Rocket

Prelim. data, rocket engine 123.33(E229)
X 422

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DECLASSIFIED
~~803910~~ Date 10/3/84

JUL 13 1949

Cleveland, Ohio.
JUL 11 1949

From Lewis
To NACA Headquarters

Subject: R.A. E-229, Altitude Ignition of Liquid Propellant Rocket Engines.

1. The laboratory has recently been conducting an accelerated program on the altitude ignition characteristics of the Naval Model CML5N rocket engine under research authorization E-229, requested by the Bureau of Aeronautics, Navy Department. Phase I of this accelerated program has been concerned with the nature of the ignition difficulties encountered at high altitude and is now complete. Enclosed are six copies of a memorandum covering the results obtained in Phase I. Extra copies are for transmittal to Lt. Commander R. C. Truax, Bureau of Aeronautics, Navy Department.

2. Phase II of the accelerated program, namely research on the cures for altitude ignition difficulties, is now starting and a conference with personnel from the Bureau of Aeronautics is desired to determine the urgency of this work and to discuss in more detail the conditions at which the rocket engine will be required to start.

3. When the work is completed a regular report will be prepared by the laboratory.

Edward R. Sharp
Edward R. Sharp,
Director.

WTO:ytb
HP

Enclosures:
1. Six copies of memorandum regarding Navy Model CML5N rocket engine.

4 copies ... sent to Mr. TRUAX ... July 14, 1949 ...

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... (7-14-49)

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Cleveland, Ohio,
July 11, 1949.

MEMORANDUM For Director

Subject: R.A. E-229, Altitude Ignition of Liquid Propellant Rocket Engines.

Reference: (a) Memorandum from Lewis to NACA, WTO:ytb, BP, June 24, 1949, A84055.

1. Summary: The ignition characteristics of a 220-pound thrust cylinder from the Navy Model CML5N rocket engine operating on monoethylaniline and mixed acid were determined at sea level pressure and ambient temperature, at pressures corresponding to 55,000 feet altitude and sea level ambient temperatures, at sea level pressure and low temperatures, and at pressures corresponding to 55,000 feet and low temperatures. Satisfactory ignition occurred at all pressures and at all temperatures above -28° F. Below -35° F and at pressures corresponding to 55,000 feet, the rocket failed to ignite; in one instance an explosion occurred. The principal cause of the ignition difficulties appeared to be the high viscosity of the hydraulic fluid that operates the propellant valve and the high viscosity of the monoethylaniline as a result of the low temperature.

2. Introduction: On June 14, 1949 the Lewis Laboratory was asked by the NACA Headquarters office to consider conducting research to determine any ignition difficulties encountered at high altitude with the Reaction Motors, Incorporated, Navy Model CML5N rocket engine (monoethylaniline and mixed acid) and to report on the results within 30 days from the initiation of the work. The specific information was desired by the Bureau of Aeronautics, Navy Department, who stated the intention of using two CML5N engines, originally designed for the Lark Missile, as auxiliary powerplants on a fighter type aircraft that is being prepared for flight within about 30 days. This aircraft, according to the Navy, might be expected to cruise at 45,000 feet for periods up to one or two hours before requiring the auxiliary rocket engines. Several starts and stops of the rocket engines might be required at altitudes up to 45,000 or 55,000 feet. It was desired to know whether or not difficulties in ignition, such as explosions, would be encountered, the nature and possible causes of any difficulties, and suggested cures or "fixes" for the difficulties. On June 15 it was determined that the project could be accomplished and work was started.

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On June 21 Lt. Commander R. C. Truax of the Bureau of Aeronautics visited the Lewis Laboratory in the company of Mr. Henry Alquist of the NACA Headquarters office and participated with members of the staff in establishing the outline of the research on the CML5N engine. (See reference a). It was determined that the work logically fell into two phases. Phase 1 would be concerned with determining the ignition difficulties encountered and Phase 2 would be concerned with the cures or fixes for the difficulties. It was agreed that Phase 1 could reasonably be completed within 30 days, but that Phase 2 would take an indeterminate time because it was not possible to predict what difficulties might be encountered. Phase 1 has now been completed, and it is the purpose of this memorandum to outline the results obtained and the conclusions and implications based on these results.

3. Outline of Research (Phase 1): The research was limited at the Navy's request to the 220-pound thrust cylinder of the CML5N rocket engine; this engine also has a 400-pound thrust cylinder. A series of ignition attempts were to be made at the successive conditions that were believed to cover the conditions likely to be encountered in the flight operation:

Pressure	Temperature
a) Sea level	Ambient
b) 55,000 feet	Sea level ambient
c) Sea level	Low, but not to freeze propellants
d) 55,000 feet	Low, but not to freeze propellants

The engine and propellants were to be operated in the "as received" condition with the engine at a horizontal attitude to simulate closely the flight condition. Monoethylaniline and mixed acid were to be furnished by the Navy to insure conformity to materials expected to be used in flight. Propellant tank pressures were to be normal for this engine at approximately 450 pounds per square inch gage. Observations planned included, in addition to the usual measurements of ambient and initial conditions, time histories of the propellant valve position, the propellant injection pressure, rocket combustion chamber pressure, and rocket exhaust temperature.

4. Apparatus: The apparatus consisted of an altitude tank and its auxiliaries with the rocket engine attached to exhaust into the tank. The rocket engine, propellant valves, and propellant tanks were mounted in a low-temperature bath. Figure 1 shows photographs of the general setup.

5. Figure 2 shows a layout of the altitude tank and auxiliary equipment. The tank was 8 feet in diameter and 29 feet in length. It was provided with two 20-inch flanges at one end, a 10-inch flange

at the opposite end, a 6-inch connection to the vacuum pump, and a water inlet and outlet. The rocket engine was mounted on one 20-inch flange and the other 20-inch flange contained a blowout disk. A gate valve was mounted on the 10-inch flange at the opposite end of the tank as an exhaust outlet. The vacuum pump could be isolated from the tank by means of a 6-inch vacuum valve. A tee was placed in the line from the tank to the vacuum pump and a 6-inch gate valve was provided as a vacuum break. A centrifugal blower was connected by a 6-inch line to the vacuum break valve and was used to ventilate the tank after a rocket operation. The floor of the tank contained water to quench and to render harmless any unignited propellants.

6. To provide the low temperatures for the engine assembly and the propellants, the entire engine assembly and the propellant tanks were immersed in a low-temperature bath containing a mixture of chloroform and carbon tetrachloride (noninflammable; low-freezing point). This mixture was circulated through coils in a dry ice-methyl cellosolve bath until the desired low temperatures were reached by the equipment and propellants. Alternatively, the cooling mixture could be drained from the unit, or could be left in the low-temperature bath. For some of the runs the chloroform-carbon tetrachloride mixture was cooled in the low temperature bath by the addition of small chunks of dry ice to the bath.

7. A diagrammatic sketch of the Navy Model CML5N rocket engine (220-pound thrust cylinder) and propellant valve assembly and the flow system used with it is shown by figure 3. A regulated supply of helium was used to force the propellants into the rocket engine and to force the hydraulic fluid into the propellant valve. The pressurization of the propellants tanks and the accumulator of the propellant valve was accomplished by means of helium-operated three-way valves which were controlled by electric valves. When the electric valves were de-energized the propellant tanks and the accumulator were vented to the atmosphere. The propellant tanks were made of Inconel and had a capacity of about 1.3 gallons. The filler caps of the tanks contained safety blowout disks. Taps were provided on the tanks for the measurement of pressure and temperature and to drain the propellants. Pressure taps on each propellant line to the injector and a pressure tube to the combustion chamber were on the engine when received.

8. The injection pressure of the fuel and of the oxidant were measured by Bourdon-tube type pressure recorders located approximately seven feet from the engine. The combustion chamber pressure was measured by three recorders. Two differential pressure gages were modified to serve as pressure cells for sub-atmospheric pressures. They were mechanically linked to a resistance-type position transmitter and their outputs were recorded on self-balancing, recording potentiometers. Combustion pressures from atmospheric to the normal engine operating pressure of 300 to 350 pounds per square inch were measured by a Bourdon-tube type pressure recorder. The combustion chamber pressure line was approximately 30 feet long. The injection and combustion chamber pressure lines were filled with liquids for part of the runs to minimize lag in the response of the pressure

recorders. The pressure in the propellant tanks and in the altitude tank were also measured by Bourdon-tube type recorders. The position of the propellant valve was measured by means of a resistance-type position transmitter the output of which was recorded on a self-balancing recording potentiometer. The transmitter was linked to the propellant valve by means of a gear and rack arrangement. The rack was mounted on an "I" shaped bar that was bolted to the yoke of the propellant valve. Propellant temperatures were measured by copper-constantan thermocouples installed in the propellant tanks. In addition, a copper-constantan thermocouple was placed on the surface of the engine jacket. A chromel-alumel thermocouple was installed at the rocket nozzle exit as a qualitative indication of ignition. The outputs of the copper-constantan thermocouples were measured by a manual-balance potentiometer. For most of the runs the output of the thermocouple in the rocket exhaust was recorded by a microammeter and for the remainder of the runs the output was recorded on a self-balancing recording potentiometer. All recorders except the altitude tank pressure recorder and the microammeter were time-synchronized by marker pens actuated from a common time signal.

9. Procedure: The quantities of propellants charged to the tanks were small enough so that if all of the propellants ignited and burned, the pressure in the altitude tanks would not exceed one atmosphere. No more than 4.5 pounds of monoethylaniline and 16 pounds of mixed acid were used for any run. Although the design oxidant to fuel weight ratio is from 3.15:1 to 3.45:1, an excess of acid was charged to insure that the cooling jacket on the combustion chamber would remain filled. The piping and the combustion chamber jacket were filled with the appropriate propellant down to the propellant valve by bleeding at the connection to the propellant valve for each propellant; this condition of filled lines would exist for all firings subsequent to the first one during the anticipated flight operation. Subsequent to charging, the instruments were started, the propellant tanks were pressurized, and the firing switch that pressurizes the propellant valve was closed. The propellant valve was permitted to open until operating combustion chamber pressure was obtained (about 300 lbs/sq in), then closed. If no ignition occurred, the propellants were completely ejected from the system and the subsequent flow of helium gas flushed out the combustion chamber before the injection valve was closed. All records that were time synchronized were plotted on a single time scale.

10. Propellants: The monoethylaniline used for the results reported here was furnished from the Naval Aviation Supply Depot, Philadelphia, Pennsylvania and was Stock Number K51-M-1105-ethylaniline 75-100-170. The lot is expected to be of approximately the following composition (Navy specification for Model CML5N engine):

Monoethylaniline	62-66 percent
Aniline	24-28 percent
Diethylaniline	8-12 percent

The mixed acid used was obtained at Reaction Motors, Incorporated and was manufactured by the Calco Chemical Corporation. The acid is expected to be of approximately the following composition: (Navy specification for Model CML5N engine):

Nitric acid	81-85 percent
Sulphuric acid	14-17 percent
Water	Not over 4 percent
Nitrosylsulphuric acid	Not over 1 percent
Water and nitrosylsulphuric acid combined	Not over 4.5 percent

Some of the physical properties of these materials as determined at the Lewis Laboratory were as shown.

Physical Properties	Monoethylaniline	Mixed Acid[1]
Density, g/ml, at 89.6° F	0.967	—
at 36.5° F	0.990	—
Freezing point, °F	Very thick at -80*	-65**
Viscosity, centistokes, at 70° F	—	1.4
at 32° F	—	2.0
10° F	8.5	—
-20° F	30	—
-40° F	90	6.0
-60° F	390	—

[1]A different batch was used in the ignition tests

*Cloudiness appeared at -40° F.

**Crystal formation started at -47° F.

11. Results and Discussion: The principal data for the runs comprising Phase 1 are presented in figures 4 to 7 inclusive and are summarized in table I. The figures show the propellant valve position, the propellant injection pressures, the rocket-combustion chamber pressure, and, in one instance, the rocket exhaust temperature plotted

against time measured from the initial movement of the propellant valve. The table summarizes the experimental measurements and indicates those runs for which ignition occurred satisfactorily and those runs for which ignition difficulties were encountered. No ignition was obtained in runs 15, 16, and 21, and an explosion took place on run 22.

12. The data have not been analyzed exhaustively at this date; several indications are apparent, however. First, examination of the data indicates that the principal ignition difficulty is associated with low temperatures. Runs 15, 16, 21, and 22 that evidenced either no ignition or exploded (run 22) were all at mean temperatures below -35° F. All runs at mean temperatures above -28° F evidenced satisfactory ignition. Runs 7, 9, and 10 that were at low pressures but not at low temperatures evidenced satisfactory ignition. Some effect of low pressure is noted in that the time for combustion chamber pressure to increase to its maximum is longer at low pressure (runs 7, 9, 10) than at one atmosphere at similar temperatures (runs 5, 6, 11).

13. Second, the low temperatures that caused ignition difficulties affect both the rate of opening of the propellant valve and the rate of flow of the monoethylaniline fuel by greatly increasing the viscosity of both the hydraulic fluid and the fuel. The following table taken from the data on figures 4 to 7 and table I illustrates the effect of temperature on the time for the propellant valve to open 75 percent. When the propellant valve is open 75 percent, full combustion chamber pressure usually has been reached. (see runs 5, 6, 11). (Fig. 4).

Run No.	Mean Temperature, °F	Time for Valve to Open 75%, secs.
5, 6, 7, 9, 10	86 ± 4	3.5 ± 0.2
18	13	7.3
17	2	11.5
14, 19	-11 ± 1	12.5
12	- 28	27*
15	- 35	41*
21	- 38	63*

*Extrapolated values

Low temperature greatly increases the viscosity of the monoethylaniline fuel as contrasted with the mixed acid oxidant (see paragraph 10 above). This increased viscosity for the monoethylaniline tends to increase the time between the first indication of injection pressure for the acid and the first indication of injection pressure for the aniline. A comparison of the injection pressure data for runs 5, 6, 7, 9, 10, and 11 at ambient temperatures with the injection pressure data for runs 14, 15, 19, 20 and 22 at low temperatures illustrates this point.

14. Finally, at low temperatures ignition lag of the propellants after they mix may be anticipated. This has not been studied in detail yet, but observations made by dropping one milliliter of monoethylaniline into 5 ml. of acid indicate that whereas at 23° F ignition is nearly instantaneous, at -22° F, ignition delays of about 1/2 second were obtained, and at -49° F a delay of about one second was obtained.

15. Several possibilities exist as cures or "fixes" for the principal difficulties attributable to low temperatures. These have not been studied in detail.

(a) Maintain rocket engine and propellants at moderate temperatures.

(b) Alter the composition of the hydraulic fluid and of the monoethylaniline fuel to give more suitable viscosity characteristics for the entire temperature range encountered. Alteration to the fuel should not destroy the hypergolic effect with mixed acid. No changes in the fuel have been tried. As an example of what can be accomplished with different hydraulic fluids, the following data were obtained for the propellant valve:

Temperature, °F	Time for Valve to Open, secs			Time for Valve to Close, secs		
	AN-0-366 (Hydraulic fluid specified for the engine)	AN-0-366 plus 15 % n-heptane	Silicone (Dow-Corning 500 20 cs. grade)	AN-0-366	AN-0-366 plus 15 % n-heptane	Silicone (Dow-Corning 500 20 cs. grade)
88 ± 7	5.0	4.4	5.0	9.0	8.8	8.4
-69 ± 2	180	17	13	330 - 400	46	32

(c) Make appropriate mechanical alterations to the equipment to permit suitable operation for the temperature range in which operation is desired.

16. Concluding Remarks: Satisfactory ignitions of the 220-pound thrust cylinder of the Navy Model CML5N rocket engine operating on monoethylaniline and mixed acid were obtained at all pressures investigated (altitudes up to 65,000 feet) and at temperatures above -28° F. At temperatures below -35° F no ignitions were obtained and one explosion occurred. The principal cause of the ignition difficulties appeared to be the high viscosity of the hydraulic fluid and the high viscosity of the monoethylaniline as a result of the low temperatures.

John L. Sloop

John L. Sloop, Acting Head,
Rocket Section.

Walter T. Olson

Walter T. Olson, Chief,
Combustion Branch.

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TABLE I - SUMMARY OF DATA

Run	Altitude (ft)	Mean temperature (°F)	Ignition	Combustion pressure (lb/sq in. abs)	Temperature in fuel tank (°F)	Temperature of oxidant (°F)	Temperature of engine jacket (°F)	Fuel tank pressure (lb/sq in. abs)	Oxidant tank pressure (lb/sq in. abs)	Indication by thermocouple in rocket exhaust	Remarks
				Sea level; ambient temperature							
5	0	82	Yes	308	82	82	82	447	447	----------	
6	0	82	Yes	308	82	82	82	447	447	----------	
11	0	85	Yes	328	85	85	85	460	460	----------	
				High altitude; ambient sea level temperature							
7	54,000	90	Yes	305	90	90	90	445	445	----------	
9	56,800	88	Yes	304	88	88	88	450	450	Yes	
10	64,500	85	Yes	171	85	85	85	457	457	Yes	Combustion chamber pressure tap burned out.
				Sea level; low temperature							
13	0	-8	Yes	302	-16	--------	0	460	460	Yes	
12	0	-28	Yes	280	-27	--------	-29	460	460	----------	
				High altitude; low temperature							
18	53,000	13	Yes	315	8	--------	19	451	451	Yes	Exhaust thermocouple indicated 1.4 seconds after valve moved.
17	53,700	2	Yes	300	0	--------	3	438	438	----------	
14	55,500	-10	Yes	--------------	-13	--------	-7	471	471	Yes	Combustion chamber pressure line clogged.
19	55,300	-12	Yes	285	-7	-18	-5	443	443	Yes	Exhaust thermocouple indicated 2.4 seconds after valve moved.
20	52,800	-28	Yes	287	-18	-29	-28	439	439	Yes	
15	54,900	-35	No	--------------	-31	--------	-39	460	465	Yes	
22	55,700	-38	Explosion	--------------	-30	-33	-43	443	443	Yes	Explosion about 12.2 seconds after valve moved. Exhaust thermocouple indicated about 2.7 seconds after valve moved.
21	54,900	-38	No	--------------	-29	-45	[illegible]	[illegible]	[illegible]		
16	54,500	[illegible]	[illegible]								

											pressure tap burned out.
Sea level; low temperature											
13	0	-8	Yes	302	-16	--------	0	460	460	Yes	
12	0	-28	Yes	280	-27	--------	-29	460	460	----------	
High altitude; low temperature											
18	53,000	13	Yes	315	8	--------	19	451	451	Yes	Exhaust thermocouple indicated 1.4 seconds after valve moved.
17	53,700	2	Yes	300	0	--------	3	438	438	----------	
14	55,500	-10	Yes	---------------	-13	--------	-7	471	471	Yes	Combustion chamber pressure line clogged.
19	55,300	-12	Yes	285	-7	-18	-5	443	443	Yes	Exhaust thermocouple indicated 2.4 seconds after valve moved.
20	52,800	-28	Yes	287	-18	-29	-28	439	439	Yes	
15	54,900	-35	No	---------------	-31	--------	-39	460	465	Yes	
22	55,700	-38	Explosion	---------------	-30	-33	-43	443	443	Yes	Explosion about 12.2 seconds after valve moved. Exhaust thermocouple indicated about 2.7 seconds after valve moved.
21	54,900	-38	No	---------------	-29	-45	-48	449	449	----------	
16	54,500	<-40	No	---------------	-36	--------	-60	437	437	No	All acid (16 lb) discharged. Only 0.3 of aniline in the tank discharged (1.4 lb) in over 100 seconds.

Notes: 1. Thermocouple in aniline tank was not immersed in the aniline for quantities used.

2. For runs 12 to 19 coolant was drained from the low-temperature bath, approximately 2 to 5 minutes before operation. For runs 20 to 22, engine, valve, and tank assembly were immersed in coolant up to and during rocket operation.

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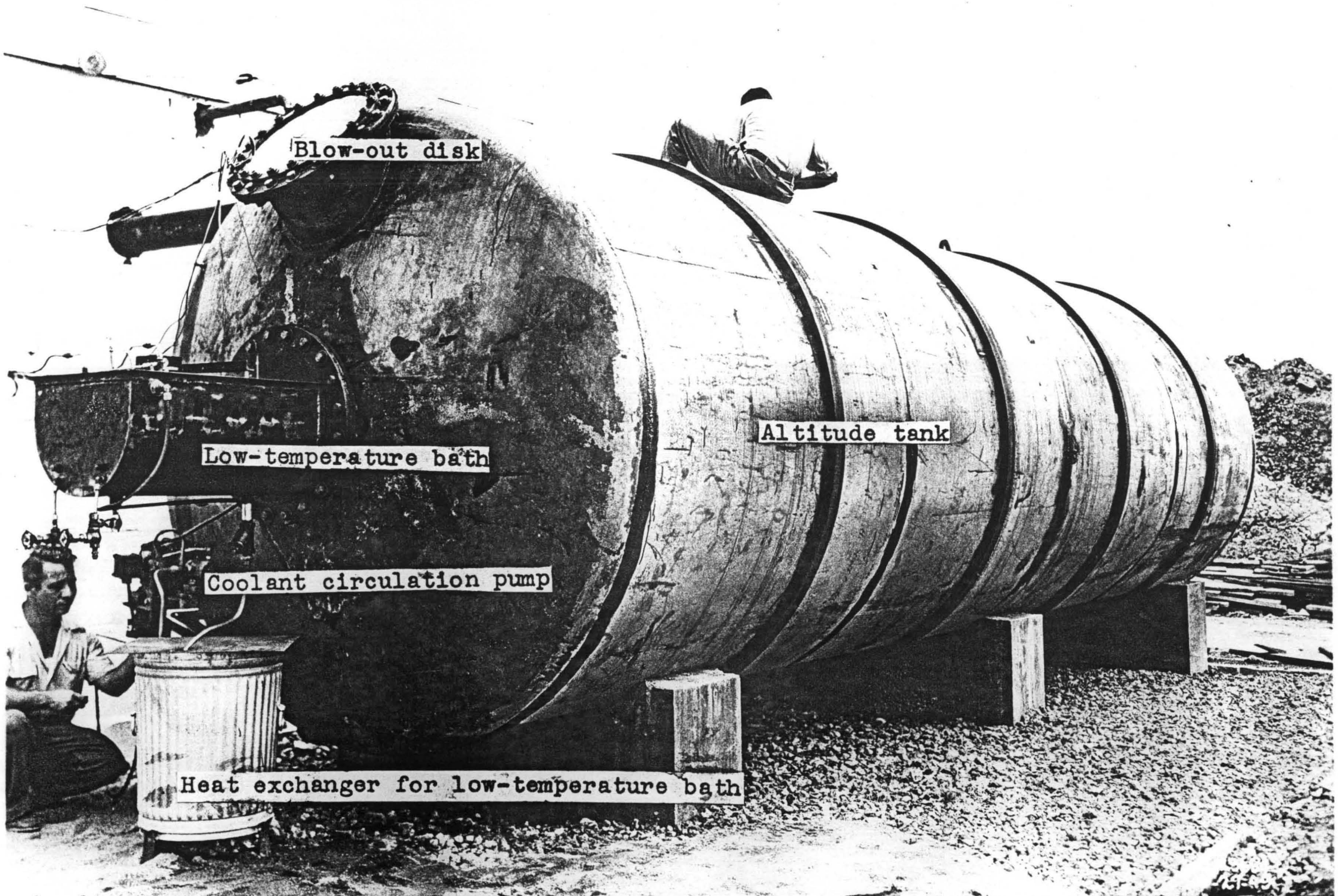
3. Mean temperature for runs 5 to 18 taken as mean of temperature of fuel tank and engine jacket just before run.

4. Mean temperature for runs 19 to 22 taken as mean of the temperature of the acid in the tank and the temperature of the engine jacket.

5. Runs 1 to 4 were trial runs with the rocket engine at sea level and ambient temperature. They started satisfactorily.



(a). Altitude tank and low-temperature bath for rocket engine for simulating high altitude conditions.

Figure 1. - Altitude tank and auxiliary equipment.

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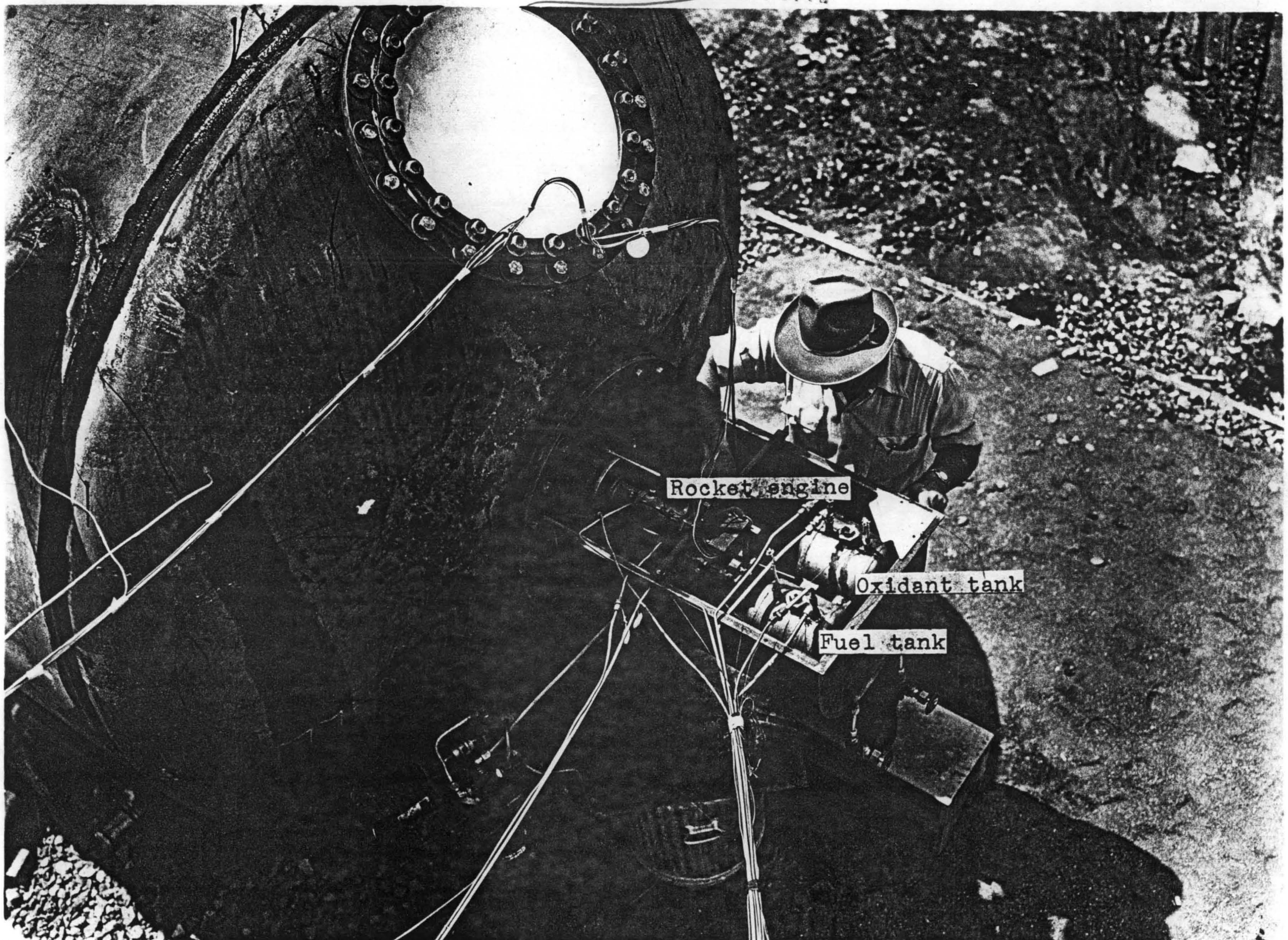

(b) The rocket engine, propellant valves, and propellant tanks mounted in low-temperature bath with engine attached to exhaust into altitude tank.

Figure 1. - Continued. Altitude tank and auxiliary equipment.

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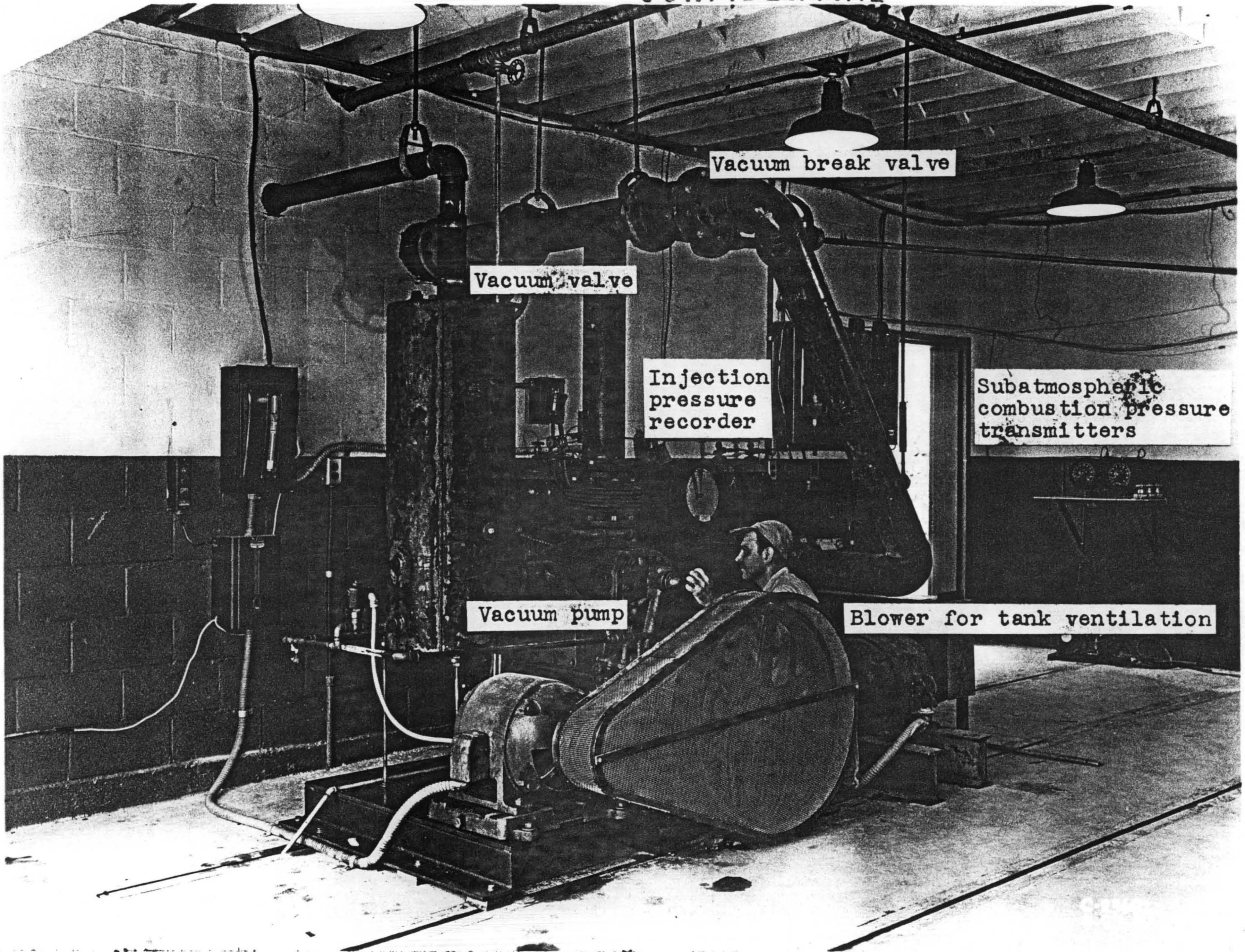

(c) Vacuum pump and tank ventilation equipment.

Figure 1. - Concluded. Altitude tank and auxiliary equipment.

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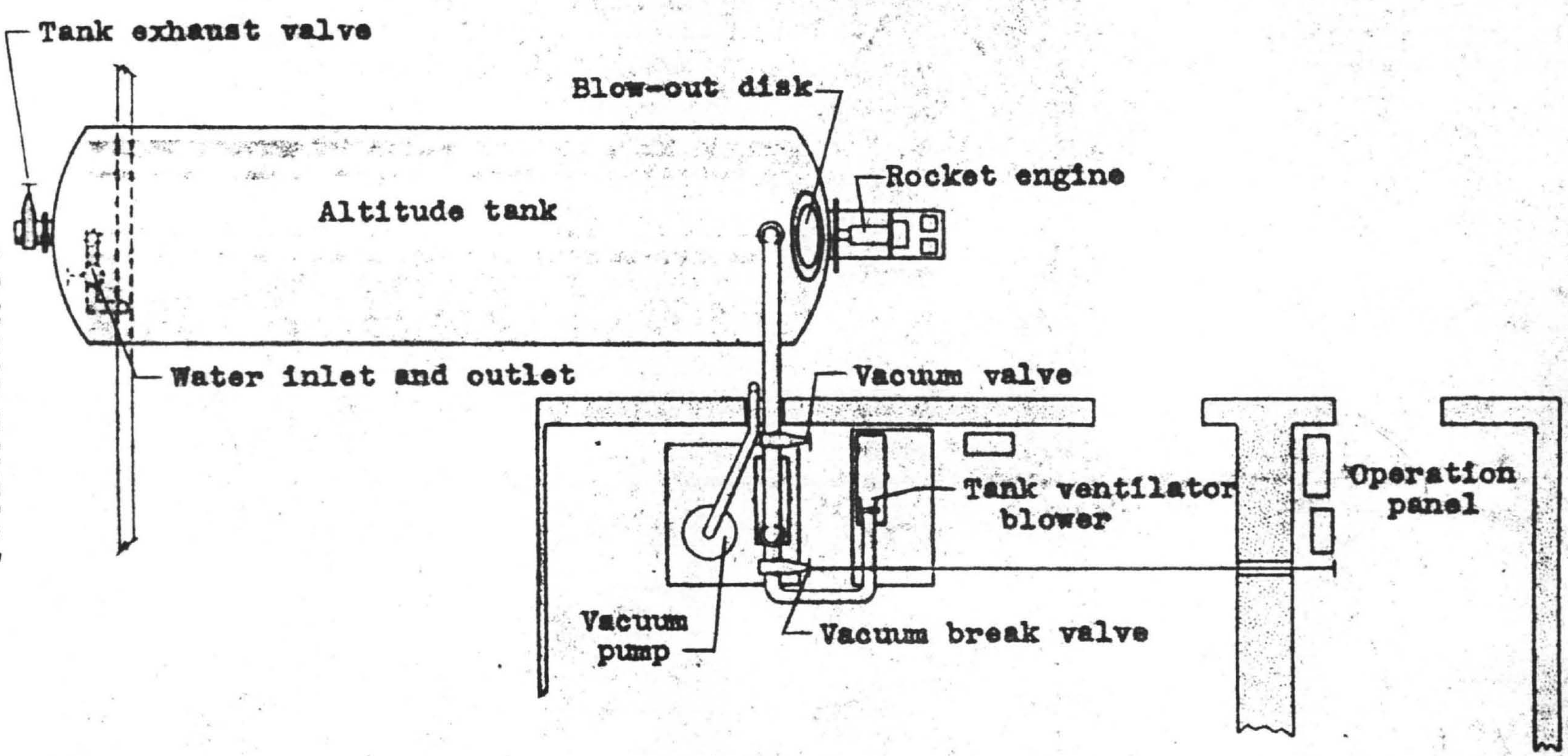

Figure 2. - Layout of altitude tank and auxiliary equipment.

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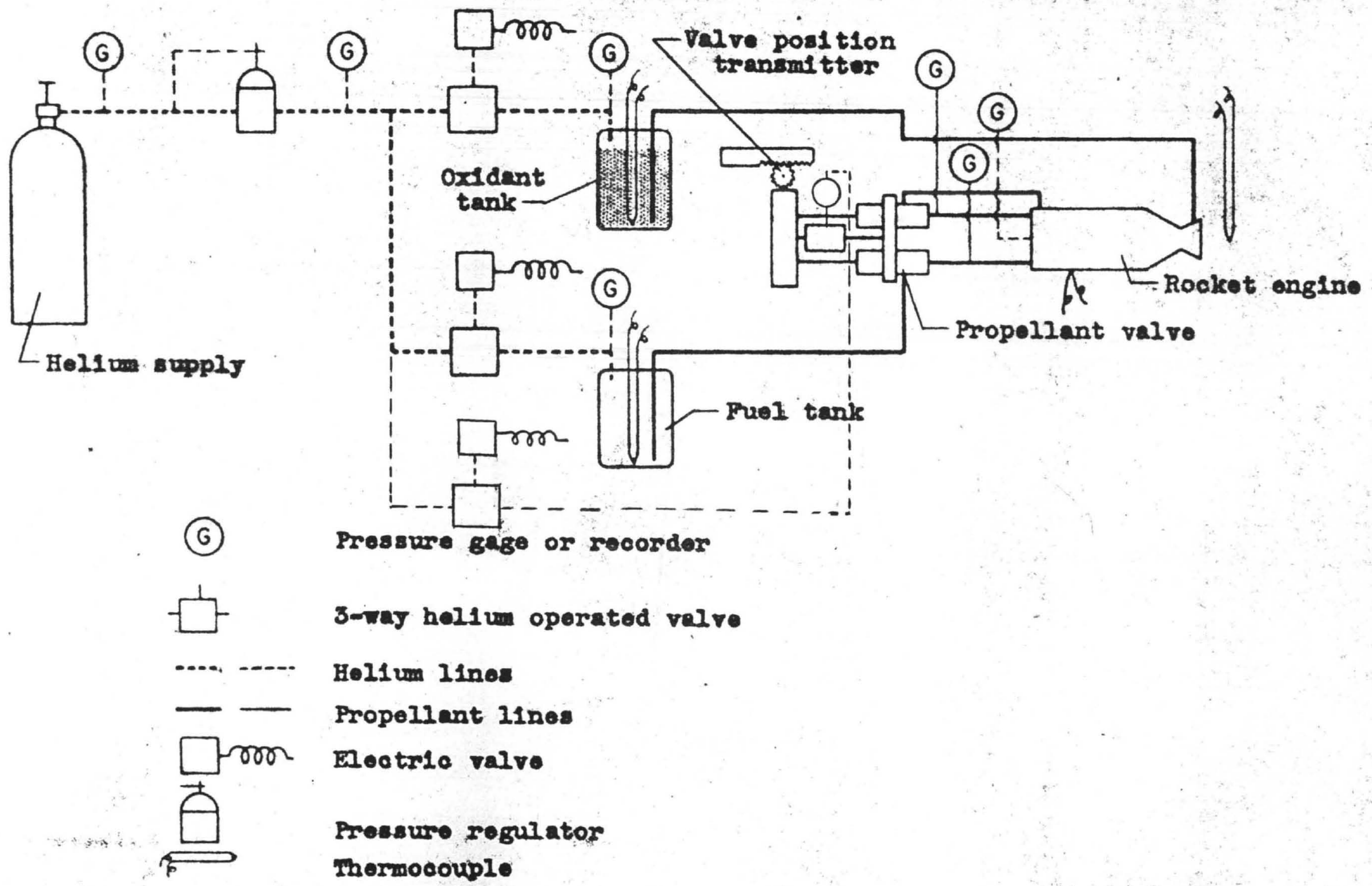

Figure 3. - Diagrammatic sketch of rocket engine and flow system.