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PRELIMINARY EVALUATION OF FLIGHT-WEIGHT XRJ47-W-5

RAM-JET ENGINE AT A MACH NUMBER OF 2.75

By Henry J. Welna and Dwight H. Reilly

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

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RESEARCH MEMORANDUM

PRELIMINARY EVALUATION OF FLIGHT-WEIGHT XRJ47-W-5 RAM-JET ENGINE AT A MACH NUMBER OF 2.75

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SUMMARY

A free-jet investigation of the performance, burner-shell cooling, and ignition characteristics of a flight-weight 48-inch-diameter XRJ47-W-5 ram-jet engine was conducted in a Lewis altitude test chamber at an inlet Mach number of 2.75 and an angle of attack of 3°. Data were obtained over a range of altitudes from 58,000 to 73,000 feet (engine-inlet air flows of 120 to 60 lb/sec, respectively), a range of inlet temperatures from 860° to 990° R, and a range of fuel-air ratios from lean blow-out to about 0.050. The range of combustor-inlet total pressures for these conditions was from 2150 to 960 pounds per square foot absolute.

A combustion efficiency of 0.85 was obtained at an air flow of 60 pounds per second, inlet temperature of 990° R, and the approximate design fuel-air ratio of 0.043. The combustor total-pressure ratio at this condition was 0.80. This combustion efficiency and combustor total-pressure ratio closely compare with the performance of the heavy-duty version of this engine previously investigated. At an inlet temperature of 990° R, flight-engine lean blow-out occurred between a fuel-air ratio of 0.027 to 0.028 for the range of altitude investigated.

Three methods were investigated for mounting the cooling liners; the most promising one was used in the engine for approximately $3\frac{1}{2}$ hours of burning operation without significant damage. The only consistently reliable method of igniting the engine was spontaneous ignition of 1 to 6 pounds of aluminum trimethyl injected into the engine combustor. This method was reliable only when the injection rate exceeded 2000 pounds per hour, and when the specific gravity of the aluminum trimethyl was 0.85.

INTRODUCTION

The 48-inch-diameter XRJ47-W-5 ram-jet engine is being developed for use in the Navaho II missile as part of the MX-770 long-range surface-to-surface missile program sponsored by the U.S. Air Force. Full-scale direct-connect and free-jet investigations of the engine are being conducted in a Lewis altitude test chamber at simulated altitude conditions that will be encountered in the cruise portion of the missile flight plan. The direct-connect and free-jet investigations of the heavy-duty test engine have been completed and the results are presented in references 1 and 2.

The purpose of the investigation reported herein was to evaluate the flight-weight engine version of the XRJ47-W-5 engine. This evaluation included the determination of the performance, burner-shell cooling characteristics, liner durability, and starting characteristics. The internal geometry of the engine used in this investigation conformed to the final combustor configuration arrived at during the development of the combustor in the heavy-duty engine earlier in the program (ref. 2).

Performance data were obtained over a range of air flows from 120 to 60 pounds per second, nominal inlet temperatures from 860° to 990° R and fuel-air ratios from about 0.050 to lean blow-out. These values of air flow, inlet temperature, and fuel-air ratio correspond to a range of combustor-inlet total pressures of 2150 to 960 pounds per square foot absolute.

Three cooling-liner configurations were investigated. Liner length and average cooling-passage height were approximately the same for all configurations. Principal differences included the type of liner corrugations and the method of securing the liner in the combustor. Three different ignition sources were evaluated. One ignition source consisted of flares attached to the flameholder. The other two sources were injection of small quantities of boron triethyl and aluminum trimethyl into the combustor in the region of the flameholder.

APPARATUS

Engine and Installation

The XRJ47-W-5 ram-jet flight engine was installed in the 14-foot-diameter test section of an altitude test chamber (fig. 1). The facility air system supplied dried and heated air which entered the supersonic nozzle and was accelerated to a Mach number of 2.75 at the entrance of the inlet duct. The operation and performance details of the facility are presented in reference 3.

The installation consisted of a heavy-duty supersonic inlet diffuser and the flight-weight conical engine diffuser, combustor, and exhaust nozzle. A diagram of the flight engine without the supersonic inlet diffuser is shown in figure 2. The supersonic inlet diffuser was designed for a Mach number of 2.75; performance of this diffuser is presented in reference 4. Attached to the supersonic diffuser outlet was the 30° conical engine diffuser, which was 29.9 inches long and terminated at the entrance to the 48-inch-diameter combustor. The combustor was 60.4 inches long and was made of 0.060-inch Inconel. Affixed to the combustor was a convergent-divergent exhaust nozzle. Two different exhaust nozzles were used during this investigation. The flight nozzle (fig. 2) was 42.2 inches long with a throat diameter of 38.8 inches and a 48-inch exit diameter (nozzle-throat to combustor-area ratio, 0.66). This nozzle was used during the entire engine investigation, except the performance investigation when a heavy-duty convergent-divergent nozzle that contained total-pressure rakes was installed. The heavy-duty nozzle had the same dimensions as the flight nozzle and thus had the same nozzle-throat to combustor-area ratio.

The flameholder system is illustrated in figure 2 by a schematic diagram and in figure 3 by a photograph. The flameholder configuration consisted of a center pilot and three annular V-gutters interconnected by slanted radial V-gutters. The fuel system consisted of the pilot fuel and main fuel system, both using a common fuel supply. The pilot burner contained three variable-area-type fuel nozzles. The main fuel system had three fuel rings of 13-, 20-, and 26-inch diameter fitted with 10, 15, and 40 variable-area-type fuel-spray nozzles, respectively. The nozzles were rated at 35 gallons per hour at a differential pressure of 100 pounds per square inch. All the fuel was sprayed downstream. The fuel used during this investigation was MIL-F-5624B, grade JP-5, which has a lower heating value of 18,625 Btu per pound and a hydrogen-carbon ratio of 0.159.

Various methods of mounting the cooling liner in the engine were investigated. The geometry of all the liners was the same (fig. 2). The average cooling passage height was 0.78 inch except at the liner inlet where the liner was slightly flared to increase the cooling passage area. The cooling liner extended from 2.4 inches downstream of the engine diffuser inlet to 10 inches downstream of the combustor outlet. All liners were made of 0.030-inch stainless steel.

Engine starting tests were made with flares installed along the radial interconnecting V-gutters in the combustor. Flares were electrically ignited with the flares pointing upstream in some cases and downstream in others. Special fuels, which included boron triethyl and aluminum trimethyl, were used as a source of spontaneous combustion in order to start the engine. The boron triethyl had a specific gravity of 0.69. Two types of aluminum trimethyl were used, one having a specific gravity of 0.85 and the other a specific gravity of 1.3. Generally, about 1 to 6 pounds of special fuel was injected through one or more of the flameholder support pins as illustrated schematically in figure 2 and photographically in figure 3. The points of injection are also indicated in figure 3.

Instrumentation

Location of the engine instrumentation and amount of instrumentation at each station are shown in figure 1. The fuel flow was measured by a variable-area orifice. Thermocouples (not shown) were placed on the engine shell, liner supports, in the cooling-air passage, and on the cooling liner. The liner thermocouples were located at four stations along the combustor with at least three thermocouples placed at the circumference of each station. Temperature readings were taken only to indicate the order of metal temperatures and exact trends were not determined.

PROCEDURE

Flight Conditions and Engine Operation

The investigation was conducted over a range of engine-inlet air flows from 60 to 120 pounds per second, which correspond to altitudes of 73,000 to 58,000 feet, respectively, at a Mach number of 2.75 and at inlet temperatures of 860°, 935°, and 990° R. The range of fuel-air ratios investigated for engine performance was from lean blow-out to about 0.050. The cooling-liner durability was investigated at a fuel-air ratio of approximately 0.045 and an inlet temperature of 990° R. The engine diffuser operated supercritically for all fuel-air ratios investigated.

The desired engine-inlet air flow was obtained by adjusting the total pressure and temperature at the free-jet supersonic-nozzle inlet. The free-jet supersonic nozzle was first started, the facility exhaust pressure was decreased until the engine nozzle was choked, the engine fuel-air ratio was set, and then the engine was ignited by either flares or special fuels. After ignition, the fuel flow was varied over the operable range to obtain the performance data. The symbols and methods of calculation are given in appendixes A and B, respectively.

RESULTS AND DISCUSSION

Engine Performance

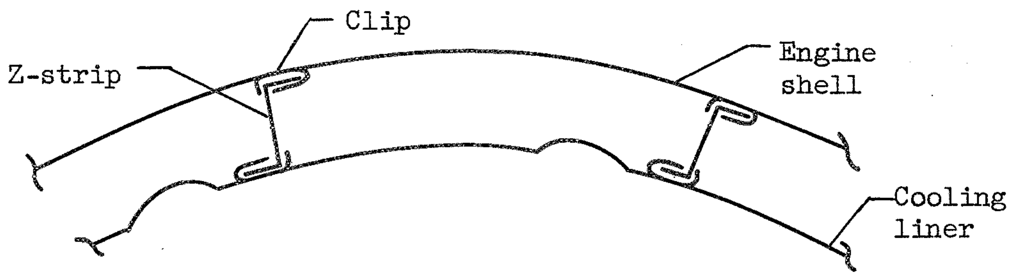
The performance of the flight engine for a range of inlet temperature is presented in figures 4 to 7 for engine-inlet air flows of 60, 80, 110, and 120 pounds per second. The performance data are also presented in tabular form in table I. The combustion efficiency, combustor-outlet total pressure, combustor total-pressure ratio, and diffuser total-pressure recovery are shown as a function of fuel-air ratio. In general, at a given air flow, the combustion efficiency increased while the combustor total-pressure ratio and diffuser total-pressure recovery decreased with an increase in inlet temperature. At approximately the design fuel-air ratio of 0.043 and an inlet temperature of 990° R, the combustion efficiency varied from 0.85 at an air flow of 60 pounds per second to about 0.88 at an air flow of 110 pounds per second. The corresponding combustor total-pressure-ratio variation with air flow at this fuel-air ratio was 0.795 to 0.812. This performance compares closely with the performance of the heavy-duty version of this engine as reported in reference 2.

The fuel-air ratio at which lean blow-out occurred decreased (fig. 4) as the inlet temperature was increased at a given altitude. At an altitude of 73,000 feet, the fuel-air ratio at which blow-out occurred decreased from 0.032 to 0.028 as inlet temperature was increased from 860° to 990° R. There was only a slight effect of altitude on lean blow-out as shown in figure 8. At an inlet temperature of 990° R, blow-out occurred between a fuel-air ratio of 0.027 to 0.028 for the range of altitudes investigated.

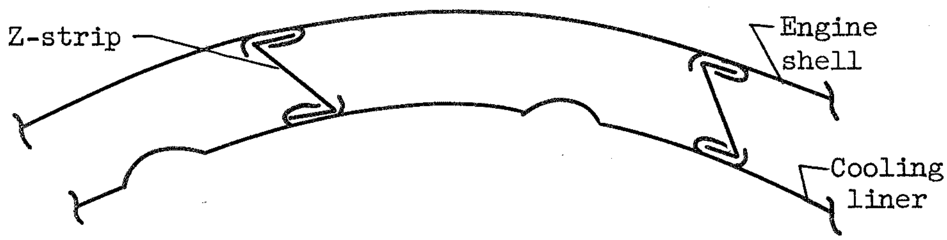
Engine Cooling

The selection of a satisfactory method for attachment of the cooling liner to the burner shell was the primary purpose of the liner investigation reported herein. The required cooling-liner length and cooling-passage height were determined in the liner tests made during the previous investigation, which is reported in reference 2.

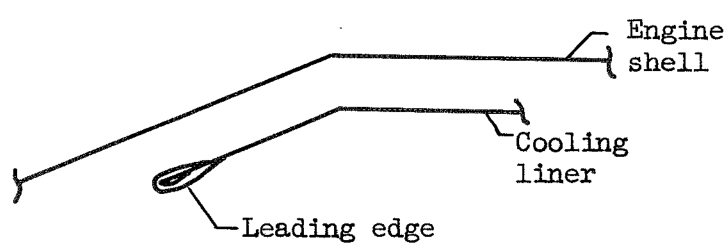
The initial liner was made of stainless steel (AISI 310) which was 0.030 inch thick. The method of liner mounting, as provided by the manufacturer, was to attach clips to both the engine shell and liner with a Z-strip retainer. The method of liner attachment in the 30°-engine-diffuser section is shown here, while the mounting details of the



liner in the combustor are illustrated in the following sketch:



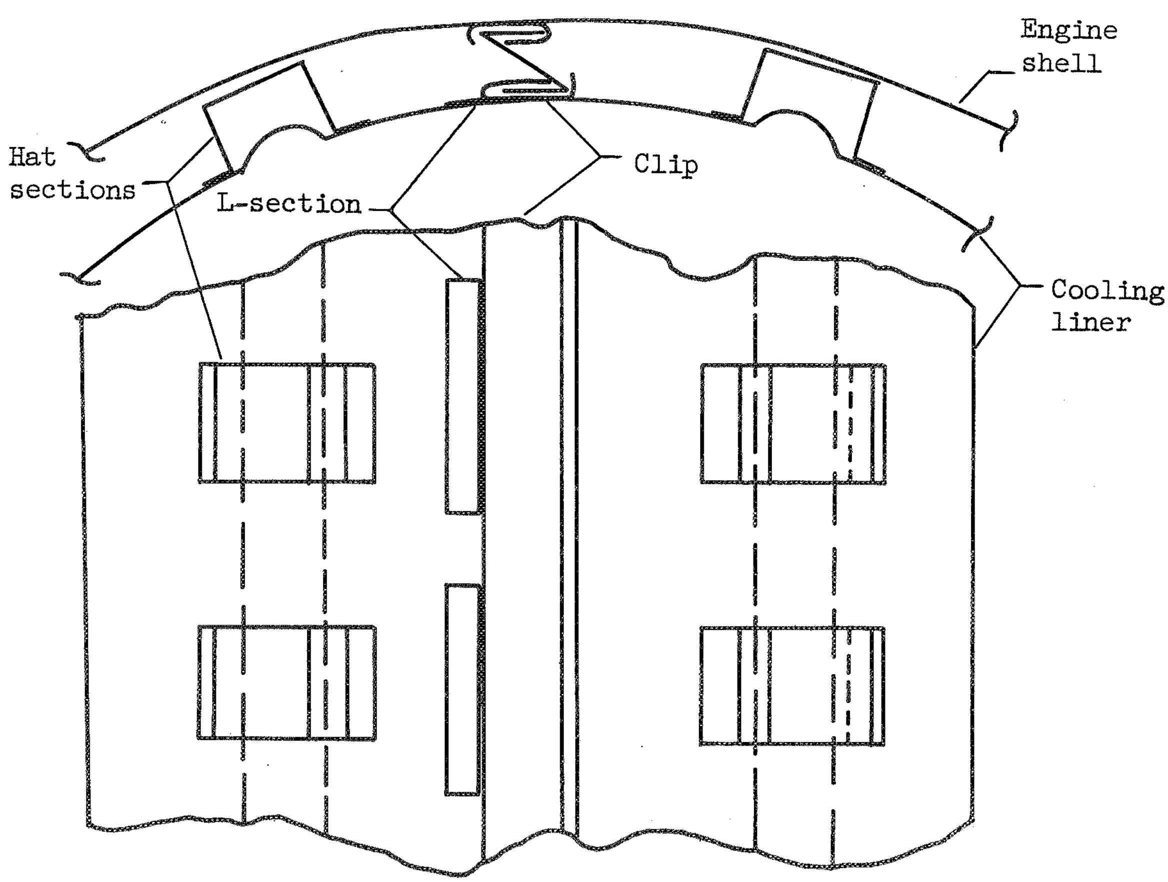
The liner was damaged (fig. 9) in the 30°-engine-diffuser section when a number of the Z-strips rotated out of the clips after about 2 hours of cold-flow engine operation at an altitude of 58,000 feet. In an effort to prevent the Z-strips from rotating out, new clips were installed on the cooling liner in the 30° conical engine diffuser beside the old clips, and new Z-strips, which conformed to those installed in the combustor section, were inserted. Additional cold-flow operation, however, resulted in the opening of the liner leading edge, which was repaired by wrapping and welding sheet metal around the leading edge as illustrated in the following sketch:



During succeeding operation, differences in the thermal expansion of the clips and liner resulted in ripples forming in the cooling liner after approximately 1/2 hour of burning time at an altitude of 73,000 feet. This damage is shown in figure 10. In a few places the liner pulled away from the engine shell. The liner was damaged extensively when an additional 33-minute burning run was made at an altitude of 58,000 feet.

During this run, hot spots were noticeable on the burner shell. Maximum engine shell temperatures of 900° R and a difference of 600° R between the shell and cooling-liner metal temperature were measured. The liner became detached from the burner shell completely around the downstream end and most of the spot welds holding the clips pulled out. The condition of the liner is shown in figure 11 after a total of 1 hour and 33 minutes of burning time. Slight bulges also occurred in the exhaust nozzle and engine shell.

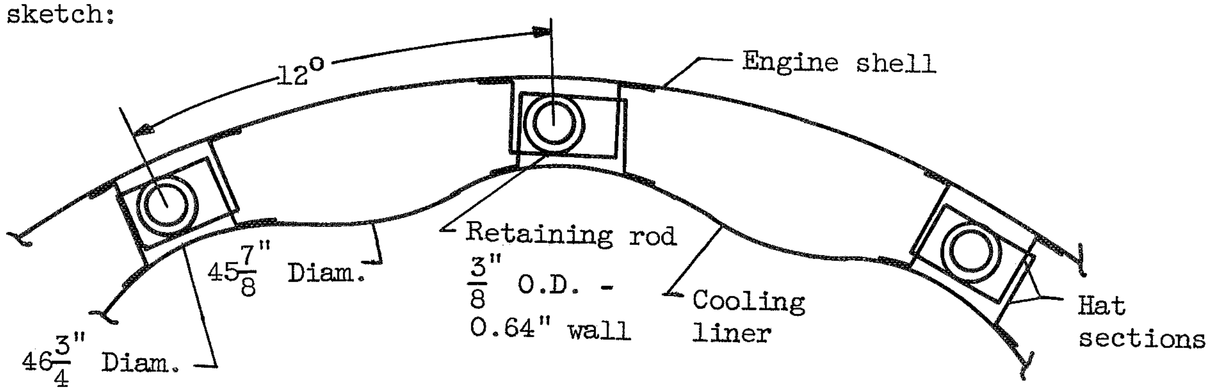
The damaged liner was removed and a completely new liner and combustor shell were installed. This second liner was modified in the 30°-engine-diffuser section with the new type of Z-strips previously described. In addition, $2\frac{1}{2}$-inch L-strips were welded to the clips and liner for added support. Small "hat" sections were spot welded to the liner, stradling the corrugations throughout the downstream half of the liner. These modifications are shown by the following schematic diagram and also by figure 12.



The leading edge of the second liner, which was formed by overlapping the liner metal, opened up as with the earlier liner after 22 minutes of burning operation at an altitude of 73,000 feet (fig. 13). This damage was again repaired by wrapping and welding sheet metal around the leading edge. With an additional 2-hour burning run, small ripples were present in the liner as shown in figure 14, and in addition, hot streaks were observed on the engine shell as much as 20 inches upstream of the exhaust-nozzle inlet during burning. A hole was burned in the liner when an additional run was made at an altitude of 58,000 feet for 69 minutes, making a total of 3 hours and 31 minutes of burning operation with this cooling liner. This failure is shown in figure 15. It was noted that many spot welds also failed throughout the liner.

Another liner of the same design as the second was installed and the engine shell was covered with an aluminum-foil blanket. A total of 1 hour and 43 minutes of burning time was accumulated with operation at altitudes from 58,000 to 73,000 feet when failure occurred (fig. 16). Maximum blanket outside-surface temperatures of 1000° F were obtained.

In an attempt to arrive at a more reliable cooling-liner design, the combustion-chamber and exhaust-nozzle portion of the liner was replaced by the liner configuration shown in figure 17 and the following sketch:



The liner was made of stainless steel that was 0.030 inch thick. The liner was corrugated with hat sections spot welded on the crests of the corrugations. Similar hat sections were placed on the engine shell. The liner was inserted into the engine shell and rods were placed through the hat sections to retain the liner in place. To simplify the modification, the section of the liner in the 30°-engine-diffuser section was not altered. During operation hot streaks were seen along the unblanketed engine shell. Maximum engine-shell temperatures of about 1100° F were recorded. After approximately 3 hours of operation at altitudes of 60,000 and 67,000 feet, only a few small cracks developed in the liner trailing edge. The unmodified section of the liner (30° transition section) failed as shown in figure 18, and after an additional run of 35 minutes at an altitude of 73,000 feet, this transition section completely buckled.

The portion of the cooling liner in the combustor was not damaged and there was no sign of deterioration or failure in the modified portion of the liner. This method of mounting the cooling liner resulted in approximately $3\frac{1}{2}$ hours of burning operation, although the liner could have been used longer if the unmodified 30°-engine-diffuser section had not failed. Therefore, this liner design was believed to be the most reliable of those investigated.

Engine Ignition

During cold-flow operation at inlet conditions corresponding to altitudes of 58,000 to 73,000 feet and with an unthrottled exhaust-nozzle area, it was found during an earlier portion of the program (ref. 2) that ignition was impossible with spark plugs because of the low pressure and high velocities existing in the combustor. The use of flares and special fuels were thus adopted as alternative methods for ignition. The starting tests reported herein were made at an inlet temperature of 410° F. This temperature was selected because it corresponds to the most critical condition for ignition, the cold day temperature at the site where the missile is to be flight tested.

The flares, varying in number from four to ten, were installed along the radial interconnecting flameholder gutters in the combustor to provide the ignition source. A summary of the starting attempts made with flares is given in table II. The engine-inlet conditions and the number, position, and type of flares used for each starting attempt are shown in this table. The chemical composition of the three types of flare that were used is given in table III.

The flares were considered unreliable as a source of ignition. Of the thirteen starting attempts that were made with the flares, in five cases all flares did not ignite, and in only five cases did the combustor ignite. These results substantiate earlier flare-starting experience during the heavy-duty engine investigation and in subsequent experience with flares during flight-weight engine tests.

The various starting attempts that were made with special fuels, boron triethyl and aluminum trimethyl, are summarized in table IV. This table includes the engine conditions, rate of fuel injection, number of injection points, fuel specific gravity, and whether ignition was successful or not. Of two starting attempts with boron triethyl, both were unsuccessful. The engine was ignited in all cases with aluminum trimethyl when the rate of fuel injection was greater than about 2000 pounds per hour for the fuel having a specific gravity of 0.85, and 3900 pounds per hour for fuel having a specific gravity of 1.3. The total quantity of fuel injected varied from 1 to 6 pounds. The results with aluminum trimethyl having a specific gravity of 0.85 were further substantiated by a subsequent investigation with an identical flight-weight engine. Because the most experience by far was obtained with aluminum trimethyl having a specific gravity of 0.85, the results with this fuel are much more reliable than those for the fuel having a specific gravity of 1.3. Additional tests are desirable to further substantiate the starting results with aluminum trimethyl having a specific gravity of 1.3.

SUMMARY OF RESULTS

An investigation was conducted in an altitude test chamber to determine the performance, engine-burner-cooling, liner-design, and ignition characteristics of the XRJ47-W-5 48-inch-diameter ram-jet flight-weight engine.

The engine combustion efficiency varied from 0.85 at an air flow of 60 pounds per second to about 0.88 at an air flow of 110 pounds per second at the approximate design fuel-air ratio of 0.043 and an inlet temperature of 990° R. The corresponding combustor total-pressure-ratio variation with air flow at this fuel-air ratio was 0.795 to 0.812. Combustor lean blow-out occurred at a fuel-air-ratio range from 0.027 to 0.028 at an inlet air temperature of 990° R.

Three liner configurations differing in the type of liner corrugations and liner support were investigated. With the best configuration, the combustor section of the liner withstood approximately $3\frac{1}{2}$ hours of burning operation without significant damage.

Aluminum trimethyl was successful as a source of ignition at all engine-inlet conditions investigated provided that the rate of injection exceeded about 2000 pounds per hour and the specific gravity of fuel was 0.85. The flares were considered unreliable as an ignition source. The two attempts made with boron triethyl were unsuccessful.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, July 26, 1955

APPENDIX A

SYMBOLS

The following symbols are used in this report:

A	area, sq ft
C_A	exhaust-nozzle-area coefficient
C_N	exhaust-nozzle-expansion coefficient
f/a	fuel-air ratio
g	acceleration due to gravity, 32.2 ft/sec^2
P	total pressure, lb/sq ft abs
R	gas constant, 53.4 ft-lb/(lb)($^{\circ}$R)
T	total temperature, $^{\circ}$R
W	flow rate, lb/sec
γ	ratio of specific heats
η	combustion efficiency

Subscripts:

a	air
b	combustor
f	fuel
g	gas
0	free-jet-nozzle inlet
3	engine-diffuser outlet
5	exhaust-nozzle throat

APPENDIX B

CALCULATIONS

Air flow. - Throughout the investigation the engine diffuser was operated supercritically; therefore, the relation between inlet conditions and engine air flow was unique. The equation, which was obtained by calibrating the engine air flow, is

$$W_a = \frac{0.979\ P_0}{\sqrt{T_0}}$$

Fuel-air ratio. - The fuel-air ratio was calculated directly from the measured fuel flow and air flow:

$$\frac{f}{a} = \frac{W_f}{W_a}$$

Combustion efficiency. - The combustor temperature ratio was first determined from the total pressure at the exhaust-nozzle-throat area and inlet total pressure. The choked-nozzle equation and continuity equation is

$$W_{g,5} = \sqrt{\frac{g\gamma_5}{RT_5}}\ \frac{P_5 C_A C_N}{\left(\frac{\gamma_5 + 1}{2}\right)^{\frac{\gamma_5+1}{2(\gamma_5-1)}}} = W_a\left(1 + \frac{f}{a}\right)$$

But $W_a = \dfrac{0.979\ P_0}{\sqrt{T_0}}$, and therefore,

$$\sqrt{\frac{T_5}{T_0}} = \frac{1}{0.979}\left(\frac{P_5}{P_0}\right)\frac{C_A C_N A_5}{\left(1 + \frac{f}{a}\right)}\sqrt{\frac{g\gamma_5}{R\left(\frac{\gamma_5 + 1}{2}\right)^{\frac{\gamma_5+1}{\gamma_5-1}}}}$$

where C_A was determined from cold-flow tests and a value of 0.987 was used. The value of C_N was obtained from the exhaust-nozzle temperature and the expansion coefficient of Inconel. A trial-and-error method was used in determining γ_5 for the final T_5 and appropriate fuel-air ratio. Ideal fuel-air ratio was determined from ideal temperature-rise curves and the value of $\sqrt{T_5/T_0}$ and inlet temperature. Combustion efficiency was then calculated as the ratio of ideal to actual fuel-air ratio.

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TABLE I. - PRELIMINARY PERFORMANCE OF XRJ47-W-5 RAM-JET FLIGHT ENGINE AT MACH NUMBER 2.75

Run	Free-stream total temperature, T_0, °R	Free-stream total pressure, P_0, lb/sq ft abs	Engine-inlet air flow, $W_{a,0}$, lb/sec	Diffuser-outlet total pressure, P_3, lb/sq ft abs	Exhaust-nozzle total pressure, P_5, lb/sq ft abs	Diffuser total pressure recovery, P_3/P_0	Engine total-pressure ratio, P_5/P_0	Combustor total pressure ratio, P_5/P_3	Fuel flow, W_f, lb/sec	Fuel-air ratio, $\frac{W_f}{W_{a,0}}$	Combustion efficiency, η_b
Altitude, 58,000 ft											
1	853	3537	118.6	1608	791	0.455	0.224	0.492	0	------	-----
2	859	3621	120.9	1688	837	.466	.231	.496	0	------	-----
3	864	3560	118.6	1644	814	.462	.229	.495	0	------	-----
4	866	3563	118.5	1651	819	.463	.230	.496	0	------	-----
5	861	3586	119.7	1867	1459	.521	.407	.782	3.57	0.0298	0.846
6	858	3597	120.2	1920	1541	.534	.428	.803	4.11	.0342	.854
7	871	3565	118.3	1933	1557	.542	.437	.806	4.31	.0364	.865
8	859	3591	120.0	1981	1622	.552	.452	.819	4.79	.0399	.855
9	849	3597	120.8	2126	1746	.591	.485	.821	5.98	.0495	.840
10	940	3785	120.2	1920	1497	.507	.398	.780	3.55	.0295	.875
11	934	3767	120.7	2010	1579	.534	.419	.786	4.13	.0342	.886
12	934	3765	120.6	2040	1648	.542	.438	.808	4.74	.0393	.870
13	938	3760	120.2	2081	1704	.554	.453	.819	5.35	.0445	.858
14	928	3765	121.0	2153	1758	.572	.467	.817	5.92	.0489	.845
15	998	3857	119.5	2119	1729	.549	.448	.816	5.36	.0448	.886
Altitude, 60,000 ft											
16	862	3397	113.3	1579	1579	0.4648	0.2208	0.475	0	------	-----
17	992	3520	109.4	1618	1618	.4597	.2259	.491	0	------	-----
18	992	3537	109.9	1799	1387	.509	.392	.771	3.22	0.0293	0.894
19	991	3540	110.1	1865	1458	.527	.412	.782	3.75	.0341	.894
20	993	3544	110.1	1900	1523	.536	.430	.802	4.33	.0393	.883
21	996	3540	109.8	1935	1583	.547	.447	.818	4.93	.0449	.875
22	994	3531	109.6	1982	1618	.561	.458	.816	5.44	.0496	.857
Altitude, 67,000 ft											
23	865	2379	79.2	1078	547	0.453	0.230	0.507	0	------	-----
24	998	2554	79.2	1186	586	.464	.229	.494	0	------	-----
25	861	2392	79.8	1231	947	.515	.396	.769	2.34	0.0293	0.788
26	859	2397	80.1	1290	1026	.538	.429	.795	2.74	.0342	.854
27	858	2394	80.0	1324	1075	.553	.449	.812	3.12	.0390	.856
28	852	2400	80.5	1377	1119	.574	.466	.813	3.53	.0439	.845
29	854	2403	80.5	1409	1153	.586	.480	.818	3.92	.0487	.831
30	923	2492	80.3	1283	981	.515	.394	.765	2.31	.0288	.854
31	923	2495	80.4	1329	1051	.533	.421	.791	2.76	.0343	.880
32	929	2501	80.3	1372	1091	.549	.436	.795	3.13	.0390	.862
33	935	2500	80.0	1397	1130	.559	.452	.809	3.52	.0440	.859
34	930	2501	80.3	1432	1159	.573	.463	.809	3.92	.0488	.830
35	991	2568	79.9	1311	1002	.511	.390	.764	2.36	.0296	.875
36	984	2573	80.3	1363	1054	.530	.410	.773	2.74	.0341	.871
37	989	2573	80.1	1394	1103	.542	.429	.791	3.14	.0392	.875
38	988	2574	80.2	1419	1137	.551	.441	.801	3.52	.0439	.852
39	993	2574	80.0	1445	1166	.561	.453	.807	3.93	.0491	.829
Altitude, 73,000 ft											
40	857	1795	60.0	962	764	0.536	0.426	0.794	2.08	0.0346	0.828
41	860	1795	59.9	984	802	.548	.447	.815	2.35	.0392	.844
42	883	1805	59.5	993	809	.550	.448	.815	2.36	.0397	.866
43	851	1814	60.9	1027	837	.566	.461	.815	2.60	.0427	.841
44	864	1795	59.8	1023	829	.570	.462	.810	2.62	.0438	.838
45	874	1790	59.3	1035	850	.578	.475	.821	2.92	.0492	.823
46	877	1800	59.5	1044	862	.580	.479	.826	2.93	.0492	.846
47	934	1877	60.1	993	780	.529	.416	.785	2.09	.0348	.845
48	931	1874	60.1	1023	813	.546	.434	.795	2.31	.0384	.865
49	926	1880	60.5	1046	844	.556	.449	.807	2.61	.0432	.840
50	934	1870	59.9	1072	865	.573	.463	.807	2.93	.0489	.828
51	988	1926	60.0	1013	786	.526	.408	.776	2.07	.0345	.855
52	991	1938	60.3	1037	822	.535	.424	.793	2.34	.0388	.856
53	989	1940	60.4	1065	848	.549	.437	.796	2.59	.0429	.844
54	998	1930	59.8	1080	873	.560	.452	.808	2.93	.0490	.833

[redacted]

TABLE II. - SUMMARY OF ENGINE STARTING WITH FLARES

Starting attempt	Flares			Starting condition			Installation		Engine ignited	Remarks
	Number installed	Direction of flame from flare	Type	Air flow, lb/sec	Inlet temperature, °F	Fuel-air ratio, f/a	Along inner slant gutters	Along outer slant gutters		
1	10	Downstream	40	120	410	0.026	5	5	No	Eight flares ignited
2	10	Upstream	40	↓	410	.045	5	5	No	Five flares ignited
3	10	Downstream	40	↓	380	.045	5	5	Yes	All flares ignited, 23 sec lag
4	8	Upstream	30	↓	400	.043	-	8	↓	Eight flares ignited
5	8	↓	30	↓	410	.032	-	8	↓	Eight flares ignited
6	8	↓	30	↓	410	.030	-	8	↓	Four flares ignited
7	4	↓	30	↓	405	.042	-	4	No	All flares ignited
8	8	↓	30	80	410	.045	-	8	↓	Six ignited
9	8	↓	30	110	↓	.030	-	8	↓	All flares ignited
10	8	↓	30	110	↓	.041	-	8	↓	All flames ignited
11	8	↓	2	120	↓	.042	-	8	Yes	All flares ignited, 3 sec lag
12	8	↓	2	↓	↓	.039	-	8	No	All flares ignited
13	8	↓	30	↓	↓	.030	-	8	No	Seven flares ignited

TABLE III. - FLARE COMPOSITION

Chemical composition of flares, percent by weight

Type	Magnesium	Boron	Potassium perchlorate	Sodium nitrate	Laminac
40	--	16	79	--	5
30	16	--	79	--	5
2	35	--	30	30	5

[redacted]

TABLE IV. - SUMMARY OF ENGINE STARTING WITH SPECIAL FUELS

Starting attempt	Type of special fuel	Starting condition			Rate of ignition fuel injection, lb/hr	Number of injection points	Specific gravity of fuel	Engine ignited	Total amount of fuel injected, lb
		Air flow, lb/sec	Inlet temperature, $^{\circ}F$	Fuel-air ratio, f/a					
1	Boron triethyl	120	410	0.045	900	(a)	0.69	No	3.2
2	Boron triethyl				2200	8	.69	No	5.5
3	Aluminum trimethyl				2600		.85	Yes	6.1
4					2600				3.7
5					3000	4			1.5
6				.041	2800				
7				.041	2800				
8				.030	3000				
9				.042	2000	2			
10				.042	2400	4			---
11		80	530	.045	2700	4			---
12		120	410	.030	900	2		No	2.7
13		120	410	.042	1000	4		Yes	2.0
14		110	530	.042	1970		1.3	No	4.0
15			530	.040	4250			Yes	4.2
16			410	.041	3900			Yes	3.8
17				.037	3400			No	5.0
18		120		.042	3100		.85	Yes	3.5
19		110		.042	3200		.85	Yes	1.0
20		120		.030	1310		1.3	No	5.3
21				.039	5700			Yes	3.2
22					1450			No	4.5
23					6080			Yes	5.0
24					----		.85	Yes	.7
25				.042	1310		1.3	No	5.5

[a]Injected through main fuel mainfold.

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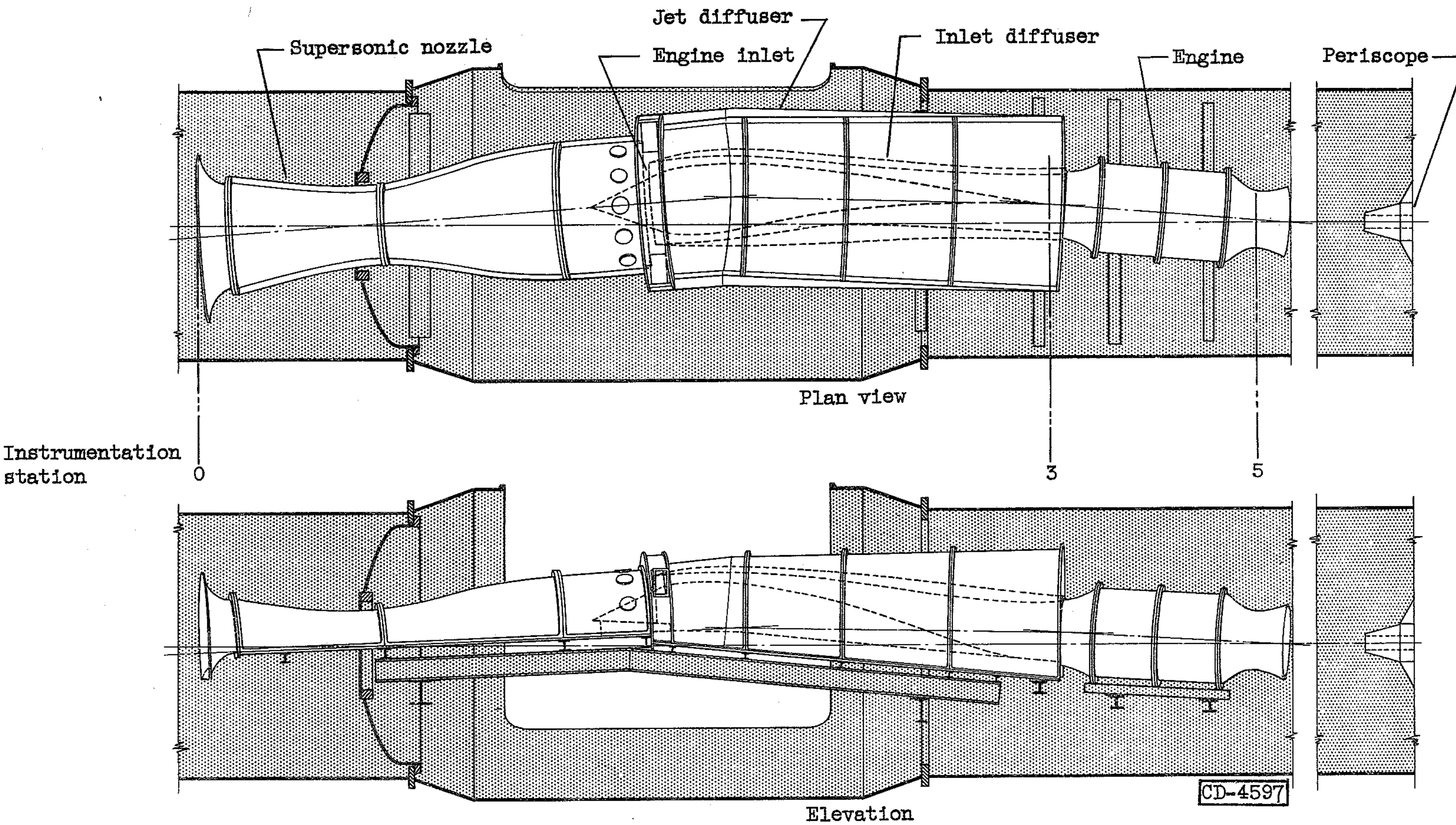

Instrument station	Total-pressure tubes	Static-pressure tubes	Thermocouples	Wall static-pressure orifices
0	17	0	14	0
3	48	21	21	6
5	32	0	0	4

Figure 1. - Free-jet installation of 48-inch ram-jet engine.

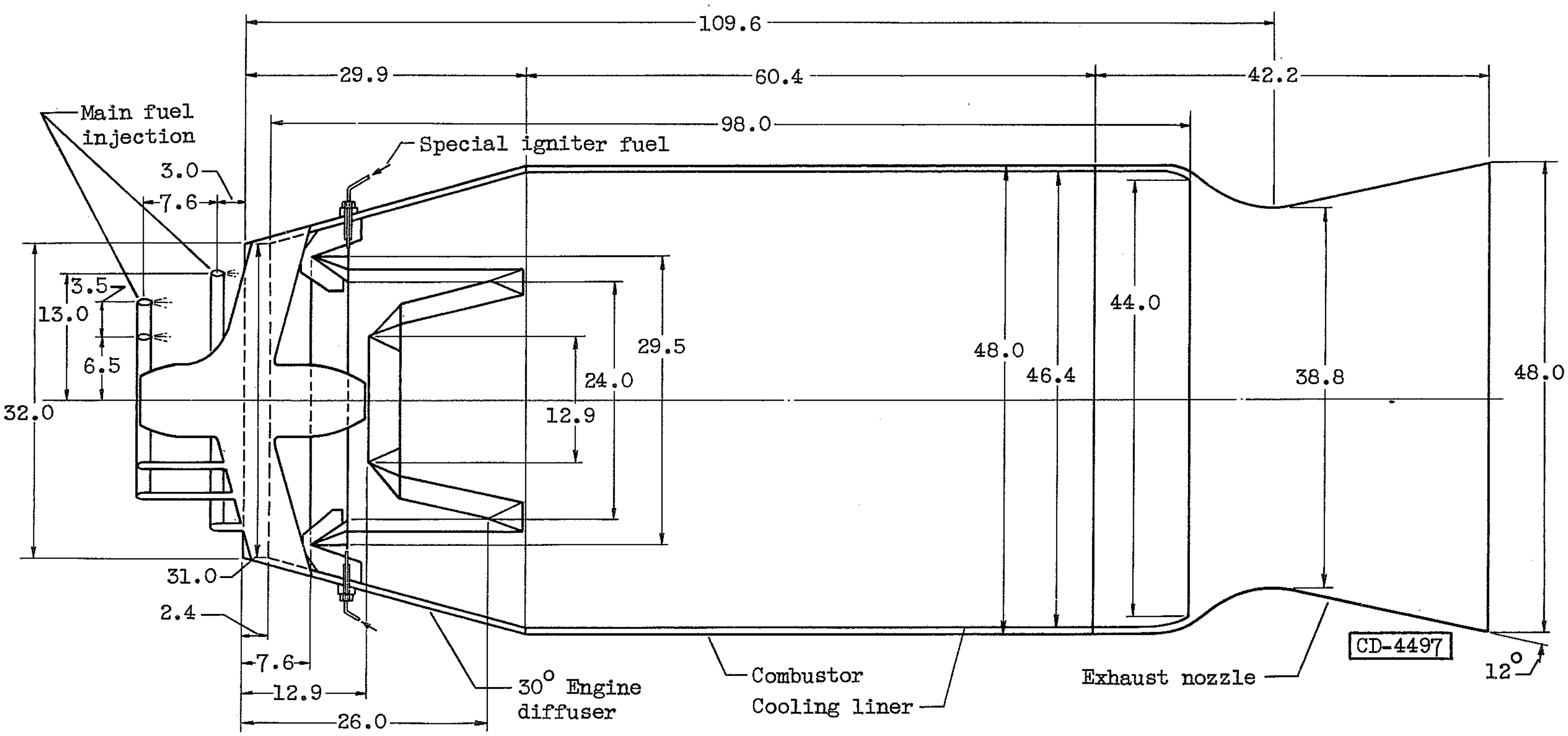

Figure 2. - Cross section of XRJ47-W-5 ram-jet flight engine. (All dimensions in inches.)

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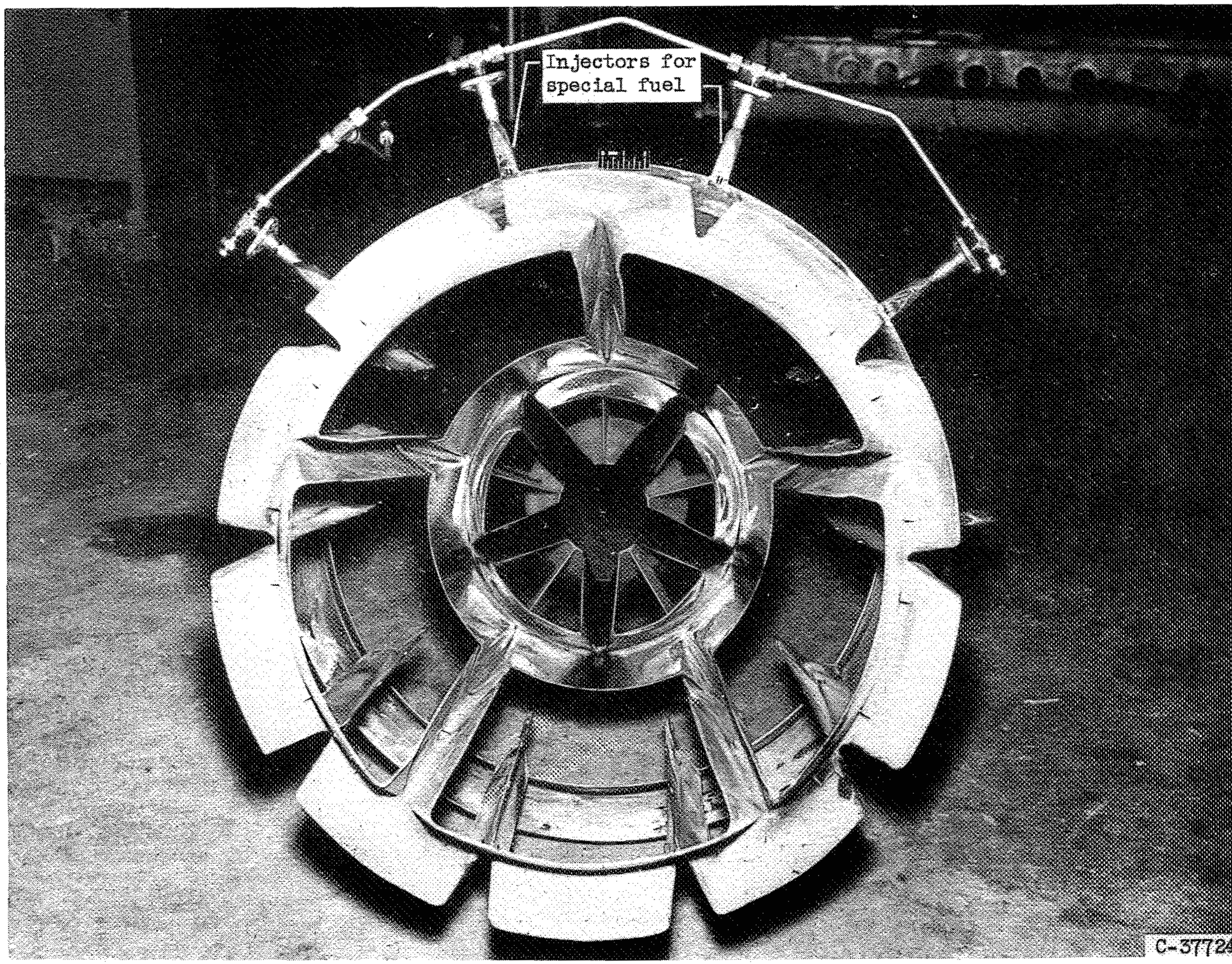

Figure 3. - Special fuel-injection manifold with four injection points.

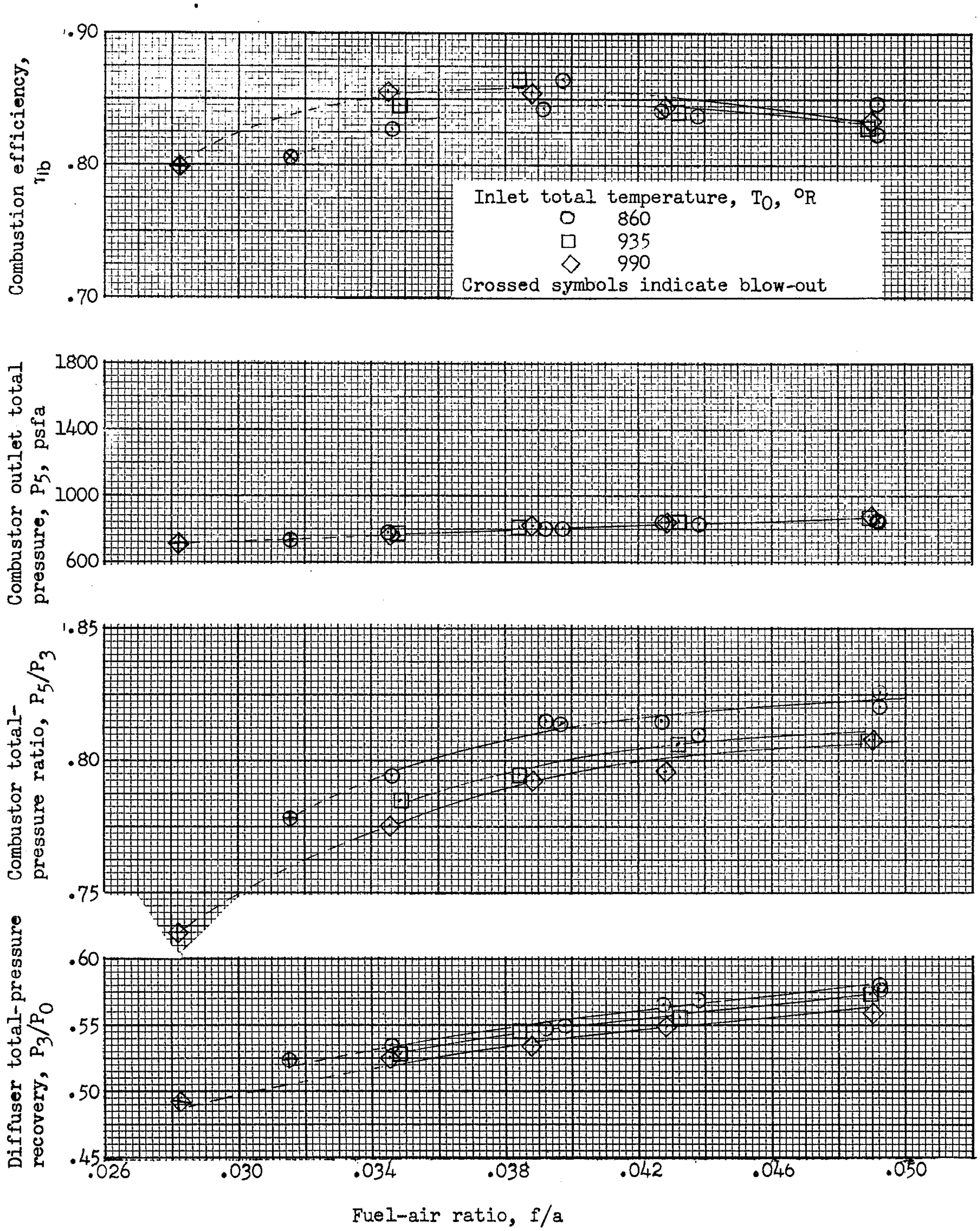

Figure 4. Engine performance. Altitude, 73,000 feet ; inlet air flow, 60 pounds per second.

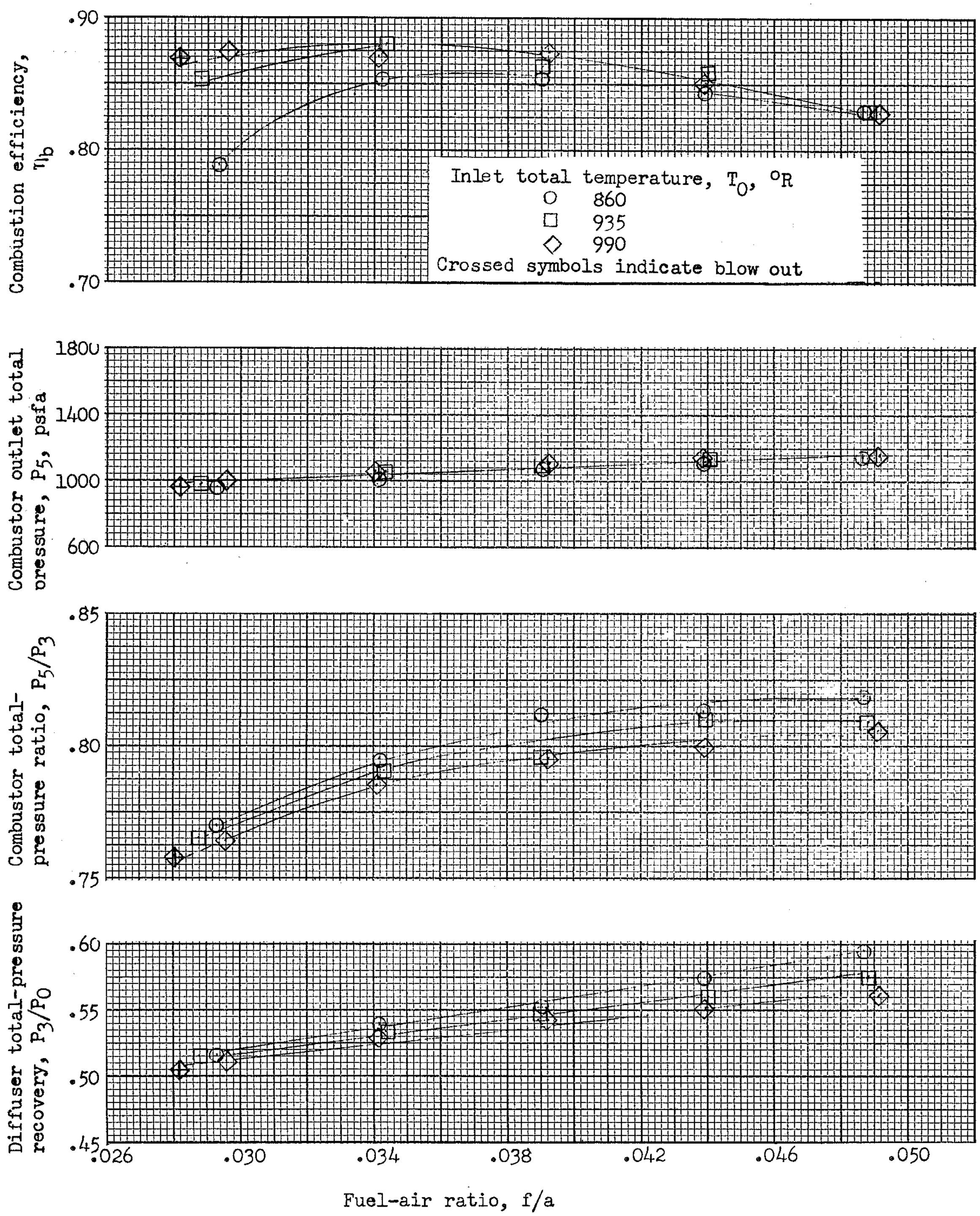

Figure 5. Engine performance. Altitude, 67,000 feet; inlet air flow, 80 pounds per second.

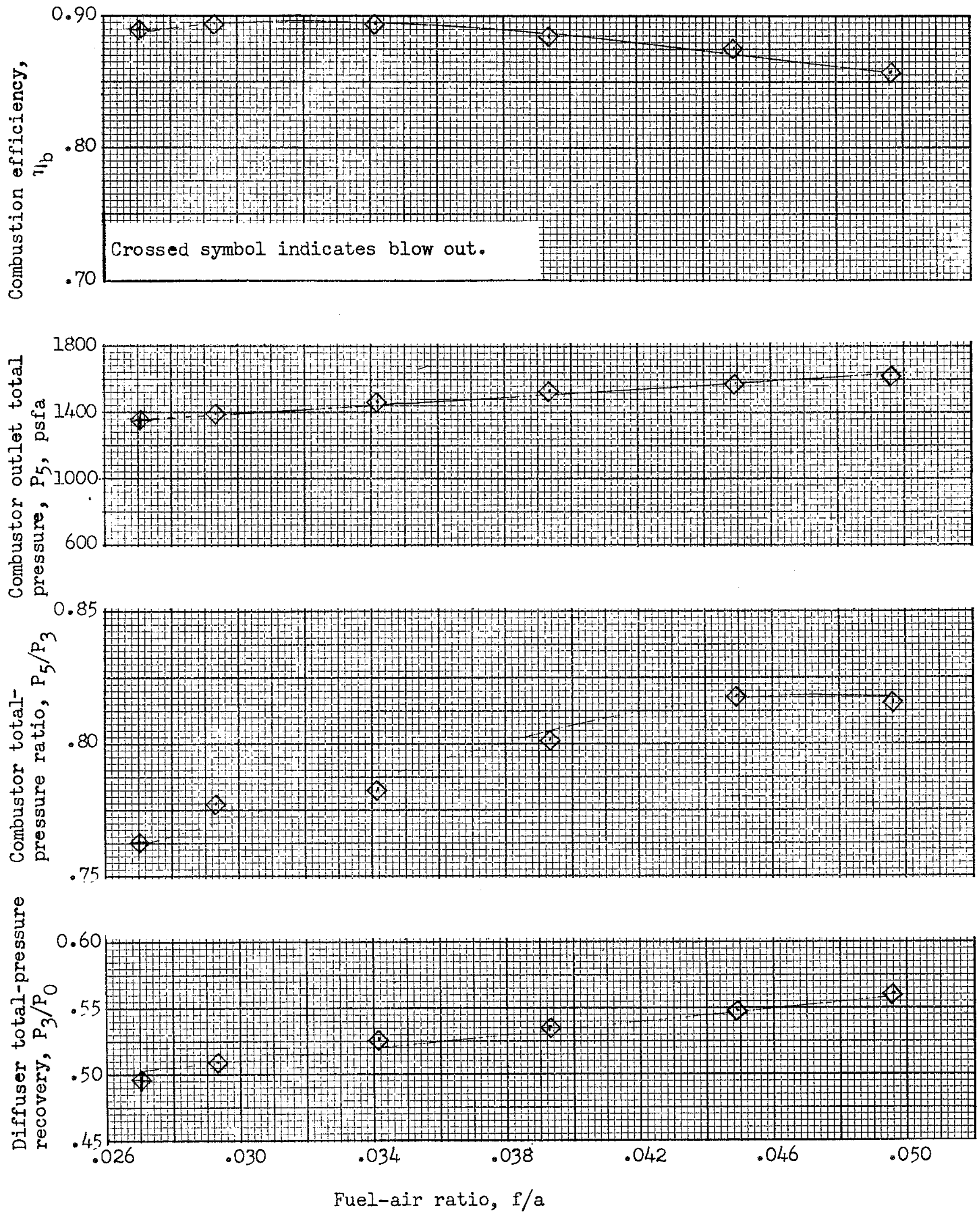

Figure 6. Engine performance. Altitude, 60,000 feet; inlet air flow, 110 pounds per second; inlet temperature, 990°R.

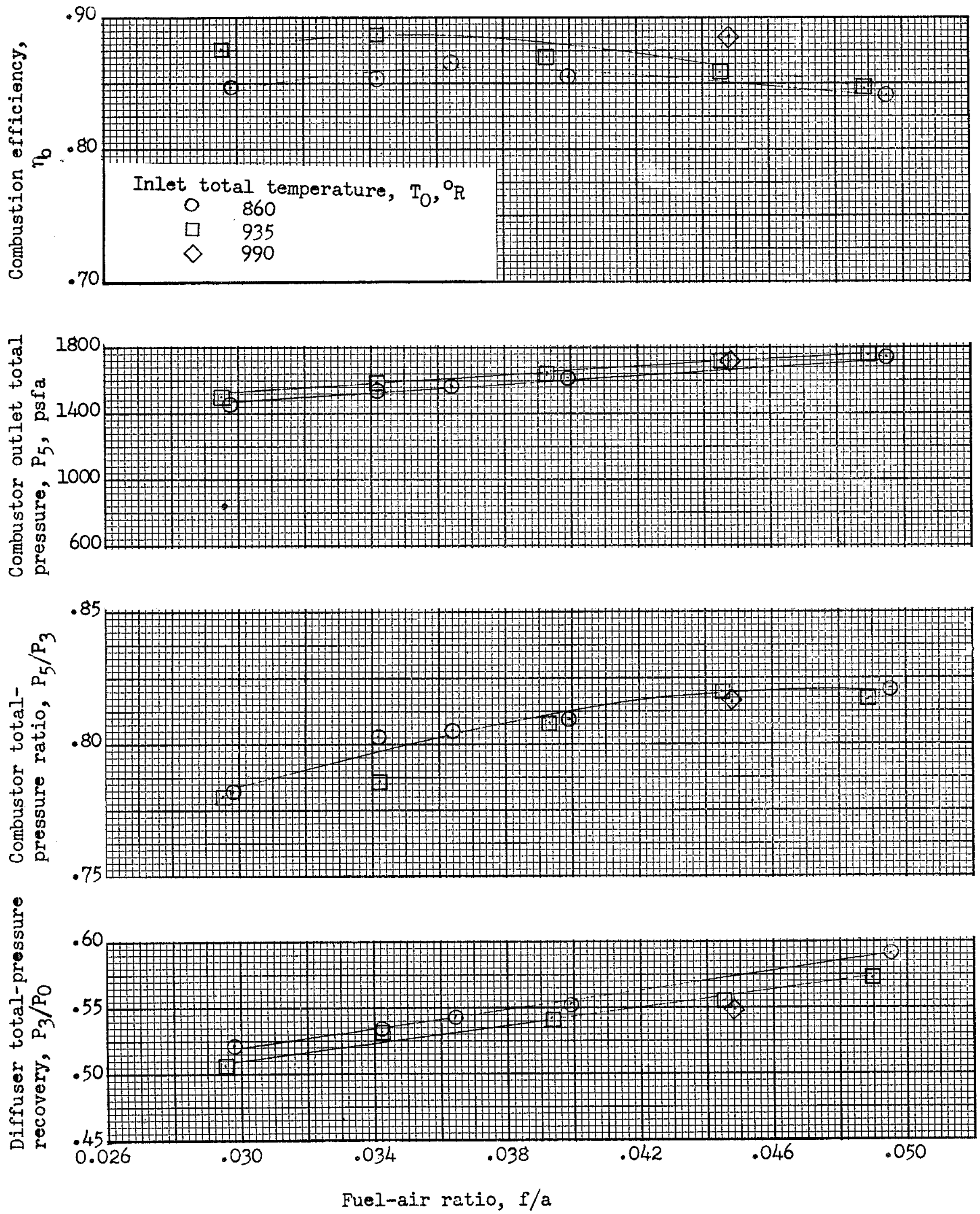

Figure 7. Engine performance. Altitude, 58,000 feet; inlet air flow, 120 pounds per second.

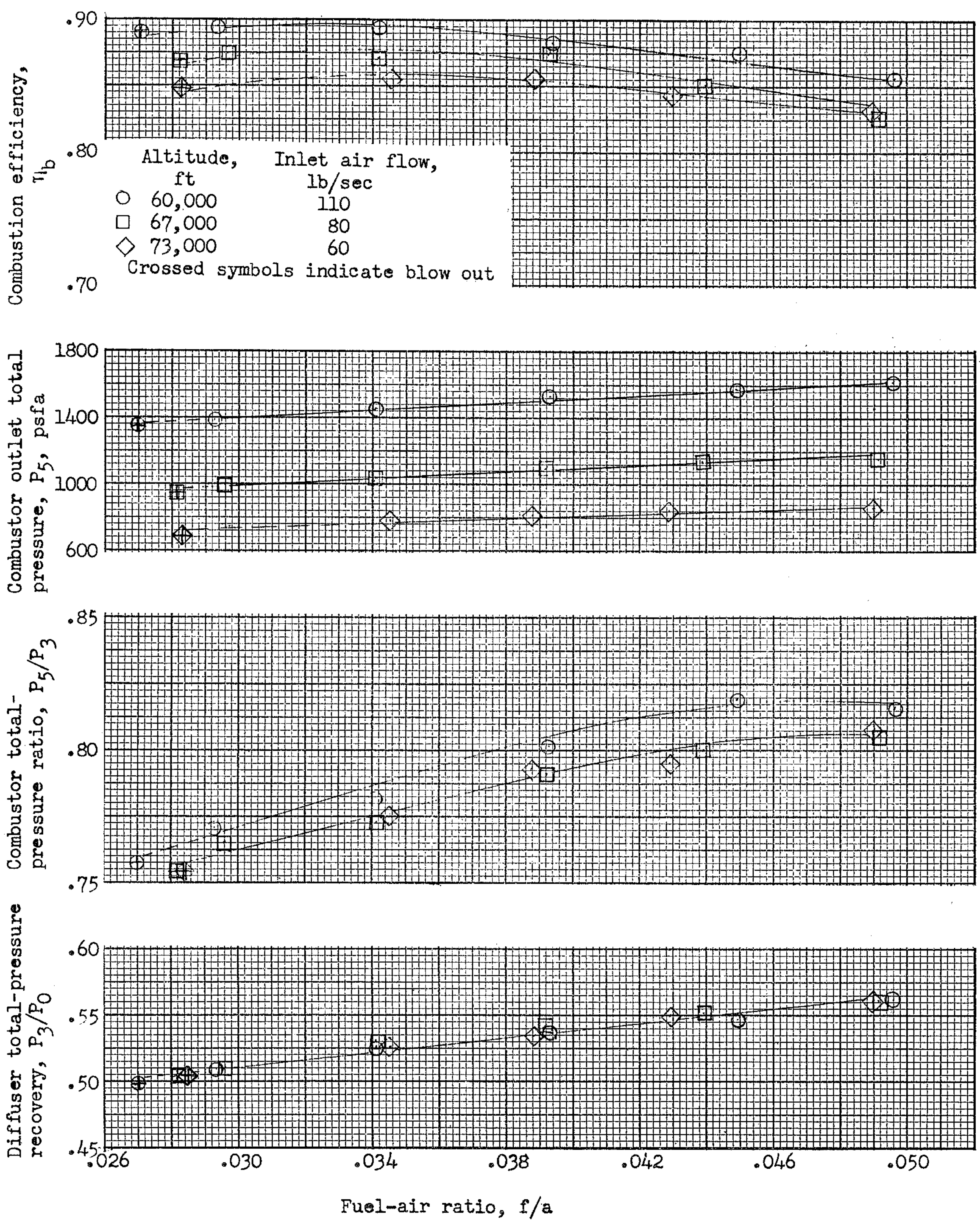

Figure 8. Engine performance. Inlet temperature, 990°R.

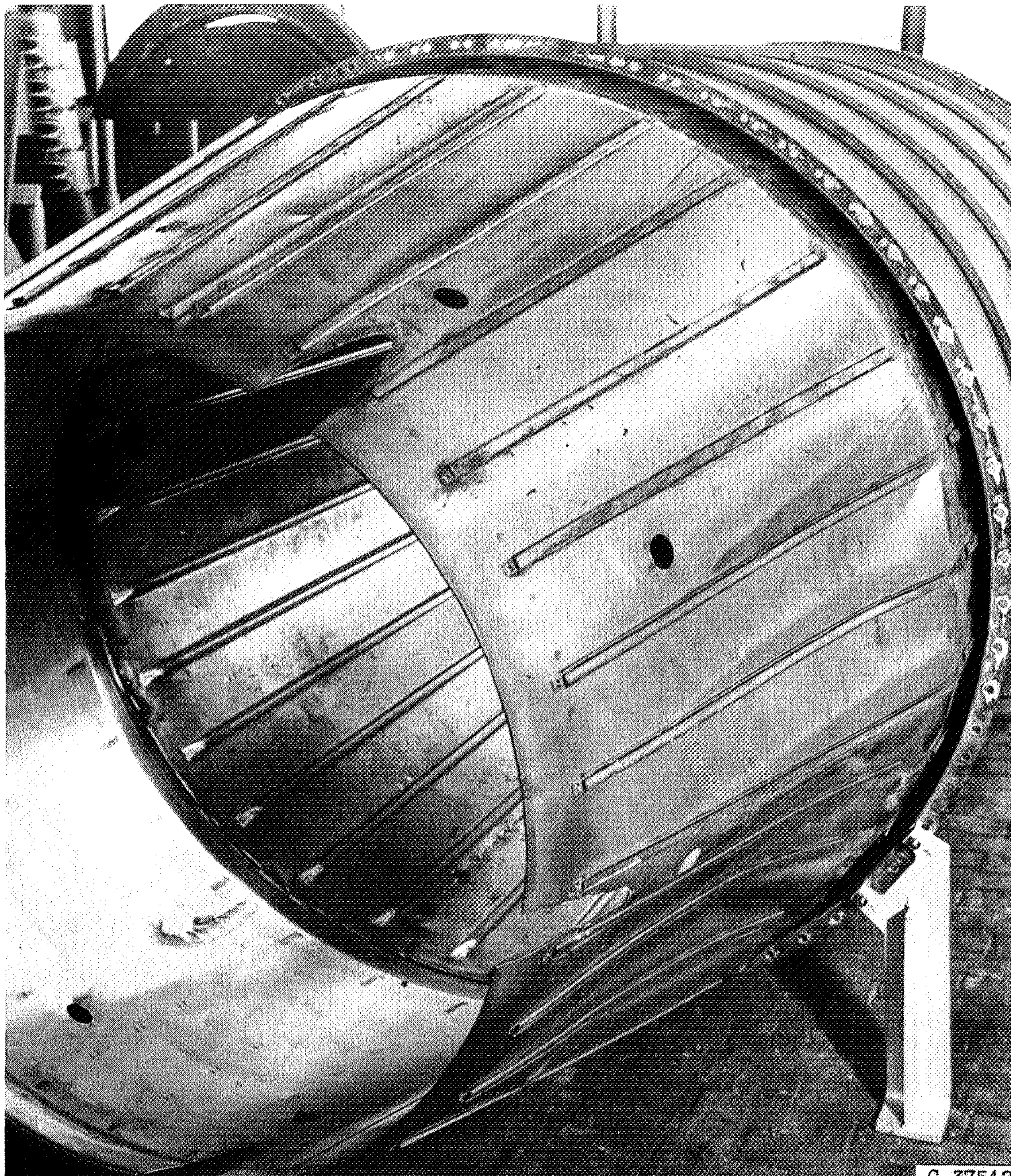

Figure 9. - Damage to 30° conical transition section after 2 hours of cold-flow operation.

[redacted]

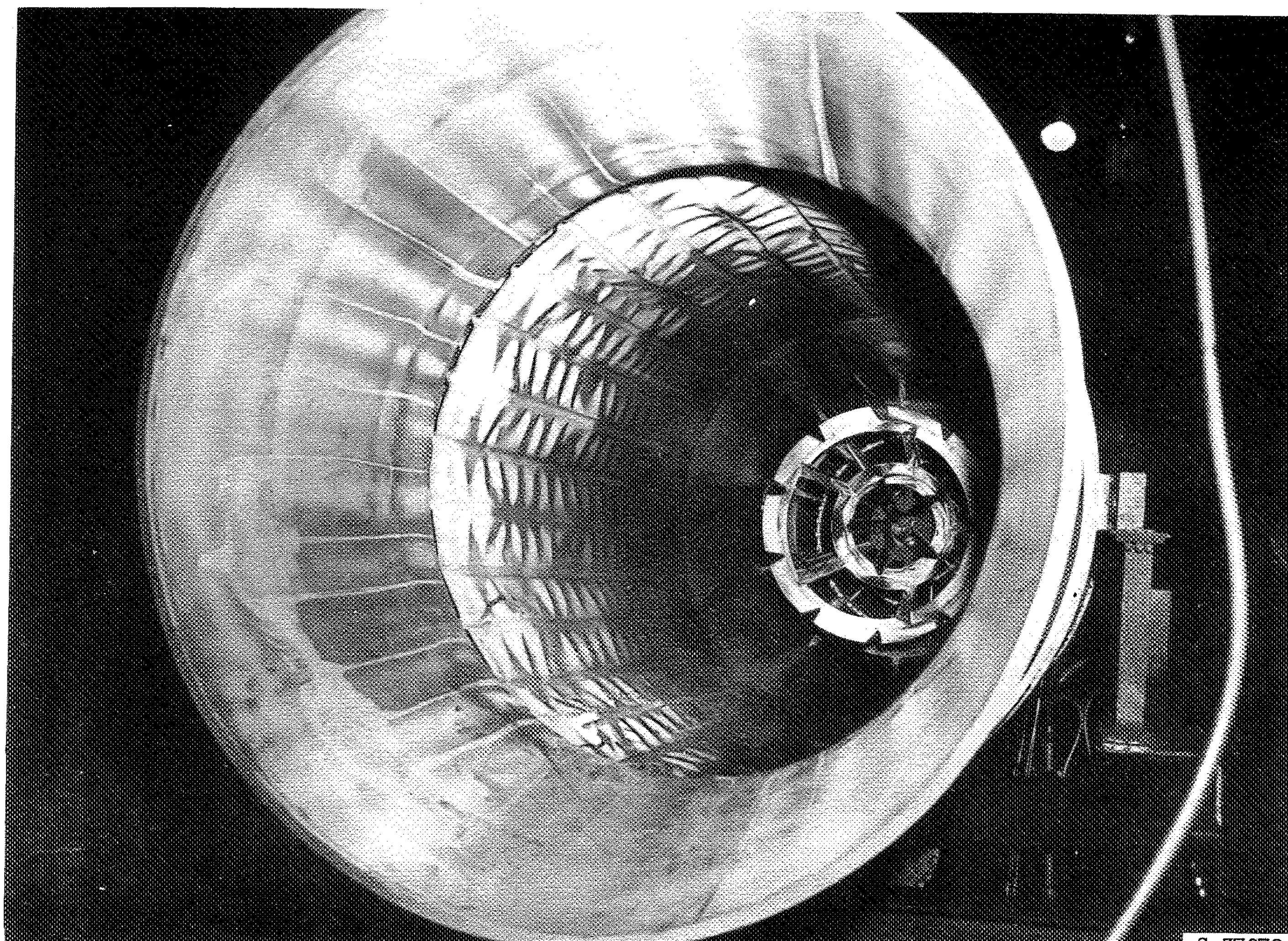

Figure 10. - Damage to original liner after 1-hour of burning operation at altitude of 73,000 feet.

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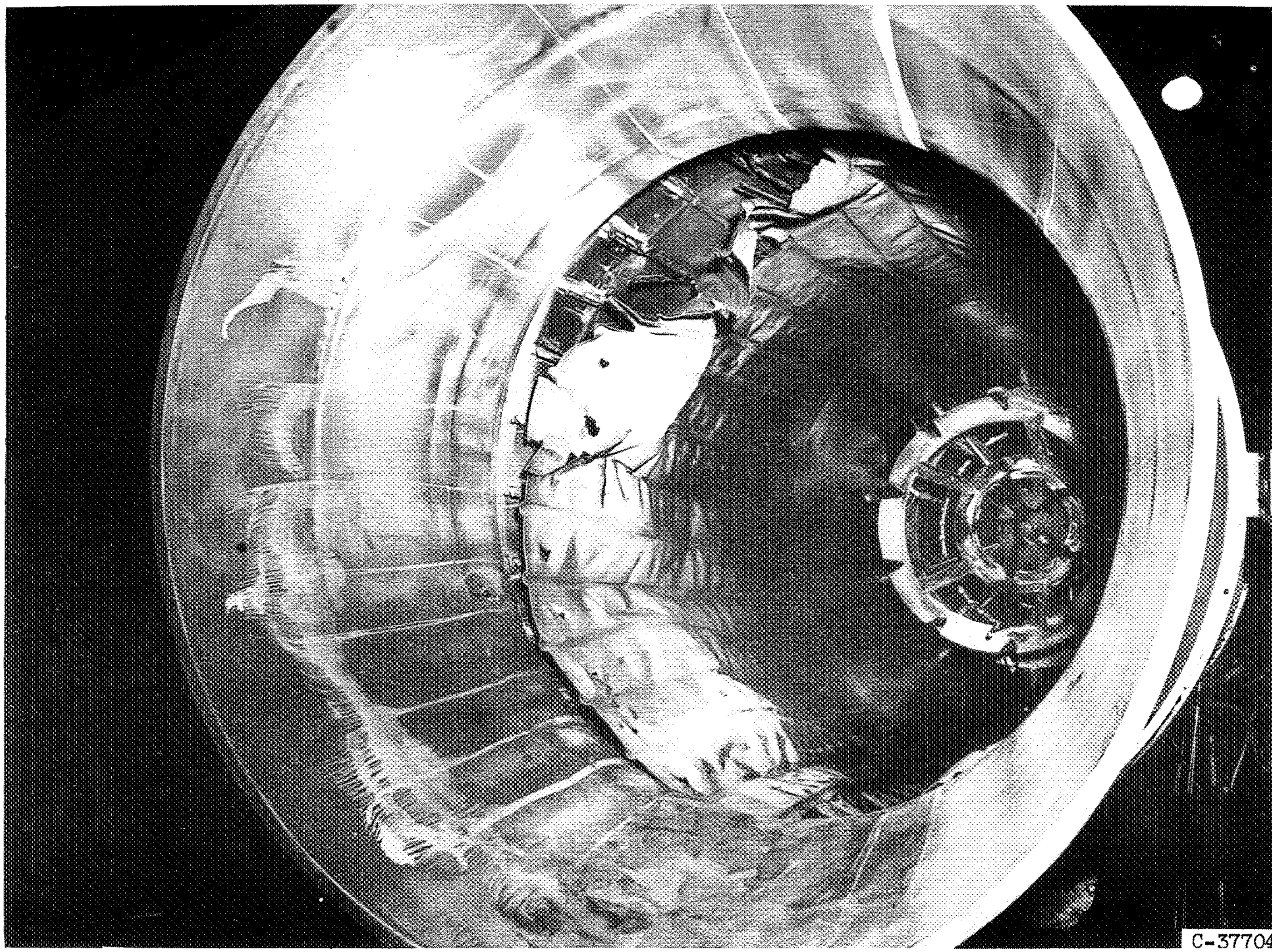
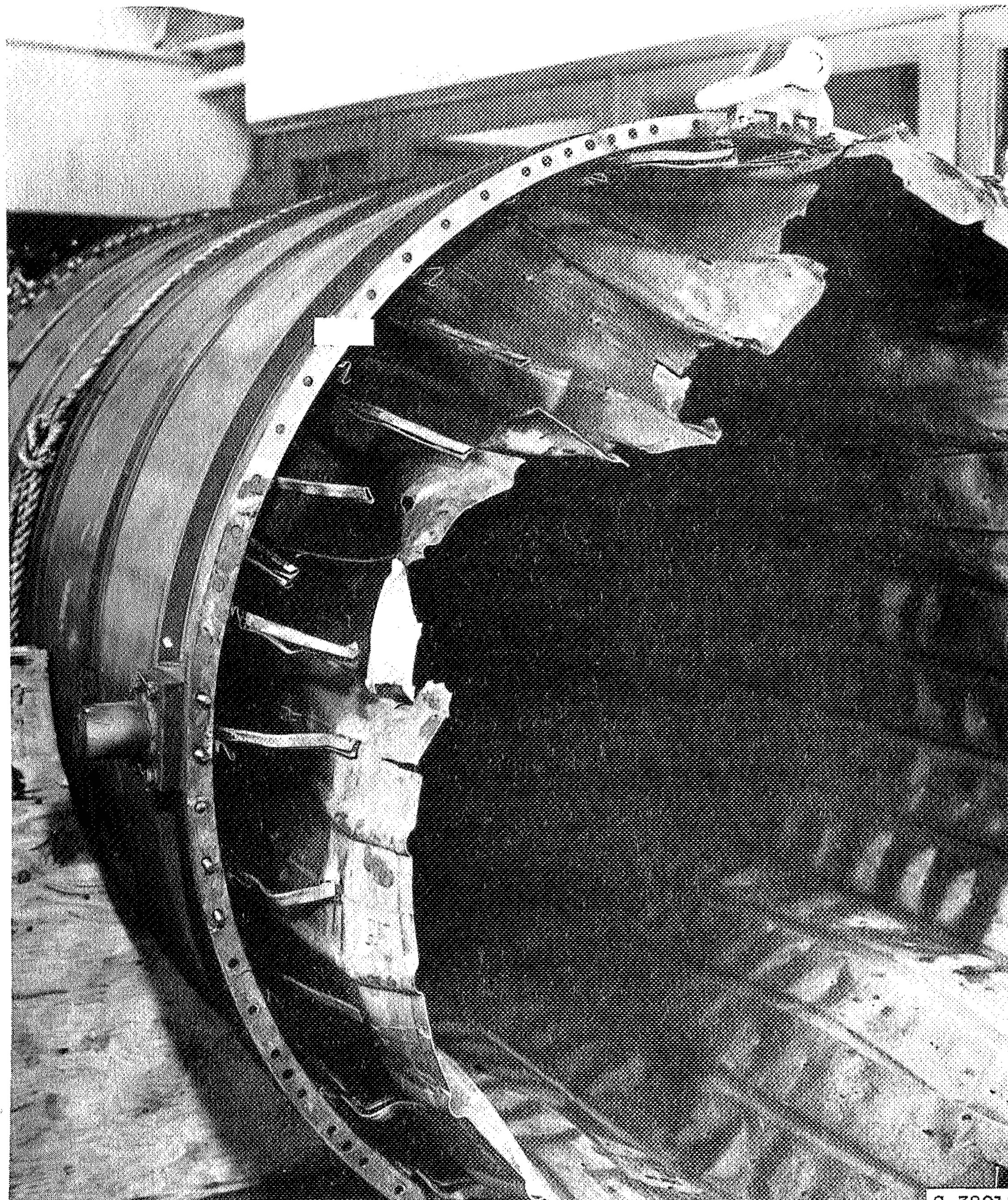


(a) Complete engine.

Figure 11. - Failure of original liner after a total burning time of 1 hour and 33 minutes.



(b) Exhaust nozzle removed.

Figure 11. - Concluded. Failure of original liner after a total burning time of 1 hour and 33 mintues.

Figure 12. - Modification of second cooling liner.

[redacted]

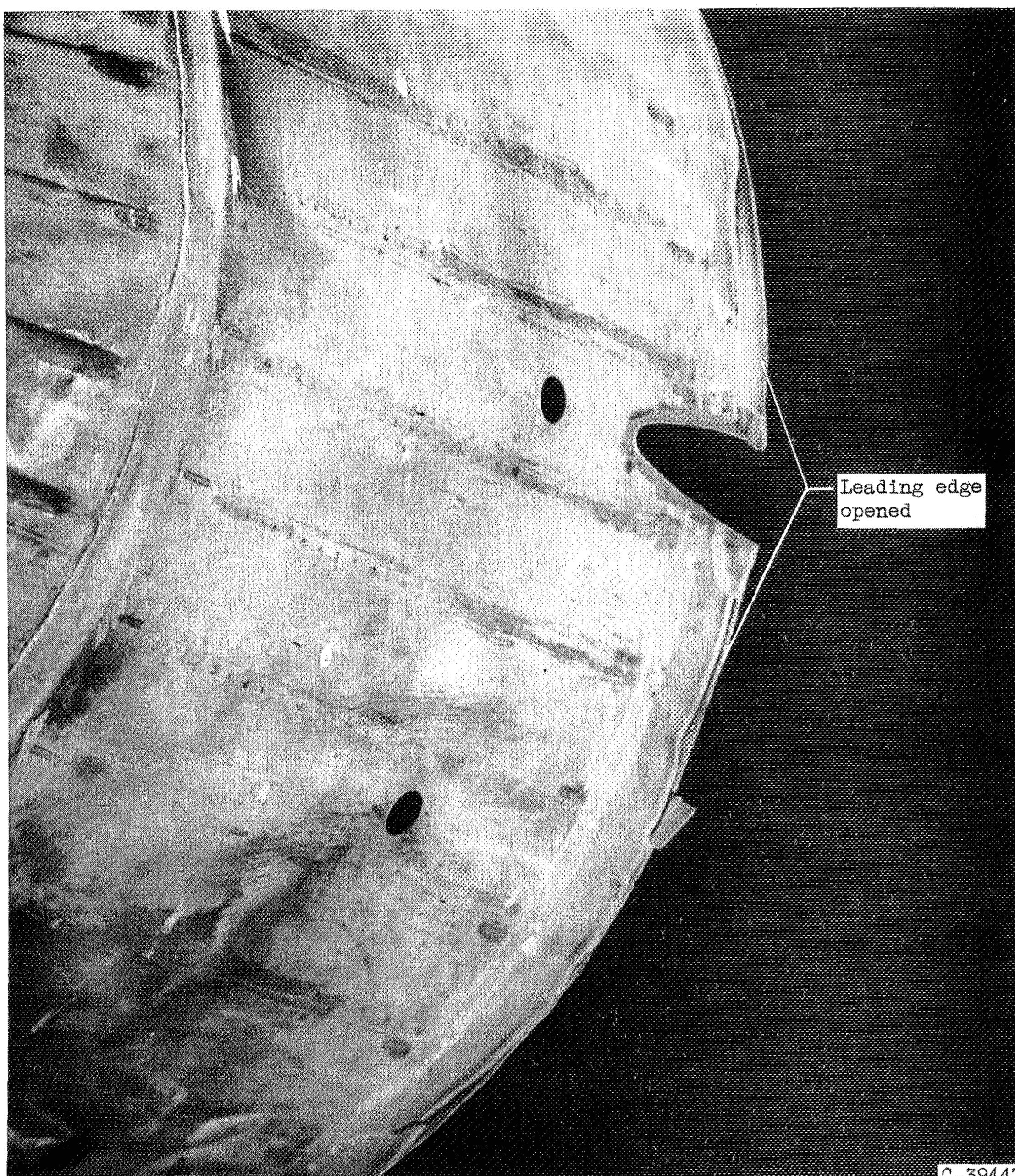


Figure 13. - Leading-edge damage of second cooling liner after a total burning time of 22 minutes.

[redacted]

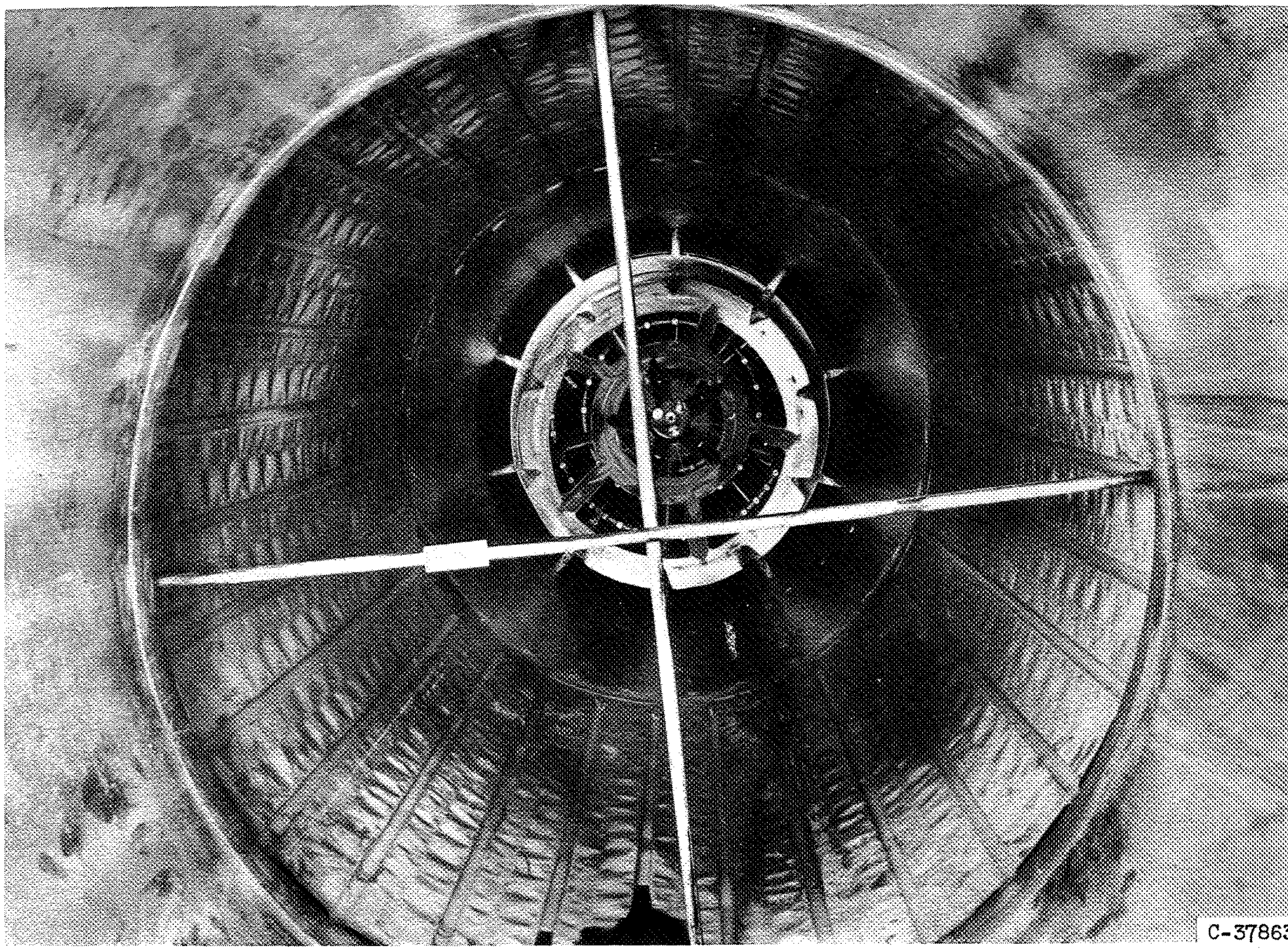


Figure 14. - Condition of second cooling liner after a total burning time of 2 hours and 22 minutes.

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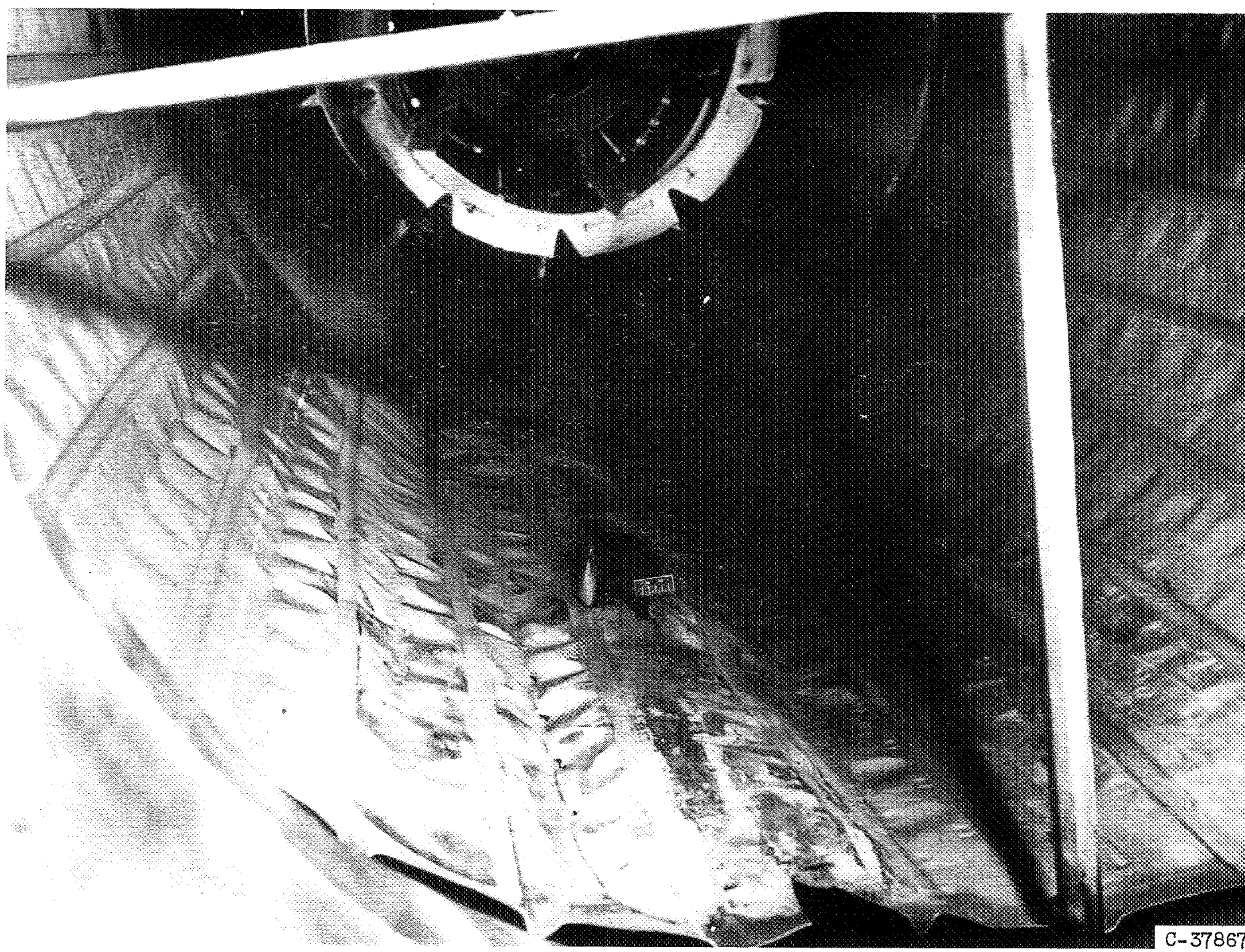
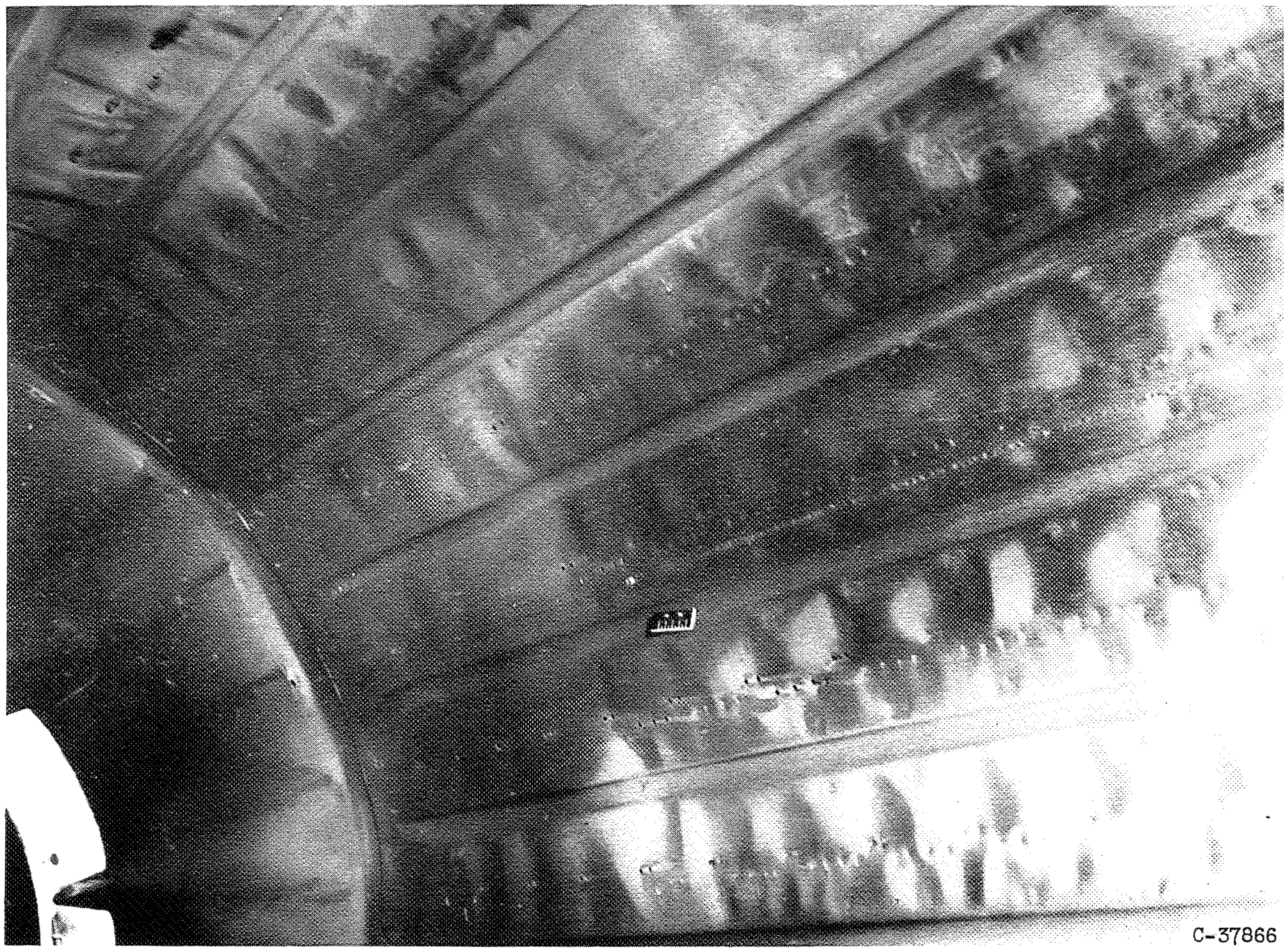


(a) Hole burned in liner.

Figure 15. - Damage to second cooling liner after a total burning time of about $3\frac{1}{2}$ hours.

[redacted]

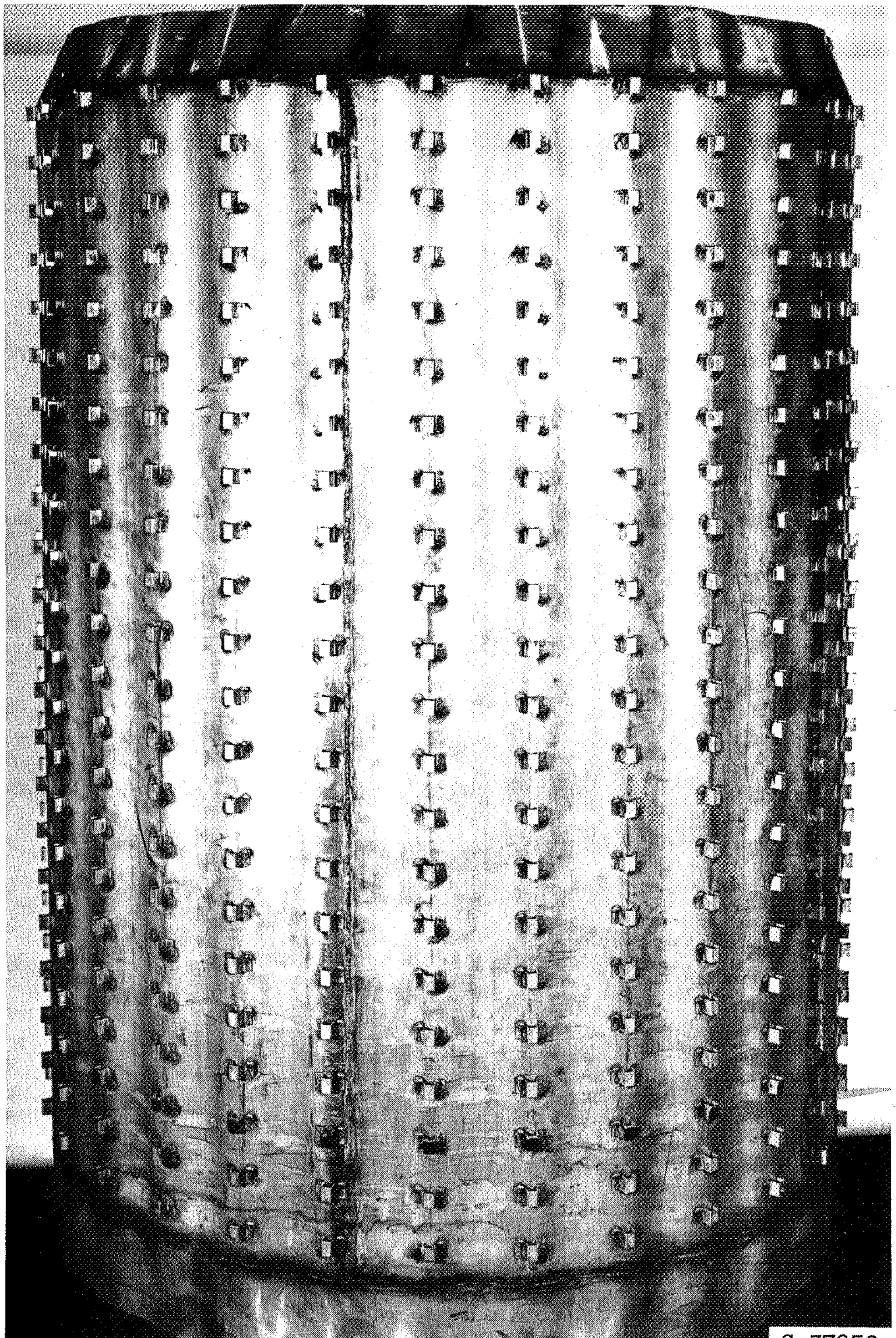


(b) Spot-weld failure.

Figure 15. - Concluded. Damage to second cooling liner after a total burning time of $3\frac{1}{2}$ hours.

[redacted]

Figure 16. - Failure of third cooling liner after a total burning time of 1 hour and 43 minutes.



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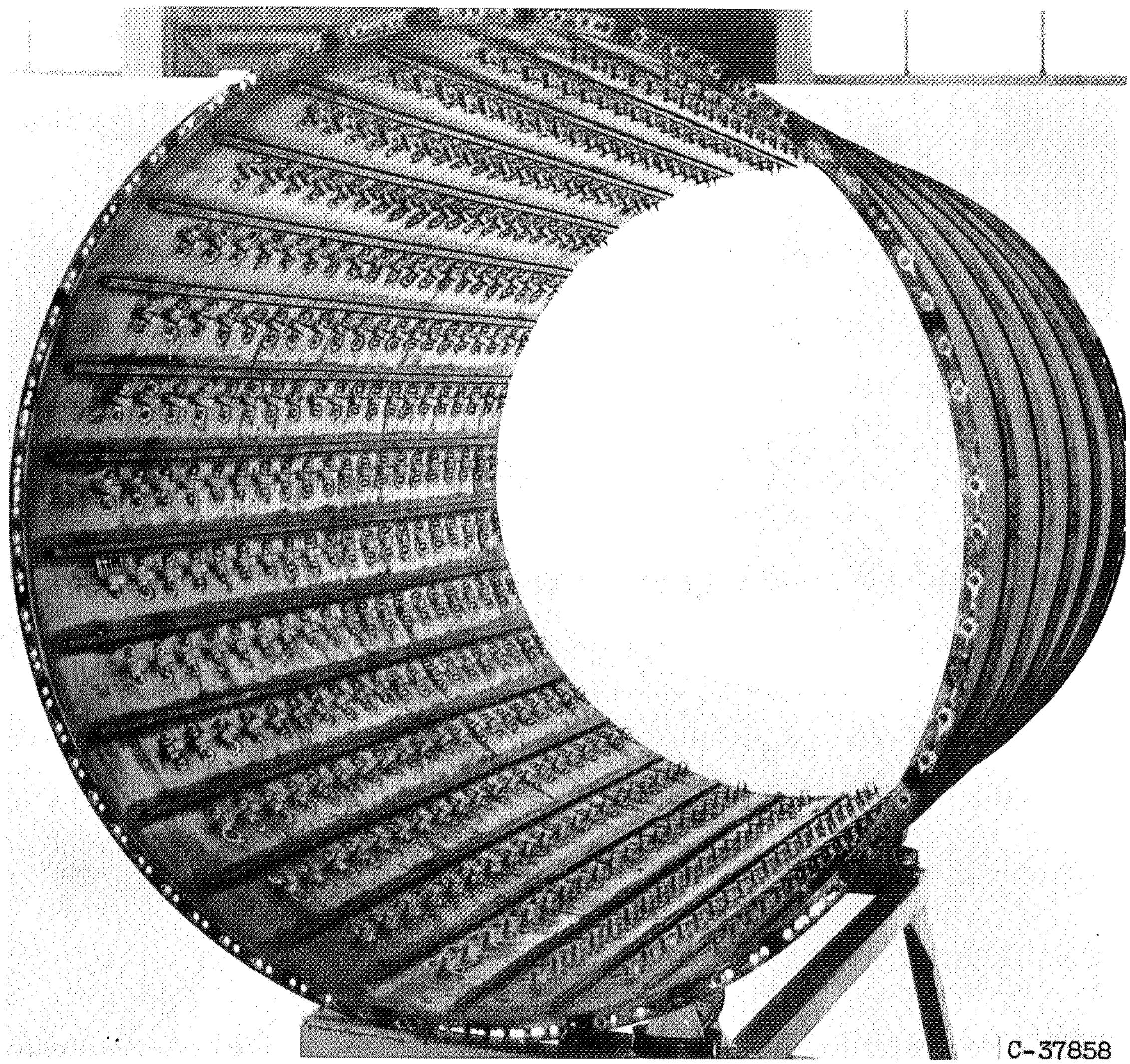
(a) Cooling liner.

Figure 17. - NACA cooling-liner design.



(b) Engine shell.

Figure 17. - Concluded. NACA cooling-liner design.

[redacted]

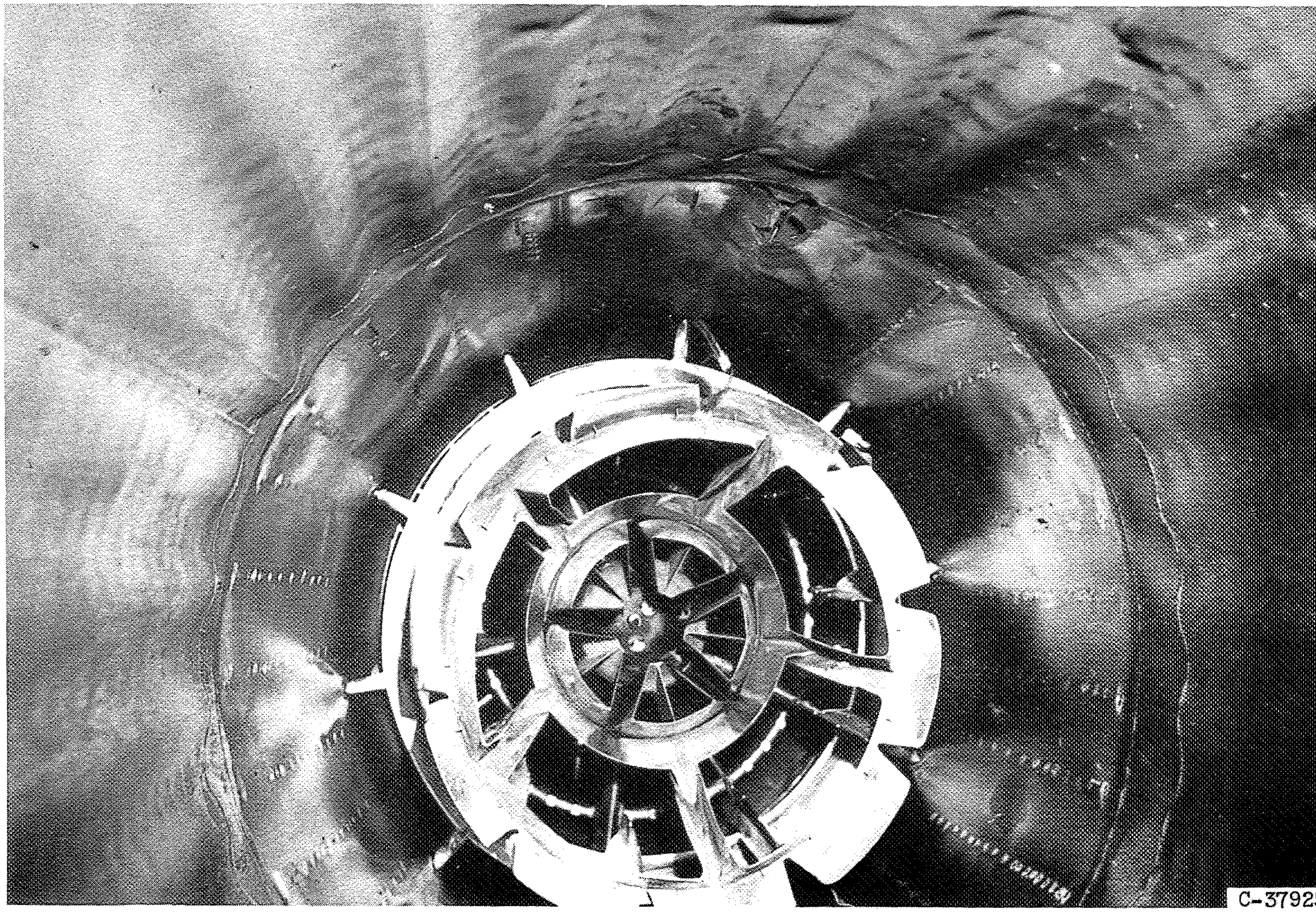


Figure 18. - Damage to 30° conical transition section with NACA liner installation.

 [redacted]

PRELIMINARY EVALUATION OF FLIGHT-WEIGHT XRJ47-W-5 RAM-JET ENGINE AT A MACH NUMBER OF 2.75

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maa - 7/26/55

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