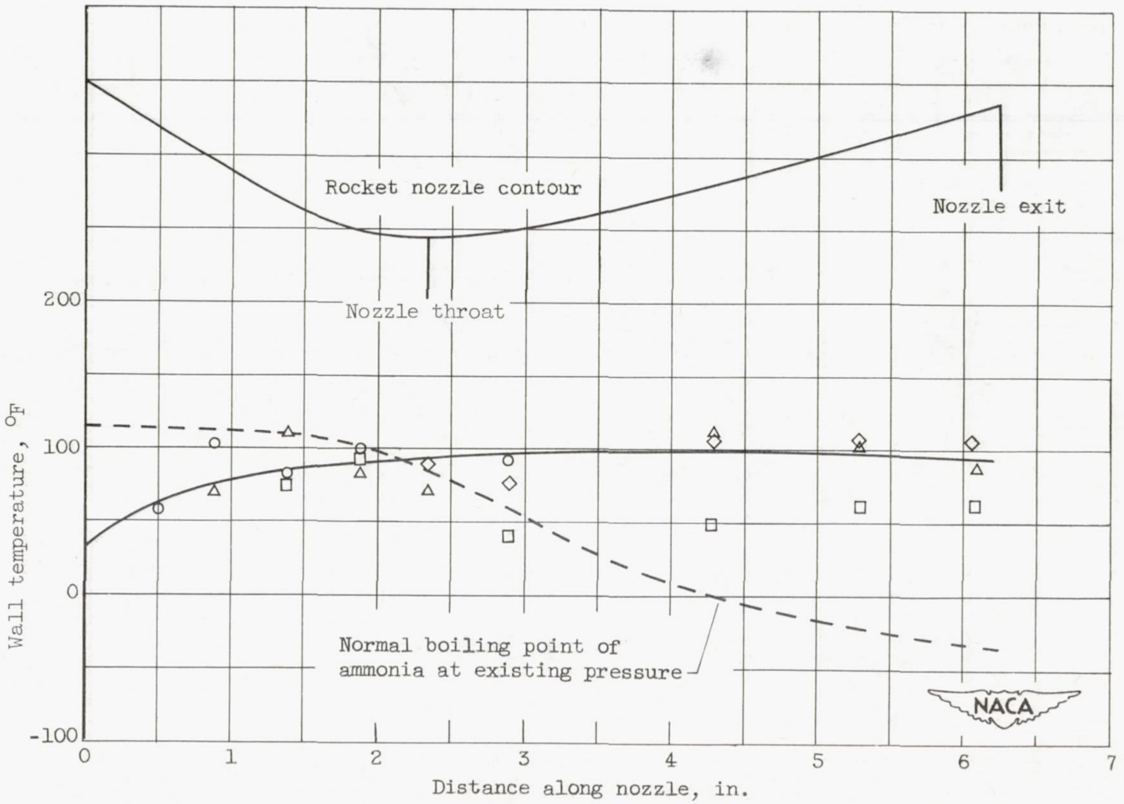


Figure 4. - Thin-walled stainless-steel exhaust nozzle.



(a) Coolant, water.

Figure 5. - Typical nozzle wall temperature profile for entire nozzle film-cooled. Symbols refer to different longitudinal rows of thermocouples.



(b) Coolant, liquid ammonia.

Figure 5. - Concluded. Typical nozzle wall temperature profile for entire nozzle film-cooled. Symbols refer to different longitudinal rows of thermocouples.

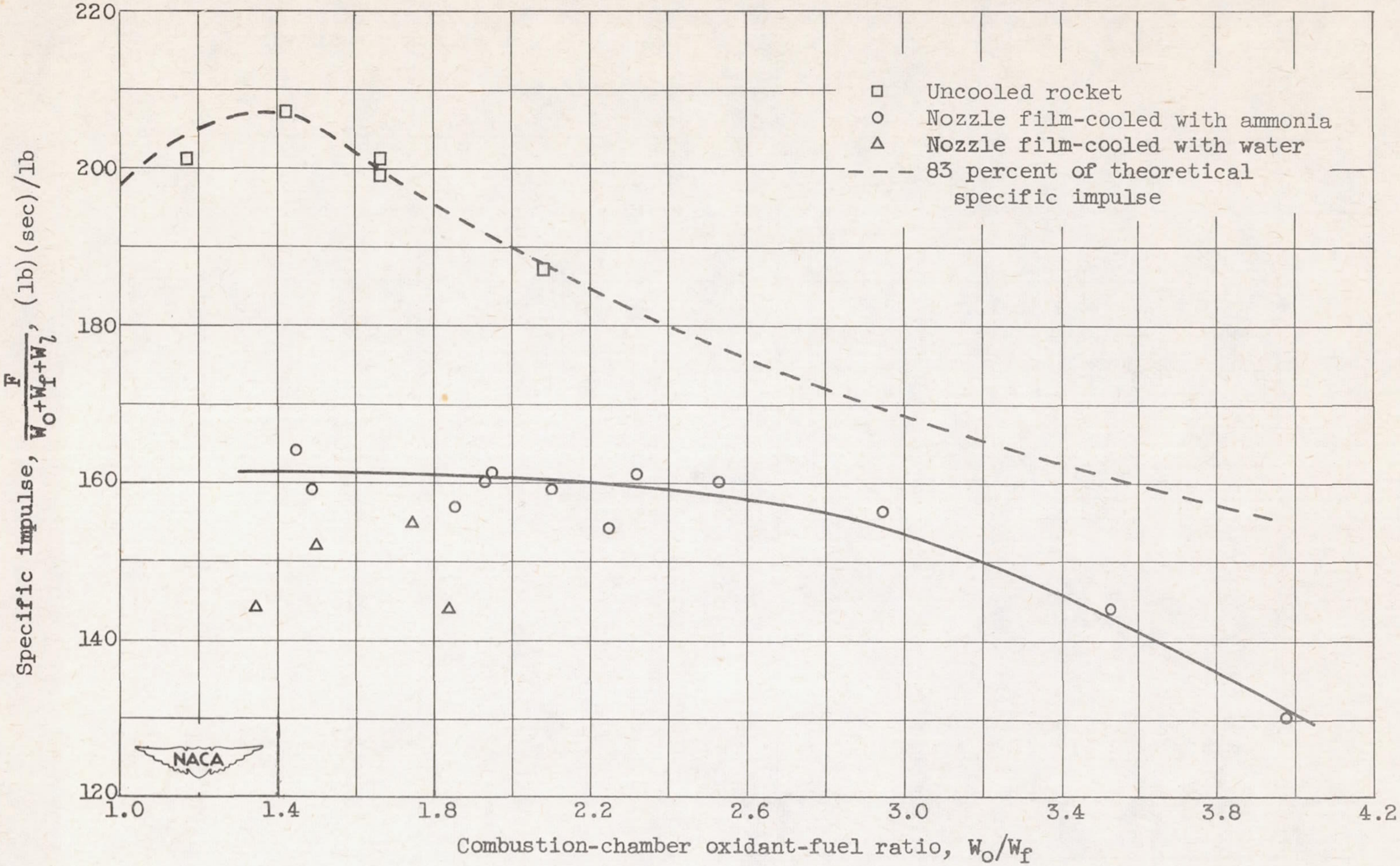


Figure 6. - Comparison of performance of uncooled rocket engine with performance obtained when exhaust nozzle was film-cooled with water or liquid ammonia. Specific impulse values are corrected to combustion-chamber pressure of 250 pounds per square inch absolute with expansion to 1 atmosphere. Coolant flow varies from 16.9 to 24.4 percent of total flow.

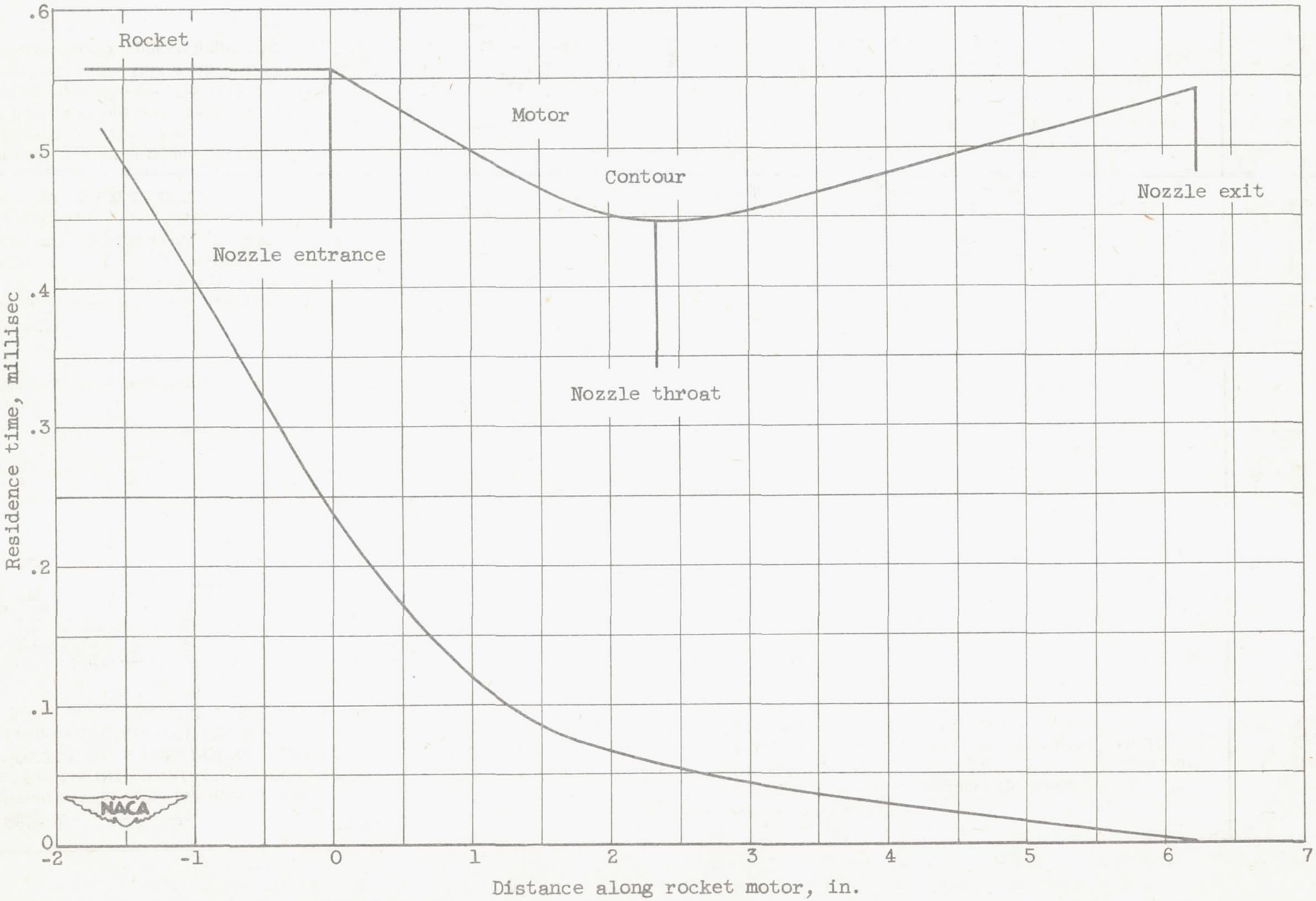


Figure 7. - Particle residence time from nozzle exit with assumption of uniform one-dimensional flow. Combustion-chamber pressure, 250 pounds per square inch absolute; total temperature, 4560° F.