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NASA PROGRAM APOLLO WORKING PAPER NO. 1202

THE HYPERGOLIC RECIPROCATING ENGINE ELECTRICAL POWER SYSTEM AND ITS APPLICATION TO THE LUNAR EXCURSION MODULE

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MANNED SPACECRAFT CENTER

HOUSTON, TEXAS

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THE HYPERGOLIC RECIPROCATING ENGINE ELECTRICAL POWER SYSTEM AND ITS APPLICATION TO THE LUNAR EXCURSION MODULE

By C. Dale Haines and T. E. Redding

SUMMARY

This paper describes the hypergolic reciprocator, its development history to date, and the system for which it is the prime mover. Performance testing both on Aerozine-50 fuel*/nitrogen tetroxide oxidizer and hydrogen/oxygen is discussed. The contract effort with Marquardt is discussed as well as applications to the LEM for extending the lunar stay time and emergency power. Some treatment is made of the problems associated with utilizing residual propulsion system propellants.

As of February 18, 1966, the reciprocator has had hot running for a total of 374.9 hours on hypergolic propellants and 3.1 hours on hydrogen/oxygen. Continuous runs up to 90.4 hours have been made thus far on hypergolics and approximately 43 minutes on hydrogen/oxygen. Best performance to date has been 6.2 lb/hp-hr on hypergolic propellants and 2.4 lb/hp-hr on hydrogen/oxygen.

The amount of residual propellants remaining in the descent stage of the LEM on the lunar surface can vary from approximately 200 to 1000 lb, depending on the hover time required in landing. This could be used to generate electrical power for additional lunar stay time. The system could also be activated at any point in the mission to supply emergency power by using any available propellant supplied positively at zero gravity.

INTRODUCTION

In 1962 the need for a power generation system capable of operating from residual propulsion system propellants was recognized at MSC. In

*Fifty-percent unsymmetrical dimethyl hydrazine and 50-percent hydrazine. (Propellant characteristics are given in appendix A.)

Apollo and LEM these propellants are Aerozine-50 and nitrogen tetroxide. Such a power supply could be utilized for emergencies, for backup power, or for extending operational periods such as extending the lunar stay time of LEM.

At that time two dynamic conversion methods appeared feasible for this application - a reciprocating engine and a pulse turbine. A decision was made to begin development of the reciprocator for the following reasons:

a. Minimum anticipated development problems based on technology status at that time.

b. Minimum specific propellant consumption at low power levels (1 to 3 kW).

c. The high state of development of small conventional reciprocating engines.

d. Ability to take high cycle temperatures which result from the combustion of rocket propellants.

The first objective of the program was to demonstrate the feasibility of the prime mover. If it proved feasible then system development could follow. A contract was signed in October 1962 with The Marquardt Corp., Van Nuys, Calif., to develop such an engine. This paper will present a description of the engine and system, its development history, and results of the test program to date. The present program will be discussed as well as application considerations. Its multifuel capability will be discussed in appendix B.

ENGINE DESCRIPTION

General Operational Description

The hypergolic engine is a single cylinder, liquid cooled, port exhausted reciprocator employing a two-stroke modified Otto cycle. The hypergolic propellant combination of Aerozine-50 and N_2O_4 is injected through dual, concentric, monoseat poppet valves near top dead center of each stroke. Nearly instantaneous burning occurs and hot gases expand, driving the piston down on the power stroke. As the exhaust ports are opened most of the gases escape to the vacuum of space. Any remaining gases are recompressed as the piston travels back toward top dead center and another propellant injection. The engine assembly is shown in figure 1 and engine characteristics are listed as follows:

Engine Characteristics

Specifications

1. Piston displacement: 2.057 cu in. (effective).
2. Bore: 1.380 in.
3. Stroke: 1.625 in.
4. Power (maximum): 4.5 hp (continuous); 6.0 hp (intermittent).
5. Maximum speed: 6000 rpm.
6. Expansion ratio: variable up to 40:1.
7. Engine weight (excluding flywheel): 32 lb.

Materials

1. Head: 6061-T6 aluminum alloy.
2. Cylinder: 17-7 stainless steel with dense chrome plated bore.
3. Crankcase: 6061-T6 aluminum alloy.
4. Crankshaft: stainless steel.
5. Connecting rod: 2024 aluminum alloy.
6. Piston: Rene' 41 dome, D-132 aluminum alloy body.
7. Rings: 440-C stainless steel.
8. Injector valves: 440-C stainless steel, dense chrome plated.
9. Camshaft: stainless steel.

Timing

Figure 2 shows a cross-sectional view of an early model of the hypergolic engine. The crankshaft is coupled to the camshaft by timing gears, as shown. Thus the position of the piston, which is directly-coupled to the crankshaft, can be set with respect to the valve timing which is adjustable through the cams. The operating cycle is shown in figure 3. In order to achieve a cycle near the ideal Otto cycle (constant volume combustion and exhaust) an extremely short injection period (5° of crankshaft rotation or 0.14 millisecond at 6000 rpm) was required. The short injection period required the development of the cam-and-rocker-arm operated, dual, concentric, monoseat, poppet valves shown in figure 4.

Injectors

The dual poppet consists of two concentric plungers that seal on a common seat. The plungers are lifted by cams and rockers and returned by springs. Flow does not occur until both plungers are raised. Since one can be returning to the seat while the other is still being lifted, very short valve-open times can be achieved without excessive accelerations or impacts. The propellants flow into the cylinder only during the brief interval when both plungers are off the seat. The injection system operation is shown schematically in figure 5. Since injection pressures are high (1000 to 3000 psia), close fits are necessary to reduce leakage. Therefore the valve plungers are lapped to approximately 1/10 000 of an inch diametral clearance.

Piston

A composite piston is utilized similar to the one shown in figure 6. A dome made of a heat resistant, low conductivity material is required due to the very high combustion temperatures (5600° F at O/F of 1.5) resulting from Aerozine-50 and N_2O_4. An aluminum body is presently used because it transfers heat rapidly and because it provides a compatible rubbing surface to the chrome-plated bore. However, alternate piston materials are also under investigation. The current piston is designed to allow brake mean effective pressures up to 200 psia.

Cooling

The engine is cooled by water-glycol circulating through a cooling jacket around the combustion chamber and cylinder. Very little cooling is realized from the lubricant. Present heat rejection rates at nominal load are on the order of 15 000 Btu/hr.

Lubrication

The lower end of the engine is lubricated by a form of "mist" lubrication system whereby a very small quantity of oil is continually admitted into the crankcase. The oil enters under pressure through one end of the crankshaft and is carried by centrifugal force to the bearing at the rod connection and through the rod to the wrist pin. The oil then escapes into the crankcase where a mist is created by the churning action of the crank in the tight-fitting case. This mist is carried by swirling action to the cylinder walls and to the main bearings. A scavenge pump recirculates the oil. The top end is lubricated by a conventional pressure and scavenge system.

For lunar surface applications the crankcase lubrication system may be a conventional gravity - dependent splash-type system. For short duration zero-g applications a once-through mist system may be utilized.

SYSTEM DESCRIPTION

Development of this power system to date has been concentrated on the prime mover. Limited work has been performed on the remainder of the system. Hence the following subsystem descriptions indicate the preferred approaches at the present time.

The complete hypergolic reciprocating engine space power system includes the following subsystems in addition to the basic prime mover: cooling, starting, lubrication, propellant pump, alternator, and electronics and controls. The overall system, excluding radiator, is shown in figure 7. The latest engine parts and assemblies are shown in figures 8 through 10.

Cooling Subsystem

Both the engine and alternator are liquid cooled. This cooling may be accomplished by a radiator system, in which a water-glycol solution is circulated, by a water evaporation system, or by utilizing the main vehicle cooling system. Flow may be provided by an integrally-mounted pump or by a separate motor-driven pump. The radiator may either be erectable or fabricated into the skin of the vehicle. The preferred approach for the LEM application is fixed vertical radiators.

Starter Subsystem

Several types of starters are possible: a piston-ratchet device driven by cold gas, hot gas, or a solid propellant cartridge, a battery-driven electrical starter built into the alternator, or a hydraulic starter. Low starting torque is required to spin the engine, and as soon as combustion begins the engine accelerates smoothly to a set speed. The hydraulic starter can operate on engine oil and is preferred at this time.

Lubrication Subsystem

The lubrication subsystem may or may not be designed to be gravity dependent. The method of lubrication considered for zero-gravity operation is the "mist" system discussed previously. An oil flowrate into the engine of 0.2 lb/hr is expected. For lunar surface applications a conventional dry sump system is preferred for the lower unit.

A flourine-based synthetic lubricant has been tentatively selected (DuPont PR-143) which is compatible with both the fuel and oxidizer. To date this oil has not been tested in the engine.

Propellant Supply Subsystem

The storable liquid propellants must be fed to the engine at high pressures (1000 to 3000 psia) for rapid injection. The propellants are boosted from normal supply pressure (approximately 200 psi) to operating pressures by a pump which is mounted on the engine and driven off the crankshaft. (The pump may be seen in figure 8.) The pumps have variable displacement and utilize the technology gained from the injector valve development.

Alternator and Electronics

The alternator converts shaft rotational energy into electrical energy. It mounts directly onto the engine and is liquid cooled by the engine coolant system. The alternator has a solid rotor and is excited by a combination electromagnetic and permanent magnet excitation system. The alternator produces ac power and then converts it to usable dc power by the conversion electronics package. Overall conversion efficiency expected is approximately 85 percent. Varo, Inc., is currently building two prototype alternators for this program.

Control Subsystem

The control system senses load and then optimizes engine speed and torque for minimum propellant consumption. One possible control system approach consists of a mechanical governor which can adjust mixture ratio and propellant flow by sensing engine speed and load. This is accomplished automatically by the governor which acts through the propellant supply subsystem described above.

PROGRAM HISTORY

The development of the hypergolic engine dates back to 1957-1958 when The Marquardt Corp. modified a commercial diesel engine and ran it on UDMH (unsymmetrical dimethyl hydrazine) and RFNA (red fuming nitric acid). Their next company-funded effort was in 1962 when they designed, fabricated and ran SPU-1 (Space Power Unit) just prior to receiving a contract from MSC in October 1962. SPU-1 employed both commercial engine and Marquardt-designed parts. It ran for only 54.8 seconds due to piston and injector galling problems. It employed a cam-operated poppet exhaust valve, a modified Bosch-type injection system, and was designed for 6 hp.

Phase I

The first engine developed under NASA contract was designated SPU-2. Due to the specified power level of 1 to 2 kW and the resulting size limitations, piston-controlled exhaust ports were employed instead of the overhead exhaust valve. The initial design of the dual, concentric, monoseat poppet valves was employed as well as an all-aluminum piston. Total test time achieved on SPU-2 was 4.75 minutes with failures occurring due to piston crown erosion and excessive injector valve leakage. In the last test with this engine an attempt was made to prevent all propellant leakage by mounting a pressurized silicone oil supply onto the propellant drains and applying back-pressures equal to the injection pressures to force all propellant through the engine. This resulted in an explosion at the oxidizer injector valve, set off by the impact of the injector stems on the silicone oil-oxidizer solution, which badly damaged the cylinder head and valve assembly and precluded any further engine tests. Tests were then conducted at Marquardt which verified that silicone oil was incompatible with N_2O_4 under impact conditions and that this factor caused the failure. As a result of these problems the cylinder head, injector valve assembly, piston, rings, and camshaft drive assembly were redesigned. The resulting configuration, designated SPU-2A, was tested for 23 minutes during Phase I and achieved an

SPC (specific propellant consumption) of 12.5 lb/hp-hr. This engine performed satisfactorily and demonstrated the feasibility of the storable propellant, hypergolic ignition, reciprocating engine space power concept. A final report (TMC No. 6076, Sept. 1964) summarizes the effort to this point.

Phase II

The Phase II effort began in July 1964 for the purpose of continuing engine development and testing and beginning development of key system components. Two new engines were to be fabricated, one of which would incorporate a ball and roller bearing crankcase designed to accept mist lubrication along with design improvements gained throughout the program, and one which would incorporate propellant pumps plus any later design improvements. The Phase I contract engine was designated SPU-2A-1 and the ball and roller bearing engine was designated SPU-2A-2. The final Phase II engine which would include propellant pumps would be given a new model designation, SPU-3.

The initial Phase II testing program began in July 1964 and continued until February 1965 with an accumulation of over 18 hours of hot running and 12 hours of component testing. Most of the testing was done with SPU-2A-1 to demonstrate performance and component endurance. Development milestones of the original Phase II program through February 1965 are shown graphically on figure 11. The principal problems during the early runs were with the piston, rings, and injectors. As these were solved both performance and endurance improved rapidly. In October 1964 an SPC value of 6.4 lb/hp-hr was demonstrated and in November 6.8 hours of continuous endurance running were achieved. In the meantime SPU-2A-2 was being fabricated and in February 1965 it was tested for 5.2 hours. This test demonstrated the initial feasibility of the mist lubrication system and the design integrity of the new engine.

During the latter part of the original Phase II effort the first propellant pump assembly and other SPU-3 parts were fabricated but not tested. They were given preliminary tests in September 1965.

In April 1965 authorization for additional endurance testing and system development was received. Testing on gaseous hydrogen/oxygen propellants was also authorized at this time. (The H_2-O_2 investigation was funded by NASA Headquarters (OART) and is described in appendix B.) The hypergolic endurance testing began in July 1965 and continued until August 2, 1965 when it became apparent that more development would be required on critical components to achieve the required endurance. Therefore, effort was reoriented to include additional piston and ring development, lubricant investigations, and studies of configuration and operational effects. In support of this effort MSC utilized an existing thermal analysis computer program for analyzing heat transfer in the piston. This effort is described in appendix C.

Approximately 31 hours of development testing were achieved during July and approximately 344 hours have been accumulated through February 1966. A 90.4-hour continuous run was successfully completed on January 4, 1966.

Significant accomplishments of the program to date are as follows:

1. Controlled combustion of Aerozine-50/N_2O_4 in an internal combustion engine; feasibility demonstrated June 1964.

2. Multifuel capability of engine demonstrated June 1965.

3. High speed injection system.

4. Mist lubrication system feasibility demonstrated

5. High pressure propellant pump design.

6. Domed piston design.

7. Continuous endurance demonstrated of 90.4-hours.

8. Demonstrated SPC of approximately 6 lb/hp-hr.

A test run summary is given as table I.

APPLICATION TO THE LUNAR EXCURSION MODULE

Considerable utility can be realized from the hypergolic fueled power systems both for current LEM and the extended LEM modules which are now under serious investigation. The storability of the propellants and the possibility of large quantities of residual propellants remaining in the descent stage propulsion tanks makes this power system attractive for an appreciable extension of lunar stay time for a relatively small weight penalty. Such extended durations are under study for the Apollo Applications Program (AAP) missions.

Lunar Excursion Module

The primary battery power supply on LEM allows for a mission energy growth capability of 15 percent. In the event that the estimated allowance is exceeded, the hypergolic power system could be utilized. A secondary benefit would be its availability for use as an emergency power supply at any point in the mission. The power system reliability and hence the mission success reliability would be significantly increased by the addition of the hypergolic system (i.e., if the battery system has a reliability of 0.99, an engine with a reliability of 0.9 would increase total power system reliability to 0.999).

It is quite probable that the LEM will land on the lunar surface with considerable residual (unused) propellants remaining in the descent stage propulsion tanks. (See appendix D.) Reserves are allotted for hover and translation during landing site selection, and it is very unlikely that the landing will ever be made with nearly dry propellant tanks. Assuming 100 pounds of residuals, which is equivalent to approximately 10 seconds of hover, the hypergolic reciprocator could produce approximately 15 kW-hr of electrical energy, which is significant when compared to the total mission energy profile. Also, if residuals are used, a significant weight advantage accrues to the hypergolic system over batteries and cryogenic systems since no tankage or propellant weight penalty is chargeable to the system. Figure 12 illustrates system weights for various system combinations at 1.5 kWe. Appendix F presents design and integration data for LEM.

Apollo Applications Program

Advanced Apollo concepts include a LEM taxi, shelter, lab, and an extended LEM. Extending mission time for additional experimentation, observation, and exploration is a major goal of these missions. The LEM shelter, for example, may be required to stay quiescent on the lunar surface for up to 6 months before it is used. Cryogenic power supplies would suffer a severe weight penalty in this case due to heat leak and the resulting boiloff. The hypergolic systems can readily provide mechanical power as well as electrical. Some possible applications for AAP missions are as follows:

1. Portable power supply (mechanical or electrical power).

2. Mechanical power for tools, pumps, or locomotion.

3. High peaking and spiking power.

4. Auxiliary power for spacecraft while performing maintenance on the primary system.

SYSTEMS ANALYSES

Systems analyses have been made to determine system weights and cooling system requirements for the hypergolic reciprocator system. For computing radiator design requirements, an existing computer program was utilized which can be used to determine the heat rejection capabilities. System variables put into the program are sink temperature, thermal emissivity, inlet temperature, flowrate, overall effectiveness, and type of coolant. The program has been run for a variety of conditions for both vertical and horizontal radiators on the lunar surface. Some of the results are presented in figures 13 and 14.

To determine system weights for any given set of mission requirements a special computer program was written by the authors. This program can give system weights for the reciprocator system with radiators or evaporative cooling and with or without special tankage and propellants. Table II presents the input and output data of this program.

CONCLUSIONS

The following conclusions can be made at this time:

1. The feasibility of the hypergolic reciprocator power system has been demonstrated.

2. Predicted performance appears to be reasonable (demonstrated 6.2 lb/hp-hr; predict 4 to 5 lb/hp-hr).

3. Continuous endurance of 90.4 hours has been demonstrated.

4. This type of power supply on LEM could increase mission reliability.

5. This system could provide power for emergency/backup and extended stay to LEM at very little weight penalty.

6. This power supply could provide a variety of additional services to the current and extended LEM missions.

APPENDIX A

PROPELLANT CHARACTERISTICS

Fuel

Aerozine-50 is a blend of 50 percent by weight of hydrazine (N_2H_4) and 50 percent UDMH (Unsymmetrical-Dimethyl Hydrazine $(CH_3)_2N_2H_2$). It is hypergolic (combusts upon contact) with nitrogen tetroxide (N_2O_4) and it combines the desirable features of its two components to yield a stable, high performance fuel. The UDMH provides nearly ideal physical and chemical properties for a storable rocket fuel, and N_2H_4 contributes higher performance.

Oxidizer

One of the most prominent of the oxides of nitrogen, N_2O_4 is used in storable liquid propellant systems. It is a vigorous oxidizing agent and combines very rapidly and efficiently with the amine-type fuels.

This propellant combination offers the following advantages:

1. Stability in closed tankage over long periods of storage.

2. Hypergolic ignition.

3. High performance.

4. Instant readiness.

5. Greater simplicity relative to cryogenic systems.

6. Greater reliability resulting from design simplicity and operation with propellants that are normal liquids.

Some of their chemical properties are given below for nominal conditions at 70° F and 14.7 psia:

	N_2O_4	Aerozine-50
Molecular weight (average)	92.0	45.0
Specific heat (Btu/lb-°F), 77° F	0.374	0.69
Thermal conductivity, 77° F (Btu/ft-hr-°F)	0.0755	0.151
Freezing temperature (°F)	11.84	18.8
Boiling temperature (°F)	70.07	UDMH: 146 N_2H_4: 235
Specific gravity	1.43	0.897
Vapor pressure (psia), 77° F	17.7	2.75
Critical pressure (psia)	1469	1696
Critical temperature (°F)	316.8	634
Viscosity, 77° F (lb/ft-sec)	27.96×10^{-5}	54.9×10^{-5}
Heat of vaporization (Btu/lb)	178	426
Density (lb/ft^3), 77° F	89.34 (18 psia)	56.1

The stoichiometric O/F ratio of these propellants is approximately 2.2:1. (Combustion temperature versus O/F ratio is given in figure A1.)

APPENDIX B

MULTIFUEL ENGINE PROGRAM (H_2-O_2 OPERATION)

MSC was requested by the Office of Advanced Research and Technology (OART), NASA HQ in January 1965 to include in the Marquardt contract a modification for testing with gaseous H_2-O_2 propellants. The purpose of this work was to establish the desirability of undertaking a multifuel reciprocating engine program by seeing if an existing hypergolic engine could be modified and run successfully on H_2-O_2. Marquardt modified the SPU-2A-1 for this program and the resulting configuration was called SPU-2A-3.

The following specific modifications were made:

1. Installed improved flat-seat valve configuration with larger diameter orifices.

2. Added a source of ignition.

3. Replaced incompatible teflon O-Rings with a compatible material.

4. Changed valve timing.

5. Made test facility changes for gaseous propellants.

The first ignition source attempted was a platinum-rhodium catalyst screen installed at the end of the precombustion chamber. No ignition was made and a palladium catalyst pellet was then tried. Again, results were marginal apparently due to the rapid injections, poor mixing, and extremely short exposure to the catalyst material. At this time a commercial glow plug was used which resulted in positive ignitions at all speeds and pressures. This glow plug was then installed in the head near the injector orifices.

Test runs were made from May 12 to June 5, 1965 with a total "hot" running time of 3.2 hours and total component running of 8 hours. Ignition delays and erratic running observed in the early runs were overcome and longer, smoother runs up to 43 minutes in duration were achieved. It was found that the glow plug could be deenergized after the engine was started and ignition could be sustained by hot spots on the piston and cylinder. The following is a summary of the best run:

1. Exhaust pressure - 0.6 psia nominal.
2. Inlet O_2 pressure - 1280 psia nominal.
3. Inlet H_2 pressure - 480 psia nominal.
4. O/F ratio - 3.5 ± 0.2.
5. Valve timing H_2: 22.5° BTDC to 10° ATDC.

   O_2: 20° ATDC to 35° ATDC.
6. Horsepower out 1.0 to 3.8 hp.
7. Speed 3000 to 6000 rpm.
8. SPC 2.44 lb/hp-hr at 5100 and 3700 rpm.
9. Mechanical efficiency 81 percent at 5200 rpm.
10. BMEP 90 to 185 psi.

Some extremely high pressure spikes (up to 9000 psi) due to ignition delays and detonations were observed which caused some ring damage and partial piston failure in early runs. Later runs were highly successful and much useful data were collected. All of the program's general objectives were met.

A final report on this effort was published under a separate cover.

APPENDIX C

PISTON ANALYSES - COMPUTER PROGRAM

As mentioned previously, an effort was undertaken to determine piston and ring temperature distributions by utilizing an existing generalized computer program designated HEATING*. Initially an effort was made to determine the quasi-steady-state temperature of the piston crown by means of the program's transient option; however, due to the extremely small time functions involved, the calculations were unstable and erroneous results were produced. This approach was therefore abandoned in favor of a method whereby the quasi-steady-state temperature of the piston crown was simply assumed based on actual operating experience.

Other assumptions made for the analysis were as follows:

1. No heat transfer from the inner surfaces of the piston nor down the connecting rod.

2 The piston has circular symmetry and can therefore be treated in two dimensions: axial and radial heat transfer.

3. Cylinder wall temperature is constant.

4. Heat transfer from the piston is independent of piston location within the cylinder.

The input information to the program consisted of:

1. The temperature distribution along the top of the piston crown - assumed constant for the steady-state calculation.

2. The heat transfer coefficients along the outer surfaces of the piston and rings.

3. The piston geometry (dimensions) and material thermal conductivities.

The computer results consisted of steady-state temperatures at the various region boundaries and the net heat flow from the piston to the cylinder walls.

*The HEATING Program (Heat Engineering and Transfer in Nine Geometries) by R. R. Liguori and J. W. Stephenson, Astra, Inc., January 1, 1961.

Nine different piston designs were analyzed. Shown in figures D1 and D2 are two representative piston configurations and some of the input and output data for each case. The heat transfer coefficients were provided by The Marquardt Corp. and are based on analytical and experimental findings.

APPENDIX D

CONSIDERATIONS OF RESIDUAL PROPELLANTS

"Residual" as used here indicates the quantity of rocket propellants (both fuel and oxidizer) remaining in the primary propulsion system tanks after the rocket engine has performed its useful function. For example, when the LEM lands on the moon the descent stage will likely contain a considerable amount of residual propellants in the descent propulsion system tanks. This is true for the following reasons:

1. Propellant loading tolerances necessitate overloading due to density variations, tank manufacturing tolerances, vertical alinement, tank stretching and shrinkage, countdown and engine bleedoffs, and loading equipment errors.

2. Some propellants are trapped in the engine and propellant system.

3. Considerable propellant is allotted for mission contingencies (ΔV contingency).

4. Hover time may be decreased depending on ease of locating a landing site and astronaut proficiency in landing.

5. The mission would probably be aborted if the propellant quantity was marginal before landing.

A primary justification for developing the hypergolic reciprocator was the need for a system that could use these residuals to produce useful power. Thus the engine has been designed to accept the same propellants in the same ratio used by the LEM descent propulsion system. ($O/F = 1.6$) It must be noted, however, that this is only the nominal O/F ratio of the descent engine and that the O/F ratio during actual burning (according to engine specifications) may vary as follows: O/F $1.6 \pm .06$ between 0 and 50 percent of maximum thrust, $O/F = 1.6 \pm .02$ between 70 and 100 percent of maximum thrust, and linearly from $1.6 \pm .06$ to $1.6 \pm .02$ between 50 and 70 percent of maximum thrust. Approximately half of the LEM descent profile is above 50 percent maximum thrust. Thus, if, for example, 97.5 percent of the propellants originally loaded are burned at a slightly high O/F (i.e., at $O/F = 1.64$) the remaining 2.5 percent (residuals) will be at a low ratio ($O/F = 0.726$). Conversely, if the majority is consumed at a low O/F, the residuals will be at an O/F higher than nominal. Thus the reciprocator was designed to accept a variable O/F and controls are provided to adjust the mixture ratio for optimum usage on the lunar surface.

APPENDIX E

ENGINE TEST FACILITY

For development testing of the hypergolic engine a semi-self-contained-portable test stand was designed and fabricated. First engine tests were performed at the main Marquardt plant in Van Nuys, California, but conflicts in test cell scheduling and expense prompted a move to the Saugus test facility where all subsequent testing has been performed.

The test stand contains all of the necessary equipment to determine engine horsepower, specific propellant consumption, heat rejection, and lubrication requirements. The propellant supply tankage, data recording equipment, and control panel are separate units.

Major components of the test stand are as follows:

1. The test base mounting with mounts for auxiliary equipment and instrumentation.

2. An engine starting mechanism and power source.

3. An engine power absorption unit (presently a water-dynamometer, later to be an alternator).

4. A self-regulating engine preheat and cooling system.

5. An engine oil supply and oil cooling system.

6. A propellant supply system.

The engine mounted in the test stand is shown is figure E1.

The cooling system pumps coolant through the engine and automatically regulates temperatures to any desired inlet condition up to 250° F. The lubrication system has a pressurizing pump and a scavenging pump and is capable of pressures from 5 to 100 psi and flowrates from 0.05 to 2.0 gpm. The propellant pressurization and supply system is capable of delivering propellant from 100 to 6000 psi at flowrates from 1 to 150 pounds per hour. The exhaust steam ejector system can provide a continuous low pressure exhaust system operating environment from ambient to 0.3 psia. The initial starting system used a battery-motor and turned the engine through an overrunning (one-way) clutch. This was later modified to use a high speed air motor operating through a mercury clutch. The water dynamometer provides a smoothly regulated load over the engine operating spectrum.

Test stand instrumentation provides the following data:

1. Propellant flowrate, temperature, and pressure.

2. Lubricant and coolant temperature, flow, and pressure.

3. Engine rpm, torque, and exhaust temperature and pressure.

4. Engine cylinder pressure, and crankshaft position for P-V diagram.

5. Data presented visually and recorded.

APPENDIX F

DESIGN AND INTEGRATION DATA FOR LEM

a. Net power - Power level dictated by statement of work is 2 kWe for 15 hours and 800 We for 240 hours design life. Marquardt (TMC) used a design point of 4.5 hp to meet the above requirements. It is believed the engine power level could be increased by increasing bore and stroke. Output voltage from present alternator is 29 Vdc ± 1/2 V.

b. Oxidizer/fuel ratio capability - The O/F is variable by changing supply pressure of either fuel or oxidizer. Usual operating range is from 1.6 to 2.0.

c Specific propellant consumption (SPC) - Present demonstrated SPC is 6.2 lb/hp-hr at the shaft at 3.2 hp, 3200 rpm, and 200 psi BMEP. The SPC at 2.4 hp, 2800 rpm, is 8 lb/hp-hr at the shaft. Data can be extrapolated to give SPC of 4 to 5 lb/hp-hr at higher BMEP. With continued development and especially with a larger engine, SPC should be approximately 5 lb/hp-hr which is approximately 8.4 lb/kW-hr at the dc terminals.

d. Heat rejection - Present heat rejection rate to coolant water (with no attempt to optimize) is 16 840 Btu/hr at 3 hp and 15 280 Btu/hr at 2 hp. The ΔT of H_2O is 30° to 36° F and flowrate is 1 gpm. It is believed that these rates can be reduced to 13 000 to 14 000 Btu/hr with development and larger engine. The ΔT of the oil is presently 30° to 50° F (Brayco 443, 30 weight). In the flight design no cooling is anticipated from the oil. It is expected that the surface temperature of the engine in an insulated compartment will be near the coolant temperature, or 200° to 280° F. It is estimated that the four shock mounts located in the engine base will operate at approximately 150° to 200° F. The exhaust pipes are presently running at approximately 1080° F at 2.4 hp and 1330° F at 3.2 hp. The engine compartment and exhaust lines should be insulated so that there is only minimum heat leak to the vehicle.

e. Radiator - Two radiator configurations have been considered: vertical-built into the vehicle skin, and a horizontal-erectable-radiator (both radiating from one side only). Evaporative cooling has been considered in conjunction with radiators and may be used to decrease radiator area. With present heat rejection rates appromimately 80 ft^2 of vertical radiators would be required on the lunar surface. This may be reduced to 60 ft^2 with development and larger engine sizes. Expected weight of the radiator is from 0.5 to 1 lb/ft^2. For the drill application the radiator may be erectable away from the shelter and two sides may be utilized for a smaller panel size. A small cold plate of 1 to 2 ft^2 may be required for electrical components. Expected weight of cold plate is 1 lb/ft^2.

f. Configuration and weight - At present the prime mover in prototype configuration and the system is in laboratory breadboard status. It is believed that the flight system can be packaged into an envelope 18 in. high, 20 in. long, and 10 in. wide, excluding radiator. Expected weight is as follows:

Prime mover	26 lb
Alternator, electronics, cold plate	21
Accessories and oil system	23
Radiator, 0.5 lb/ft^2	30
	100 lb

This is exclusive of plumbing and electrical systems.

g. Installation constraints -

(1) Insulate the exhaust pipes and engine compartment.

(2) Exhaust products can be harmful, corrosive and hot.

(3) Vibration of engine must be isolated from vehicle with shock mounts.

(4) System must be shielded to preclude EMI.

(5) If system is designed to use residual propulsion system propellants, astronaut must monitor quantity, O/F ratio, and make settings on engine control panel. Some external power is required to activate the propellant system and for start signal.

h. Insulation - An insulated compartment should be provided with the engine mounted upright in the LEM on the lunar surface. Insulation would be required behind the radiator if it is a part of the vehicle skin.

i. G load and vibration - It is believed that the engine and system are not especially sensitive to high G loads or vibrations and that LEM specifications can be met.

j. Propellant interface - The present system is designed to accept unfrozen residual propellants. However, if desirable the system could have its own propellant tanks. If using residuals, a start signal would activate solenoid valves to bring propellant to the engine. A propellant boost pump is designed to accept propellants at any pressure above 25 psi and boost them to operating pressure (usually 1300 to 1700 psi).

k. Electrical interface - The control panel would have an O/F control, manual override to the governor, fuel and oxidizer on and off, and a starter switch. External power would be required for arming the propellant system and for the start signal. The alternator output may be tied into the vehicle main bus. System monitoring equipment may be installed as required.

l. Start and stop - Start sequence is as follows, assuming using residuals:

(1) Astronaut monitors descent tanks after landing to determine quantity, pressure, and O/F.

(2) Astronaut sets O/F of engine.

(3) Vent the propellant lines between the tanks and propellant pump.

(4) Activate fuel and oxidizer.

(5) Start signal initiates hydraulic starter which spins engine and propellant pump. Ignition is immediate and engine speed increases to governor setting. For stopping, the stop command shuts off the propellant supply pressure and the engine coasts to a stop.

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TABLE I.- RECIPROCATOR TEST RUN SUMMARY

Phase	Engine	Test number	Date	Timing	O/F	Maximum rpm	Maximum hp	Minimum SPC, lb/hp-hr	Maximum BMEP, psi	Duration	Remarks
I	SPU-2	1	15 Aug 63	TDC[a] 5° dwell	----	1200	----	----	----	40 sec	Starting test, N_2O_4 leak in facility
I	SPU-2	2	15 Aug 63	10° ATDC[b], 5° dwell	----	2064	0.58	----	----	105 sec	Piston failed (all aluminum)
I	SPU-2	3	22 Aug 63	1° ATDC, 10° dwell	2.	2870	0.44		----	143 sec	Piston failed (all aluminum)
I	SPU-2	4	10 Sep 63	TDC, 10° dwell	----	450	----	----	----	(starting)	Explosion, N_2O_4 reaction with silicone oil
I	SPU-2A	5	05 Jun 64	TDC, 5° dwell	----	----	----	----	----	90 sec	Define starting characterics
I	SPU-2A	6	05 Jun 64	TDC, 5° dwell	1.12	1500	0.44	21.0	58.1	2 min	Starting condition determination
I	SPU-2A	7	05 Jun 64	TDC, 5° dwell	1.35	1850	0.87	15.4	75.4	5 min 47 sec	First run for performance evaluation
I	SPU-2A	8	16 Jun 64	TDC, 5° dwell	1.37	3450	0.60	24.4	34.4	3 min 25 sec	Evaluation of modified piston
I	SPU-2A	9	18 Jun 64	3° BTDC[c], 8° dwell	1.21	4500	1.96	12.5	104	11 min 45 sec	Feasibility demonstration
II	SPU-2A-1	10	30 Jul 64	----	0.52	3100	2.34	14.1	----	11 min 42 sec	Injector galling
II	SPU-2A-1	11	31 Jul 64	----	0.79	4900	3.55	10.9	----	21 min	Depleted fuel supply
II	SPU-2A-1	12	17 Aug 64	----	----	----	----	----	----	30 min	Lost cooling water
II	SPU-2A-1	13	18 Aug 64	----	----	----	----	----	----	18 min	Chrome plating failed on injectors
II	SPU-2A-1	14	19 Aug 64	3° BTDC, 10° dwell	1.22	2100	2.4	8.3	228	12 min	Faulty injectors
II	SPU-2A-1	15	19 Aug 64	----	----	----	----	----	----	1 min 45 sec	Faulty injectors
II	SPU-2A-1	16	11 Sep 64	----	----	----	----	----	----	1 min 30 sec	Faulty injectors
II	SPU-2A-1	17	01 Oct 64	3° BTDC, 10° dwell	1.06	5000	3.6	11.05	152	12.5 min	Sticky oxidizer valve
II	SPU-2A-1	18	06 Oct 64	3° BTDC, 10° dwell	----	3200	1.0	---	62	6.5 min	Oxidizer flow meter malfunction
II	SPU-2A-1	19	07 Oct 64	3° BTDC, 10° dwell	.95	4650	4.74	8.86	202	7.5 min	Scheduled shutdown
II	SPU-2A-1	20	07 Oct 64	3° BTDC, 10° dwell	1.37	4350	4.4	8.3	203	16.0 min	BMEP traverse
II	SPU-2A-1	21	08 Oct 64	3° BTDC, 10° dwell	1.07	1990	2.55	7.25	254	24.0 min	Oxidizer valve froze, overspeed - failed rod and piston
II	SPU-2A-1	22	16 Oct 64	----	----	----	----	----	----	30 min	Injector valve component test
II	SPU-2A-1	23	19 Oct 64	3° BTDC, 10° dwell	2.17	1870	2.77	6.41	293	13.2 min	Water seal failure
II	SPU-2A-1	24	21 Oct 64	3° BTDC, 5.5° dwell	----	3500	---	----	----	9.9 min	Water overtemperature

[a]TDC - Top dead center - piston position.

[b]ATDC - After top dead center.

[c]BTDC - Before top dead center.

TABLE I.- RECIPROCATOR TEST RUN SUMMARY - Continued

Phase	Engine	Test number	Date	Timing	O/F	Maximum rpm	Maximum hp	Minimum SPC lb/hp-hr	Maximum BMEP, psi	Duration	Remarks
II	SPU-2A-1	25	21 Oct 64	3° BTDC[c], 5.5° dwell	----	3500	----	----	----	5.1 min	Water seal failed
II	SPU-2A-1	26	23 Oct 64	3° BTDC, 5° dwell	----	4000	----	----	----	9.1 min	Readjust injector valves
II	SPU-2A-1	27	23 Oct 64	3° BTDC, 5° dwell	----	4000	----	----	----	6.5 min	Readjust injector valves
II	SPU-2A-1	28	23 Oct 64	3° BTDC, 5° dwell	----	4000	----	----	----	6.9 min	Readjust injector valves
II	SPU-2A-1	29	29 Oct 64	3° BTDC, 5° dwell	1.44	2925	2.89	9.00	195	11 min	Coolant over temperature (200° F)
II	SPU-2A-1	30	29 Oct 64	3° BTDC, 5° dwell	1.7	2250	2.32	8.91	204	20 min	Arbitrary shutdown, all OK
II	SPU-2A-1	31	02 Nov 64	----	----	----	----	----	----	2 hr 10 min	Depleted propellant supply
II	SPU-2A-1	32	06 Nov 64	----	----	----	----	----	----	20 min	Overload-condition, piston failed
II	SPU-2A-1	33	12 Nov 64	3° BTDC, 5° dwell	1.5	----	----	8.48	200	6 hr 49 min	Arbitrary shutdown (two tanks refills) all excellent except small segment of top L-ring missing
II	SPU-2A-1	34	02 Feb 65	----	----	3000	----	----	----	36.44 min	Checkout new engine (3 runs)
II	SPU-2A-2	35	04 Feb 65	----	1.5	3000	2.1	8.5	----	5 hr 11 min	Cylinder distortion, blocked water passage, resulted in failed piston
II (Mod I)	SPU-2A-2	36	01 Jul 65	3° BTDC, 10° dwell	----	3800	2.3	----	----	2.7 min	Check oxidizer flow meter
II (Mod I)	SPU-2A-2	37	01 Jul 65		----	3900	1 87	----	95	14.2 min	Water seal leak
II (Mod I)	SPU-2A-2	38	07 Jul 65		----	5000	2.5	----	59	6.95 min	Distortion of cylinder spacer plate - high chamber pressures
		39	12 Jul 65	1° BTDC, 10° dwell	----	2800	2.8	----	158	1.98 min	Repair visual RPM indicator
		40	12 Jul 65		1.96	2950	2.95	6.6	198	32.6 min	Scheduled shutdown
		41	12 Jul 65		3.04	3950	2.49	8.2	125	46.6 min	Repair leak in facility oxidizer pump
		42	12 Jul 65		1.9	4100	2.6	8.4	127	3 hr 2 min	Repair leak in facility oxidizer pump
		43	13 Jul 65		1.7	4100	4.2	7.6	204	4 hr 37 min	Lost hypergolic ignition, fuel rich
		44	13 Jul 65		0.7	5000	3.8	----	150	12.1 min	Lost hypergolic ignition, fuel rich
		45	13 Jul 65		----	4600	5	----	78	9.77 min	Piston failed when hit 5 hp. Total time accumulated, 9.3 hr
		46	20 Jul 65		1.0	3500	0.6	----	34	2 min	Repair fuel leak
		47	20 Jul 65		0.7	3500	0.6	----	34	0.66 min	Lost hypergolic ignition, fuel rich
		48	20 Jul 65		1.86	3300	1.3	13.1	83	34 min	Change oil pressure setting
		49	20 Jul 65		1.63	3950	2.2	10.3	109	1 hr 26 min	Change oil pressure setting

[b]BTDC - Before top dead center.

TABLE I.- RECIPROCATOR TEST RUN SUMMARY - Continued

Phase	Engine	Test number	Date	Timing	O/F	Maximum rpm	Maximum hp	Minimum SPC, lb/hp-hr	Maximum BMEP, psi	Duration	Remarks
II (Mod I)	SPU-2A-2	50	20 Jul 65	1° BTDC[c], 10° dwell	----	3500	0.6	----	34	15 min	Change oil pressure setting
		51	20 Jul 65		1.59	4000	1.6	9.8	109	4 hr 36 min	Scheduled shutdown
		52	21 Jul 65		----	4000	2.0	12.6	99	1.98 min	Change oil pressure setting
		53	21 Jul 65		1.44	4150	2.2	13.3	105	2 hr 43 min	Change oil pressure setting
		54	21 Jul 65		----	3500	0.6	----	34	2.5 min	Piston failed; cylinder also failed where thermocouples were welded on Total time on piston was 9.69 hr
		55	29 Jul 65	1° BTDC, 7° dwell	1.21	4200	1.8	10.2	87	1 hr 20 min	Piston failed
		56	30 Jul 65	1° BTDC, 8° dwell	1.26	4100	1.9	9.0	93	2 hr 31 min	Schedule shutdown, rings were failed
		57	31 Jul 65	1.5° BTDC, 9° dwell	1.39	4150	2.28	9.2	113	4 hr	Scheduled shutdown, cleaned rings
		58	31 Jul 65	1.5° BTDC, 9° dwell	1.37	4150	2.07	9.2	99	3 hr 30 min	Scheduled shutdown, cleaned rings, total time on piston 10.03 hours.
		59	02 Aug 65	1° BTDC, 8° dwell	----	----	----	----	----	----	Severe overpressure on startup failed piston, rod, cylinder, and spacer
II (Amend. 9)	SPU-2A-2	60	07 Aug 65	1° BTDC, 8° dwell	----	2300	0.46	----	45	18 min	Tried graphite, insulated piston dome, failed *dome only*
		61	29 Sep 65	1° BTDC, 5° dwell	----	4400	1.94	----	108	10.5 min	Corrected oil leak
		62	29 Sep 65	1° BTDC, 5° dwell	1.6	4950	1.77	11	86	55.2 min	Connecting rod bearing failure, rings
		63	04 Oct 65	1° BTDC, 5° dwell	1.48	4900	2.16	8	108	28.2 min	Rod bearing retainer failed, engine sized
		64	06 Oct 65		1.43	3950	1.78	11.9	110	30 min	Scheduled shutdown, inspection
		65	06 Oct 65		1.36	4050	1.62	11.3	98	12 min	Oil leaks
		66	06 Oct 65		1.45	4250	1.74	10.9	100	48 min	Scheduled shutdown, all rings broken
		67	08 Oct 65		1.07	2400	1.63	10.2	167	30.6 min	Scheduled shutdown, inspection
		68	11 Oct 65		1.5	2000	0.6	----	----	5 min	Facility malfunction, flow meters
		69	11 Oct 65		1.61	2250	1.6	9.8	174	31.2 min	Scheduled shutdown, rings wearing

[c]BTDC - Before top dead center.

TABLE I.- RECIPROCATOR TEST RUN SUMMARY - Continued

Phase	Engine	Test number	Date	Timing	O/F	Maximum rpm	Maximum hp	Minimum SPC, lb/hp-hr	Maximum BMEP, psi	Duration	Remarks
II (Amend. 9)	SPU-2A-2	70	12 Oct 65	1° BDTC[c], 5° dwell	1.60	2500	1.7	9.2	167	1.01 hr	Scheduled shutdown, severe ring wear
		71	13 Oct 65	10° BTDC[b], 5° dwell	1.76	4200	1.45	11.9	88	2 hr	Scheduled shutdown, inspection, rings OK
		72	14 Oct 65		1.[illegible]	4350	1.96	9.0	110	1 hr 53.4 min	Bearing failure, used Havolene 30
		73	20 Oct 65		1.5	1400	----	----	----	0.6 min	Engine not assembled properly
		74	20 Oct 65		1.94	4000	1.39	12.3	83	2.01 hr	Scheduled shutdown, rings OK
		75	22 Oct 65		1.81	4000	1.66	10.9	101	4.03 hr	Scheduled shutdown, inspection, bearings wearing
		76	23 Oct 65	F: 7.5° ATDC - 7.7° dwell Ox: 10° ATDC - 5.2° dwell	1.54	4250	1.79	9.5	103	8.01 hr	Scheduled shutdown, rings broken severely, combustion chamber erosion; pressure spikes
		77	29 Oct 65	10° ATDC, 3° dwell	1.43	3050	0.55	27.9	44	3 min	420 CRES rings, galled and froze engine
		78	02 Nov 65		----	----	----	----	----	30 min	Scheduled shutdown, inspection, erosion
		79	03 Nov 65		----	4000	1.6	----	----	1 hr	Scheduled shutdown, erosion
		80	04 Nov 65	10° ATDC, 4° dwell	----	2650	1.6	----	----	30 min	Scheduled shutdown
		81	05 Nov 65	10° ATDC, 5° dwell	----	----	----	----	----	1 hr	Scheduled shutdown, L-ring clipped
		82	08 Nov 65		----	----	----	----	----	23 min	Inspection of injectors
		83	08 Nov 65		----	2500	1.6	----	----	39 min	Rings gone, serious erosion
		84	09 Nov 65	1° BTDC, 5° dwell	----	----	----	----	----	30 min	Scheduled shutdown, no erosion
		85	10 Nov 65		----	----	----	----	----	30 min	Scheduled shutdown, no erosion
		86	11 Nov 65		----	----	----	----	----	1 hr	Scheduled shutdown, no erosion
		87	12 Nov 65		----	----	----	----	----	2 min sec	Tungsten carbide coated stainless rings, galled severely
		88	12 Nov 65		----	----	----	----	----	30 min	Scheduled first run with 4-ring piston shutdown, all OK, very cool in operation
		89	15 Nov 65	2° ATDC[b], 5° dwell	----	----	----	----	----	1 hr 30 min	Scheduled shutdown, erosion, clipped rings
		90	16 Nov 65	1° BTDC, 8° dwell	----	----	----	----	----	1 hr 30 min	Scheduled shutdown, erosion clipped rings.

[b]ATDC - After top dead center

[c]BTDC - Before top dead center.

TABLE I.- RECIPROCATOR TEST RUN SUMMARY - Continued

Phase	Engine	Test number	Date	Timing	O/F	Maximum rpm	Maximum hp	Minimum SPC, lb/hp-hr	Maximum BMEP, psi	Duration	Remarks
II (Amend. 9)	SPU-2A-2	91	16 Nov 65	1° BTDC, 8° dwell	----	----	----	----	----	4 hr	Scheduled shutdown, inspection
		92	17 Nov 65		----	----	----	----	----	4 hr	Scheduled shutdown, cast iron rings wearing
		93	18 Nov 65		----	2700	1.4	8	100	5 hr	Scheduled shutdown, on 440 C stainless steel rings, baseline piston, all OK, all Brayco
		94	19-21 Nov 65		2.2	2900	1.3	----	100	41 hr	Plug in Kistler port failed, piston and rings in excellent shape, total 46 hours on same piston and rings, baseline piston
		95	22 Nov 65	Ox: 1.3° BTDC - 8.3 dwell Fuel: 1.0° BTDC - 8° dwell	----	2700	1.2	----	88	10 hr	Piston no. 1 (4 ring configuration). Scheduled shutdown, all in excellent shape
		96	23 Nov 65		----	2200	1.2	----	88	14 hr 2 4 min	Engine configuration same as test 95; scheduled shutdown, all components excellent condition. Very heavy carbon buildup in exhaust manifold
II (Mod I)		97	01 Dec 65	1° BTDC - 8° dwell	----	3300	[d]2.2 and 3.3	9.0 and 7.0	135 and 198	19 hr 36 min	Engine configuration same as test 96 except for timing; unscheduled shutdown, depleted oxidizer supply. Engine designated "A"
		98	06 Dec 65		----	3200	2.3	9.3	143	1 hr 34.8 min	Engine "B"; endurance test; unscheduled shutdown, cam lobe failed
		99	06 Dec 65		----	3350	2.3 and 3.4	8.0 and 6.3	138 and 200	1 hr 36 min	Unscheduled shutdown, inner fuel valve stuck open, foreign material entered valve
		100	07 Dec 65		----	3300	2.5	9.8	148	19.2 min	Unscheduled shutdown, inner fuel valve cam lobe failed
		101	08 Dec 65		----	3400	2.5 and 3.5	10.0 and 8.2	143 and 200	10 hr 57 min	Unscheduled shutdown, engine inadvertently filled with oil during refilling of oil tank. Facility oil level gage malfunctioned
		102	08 Dec 65		----	3400	2.4 and 3.4	10.3 and 8.5	143 and 202	16 hr 50 min	Unscheduled shutdown, torque load cell failed in tension released dynamometer load

[c]BTDC - Before top dead center.

[d]Power profile conditions - 50 min "low", 10 min "high" cyclic.

TABLE I.- RECIPROCATOR TEST RUN SUMMARY - Continued

Phase	Engine	Test number	Date	Timing	O/F	Maximum rpm	Maximum hp	Minimum SPC, lb/hp-hr	Maximum BMEP, psi	Duration	Remarks
II	SPU-2A-2	103	09 Dec 65	1° BTDC[c] 8° dwell	----	3200	2.4	----	145	13.2 min	Unscheduled shutdown, dynamometer forward mount failed
		104	09 Dec 65	Ox: 1.25° BTDC - 7.75° dwell; Fuel: 1° BTDC - 7.75° dwell	----	----	----	----	----	4.2 min	Scheduled shutdown, facility checkout run
		105	09 Dec 65		----	----	----	----	----	1.8 min	Unscheduled shutdown, fuel valve housing top "O" ring leaked
		106	10 Dec 65	Ox: 1° BTDC - 8° dwell; Fuel: 0.7° BTDC - 8° dwell	----	3300	2.5 and 3.3	9.3 and 7.9	150 and 200	4 hr 59 min	Unscheduled shutdown, leakage from P_c port. Knurling of piston skirt caused excessive oil consumption
		107	11 Dec 65	Ox: 1° BTDC - 8.5° dwell; Fuel: 0.75° BTDC - 7.75° dwell	----	3350	2.5 and 3.4	10.0 and 7.9	148 and 198	7 hr 58 min	Unscheduled shutdown, partial blockage of crank internal oil passage with foreign material caused oil starvation resulting in partial piston seizure
		108	12 Dec 65		----	3350	2.5 and 3.4	10.0 and 7.9	148 and 198	2 hr 56 min	Unscheduled shutdown, leakage from P_c port
	"A"	109	10 Jan 66	Ox: 1.1° BTDC - 8° dwell; Fuel: 0.8° BTDC - 8° dwell (outer valves open first)	----	3200 and 3200	2.4 and 3.2	----	150 and 200	1 hr 48 min	Engine "A", flat poppet valves; model no. 1 piston w/440c rings. Slip-on cams. Shutdown because of oil filters caused excessive flow restriction. Flow flow resulted in frozen front camshaft bearing; however, no damage
	"B"	110	10 Jan 66	Ox: 1.2° BTDC - 8° dwell; Fuel: 0.9° BTDC - 8° dwell (outer valves open first)	----	3200 and 3200	2.4 and 3.2	----	150 and 200	1 hr 13 min	Engine shutdown because of loss of oxidizer injection pressure. Oxidizer injection valve stuck in open position
	"A"	111	11 Jan 66	Ox: 1° BTDC - 8° dwell; Fuel: .5° BTDC - 8° dwell (outer valves open first)	----	----	----	----	----	12 min	Engine shutdown because of oxidizer injection pressure loss. Oxidizer injection valve stuck in open position.
	"B"	112	12 Jan 66	Ox: 1.2° BTDC - 8° dwell; Fuel: 1.0° BTDC - 8° dwell (outer valves open first)	----	3400 and 3500	2.5 and 3.7	10.1 and 7.9	148 and 207	30 hr 13 min	Engine shutdown because of excessive leakage of facility oxidizer pump. Engine failed to restart because of hole burned through center of piston crown.

[c] BTDC - Before top dead center.

TABLE I.- RECIPROCATOR TEST RUN SUMMARY - Concluded

Phase	Engine	Test number	Date	Timing	O/F	Maximum rpm	Maximum hp	Minimum SPC, lb/hp-hr	Maximum BMEP, psi	Duration	Remarks
II (Mod I)	SPU-2A-2 "A"	113	13 Jan 66	Ox: 1° BTDC - 8° dwell; Fuel: .5° BTDC - 8° dwell (outer valves open first)	----	3400 and 3500	2.5 and 3.7	----	148 and 207	6 hr 3 min	Retapped oxidizer valve; engine stopped. Hole burned through center of piston crown
	"A"	114	28 Jan 66	1° BTDC - 8° dwell (inner valves open first)	----	2800 and 3200	2.4 and 3.3	7.8 and 6.3	170 and 202	90 hr 25 min	Engine "A" w/thick piston crown center, silver filled bolt. Engine shutdown because of abrupt change in oil consumption characteristics. Worn connecting rod wrist pin bearing allowed a higher oil flow rate than the facility scavange pump was set to handle. When scavange pump saturated, engine oil consumption abruptly increased
	"B"	115	02 Feb 66		----	2800 and 3200	2.4 and 3.2	8.7 and 6.8	172 and 200	8 hr 8 min	Unscheduled shutdown, wrong bolt used to connect dome to piston skirt; dome came loose from piston
	"B"	116	08 Feb 66		----	2800 and 3200	2.4 and 3.[illegible]	7.3 and 5.9	170 and 200	4 hr 50 min	Controlled shutdown, unstable operation apparently because of too cold propellants
	"B"	117	10 Feb 66		----	2800 and 3200	2.4 and 3.2	9.4 and 8.1	168 and 198	4 hr 32 min	Tried a low O/F ratio ~ 1.4 to keep piston cool, burned piston. Failure attributed to unstable combustion caused by use of SPU-2A-1 cams and hydrazine rich fuel

[c]BTDC - Before top dead center.

TABLE II.- INPUT AND OUTPUT DATA FOR SYSTEM WEIGHT COMPUTER PROGRAM

(a) Input data

No.	Symbol	Definition	Present Value	Attainable value
1	____	Theoretical energy content	2750 Btu/lb	2950 Btu/lb
2	____	Combustion efficiency	≈54.5 percent	≈72.5 percent
3	EE	Cycle efficiency	≈29 percent	≈35 percent
4	EM	Mechanical efficiency	85 percent	85 percent
5	____	Percent of combustion energy lost to exhaust	≈25 percent	≈25 percent
6	CL	Percent of combustion energy lost to coolant	≈46 percent	≈40 percent
7	AE	Alternator efficiency	85 percent	89 percent
8	PC	Power conditioning efficiency	85 percent	85 percent
9	A	Average power level	variable (kWe)	variable (kWe)
10	REM	Total energy requirement	variable (kW-hr)	variable (kWe-hr)
11	SPC	Specific propellant consumption	5.9 lb/hp-hr	4-5 lb/hp-hr
12	OFR	Oxidizer to fuel ratio	1.6	1.6
13	FD	Fuel density	54.7 lb/ft^3	54.7 lb/ft^3
14	OD	Oxidizer density	88.8 lb/ft^3	88.8 lb/ft^3
15	FTCA	Fuel tank factor	6.03 lb/ft^3 propellant	6.03 lb/ft^3 propellant
16	FTCB	Fuel tank fixed weight	3 lb	3 lb
17	OTCC	Oxidizer tank factor	5.46 lb/ft^3 propellant	5.46 lb/ft^3 propellant
18	OTCD	Oxidizer tank fixed weight	3 lb	3 lb

TABLE II.- INPUT AND OUTPUT DATA FOR SYSTEM WEIGHT COMPUTER PROGRAM - Continued

(a) Input data

No.	Symbol	Definition	Present value	Attainable value
19	PSCE	Pressurization system factor	2.7 lb system/ft^3 propellant	2.7
20	CLR	Coolant flowrate	300 lb/hr, variable	150 lb/hr, variable
21	CSH	Coolant specific heat	1.0 Btu/lb °F	0.82 Btu/lb °F
22	TCM	T maximum coolant, °R	646°R	700°R
23	—	—	—	—
24	RSE	Radiator surface emmissivity	0.85	0.85
25	AST	Average sink temperature, °R, horizontal	535°R	535°R
26	RWC	Radiator weight factor, lb/ft^2	0.75	0.75
27	FIW	Fluid inventory, lb	6	4
28	PHE	Pump and heat exchange weight, lb	0	0
29	ESW	Engine and accessories weight, lb	50	35
30	AEW	Alternator and electronics weight, lb	25	20

TABLE II.- INPUT AND OUTPUT DATA FOR SYSTEM WEIGHT COMPUTER PROGRAM - Concluded

(b) Output data

Symbol	Definition
B	Shaft hp, average
CC	Indicated hp, average
D	Energy after combustion, horsepower, average
E	Total propellant weight, lb
F	Lb, oxidizer
G	Lb, fuel
H	Oxidizer tankage system weight, lb
AI	Fuel tankage system weight, lb
AJ	Pressurization system weight, lb
DD	Rate of heat lost to coolant, Btu/hr
AK	Water evaporator weight, lb
AL	Water weight, for evaporator, lb
AM	Water container weight, lb
AN	Average radiator surface temperature, °R
O	Radiator area, ft^2
P	Radiator weight, lb
Q	Total radiator system weight, lb
R	TOTAL POWER SYSTEM WEIGHT WITH EVAPORATIVE COOLING
S	TOTAL POWER SYSTEM WEIGHT WITH RADIATOR

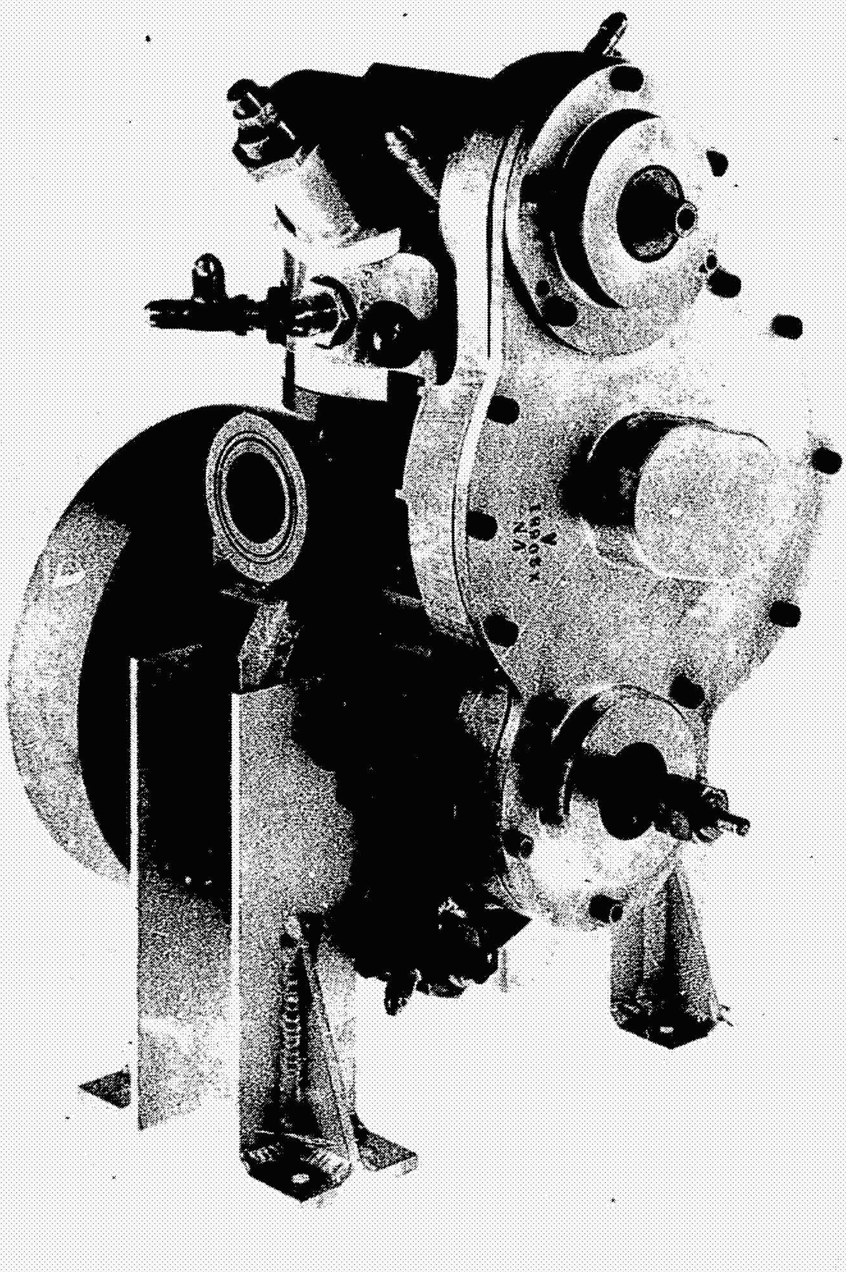


Figure 1.- Engine assembly.

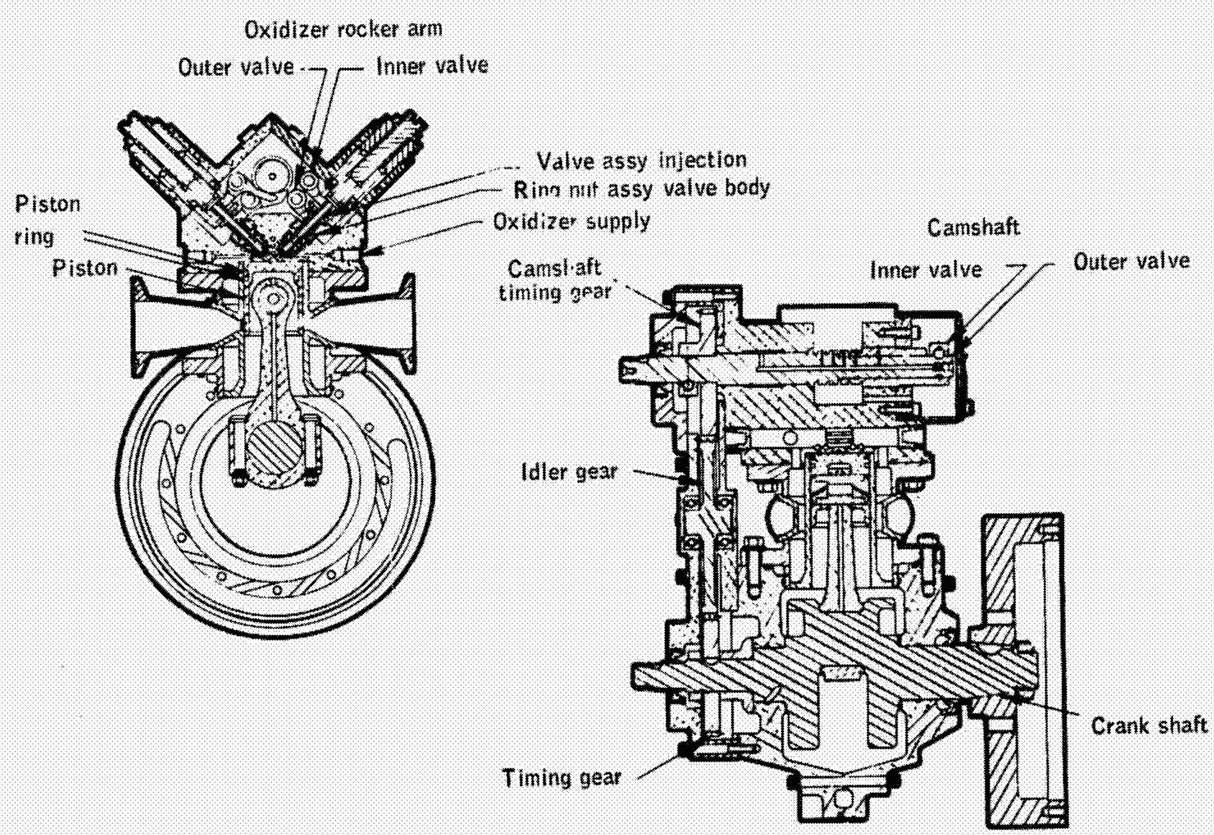


Figure 2.- Engine cross-section.

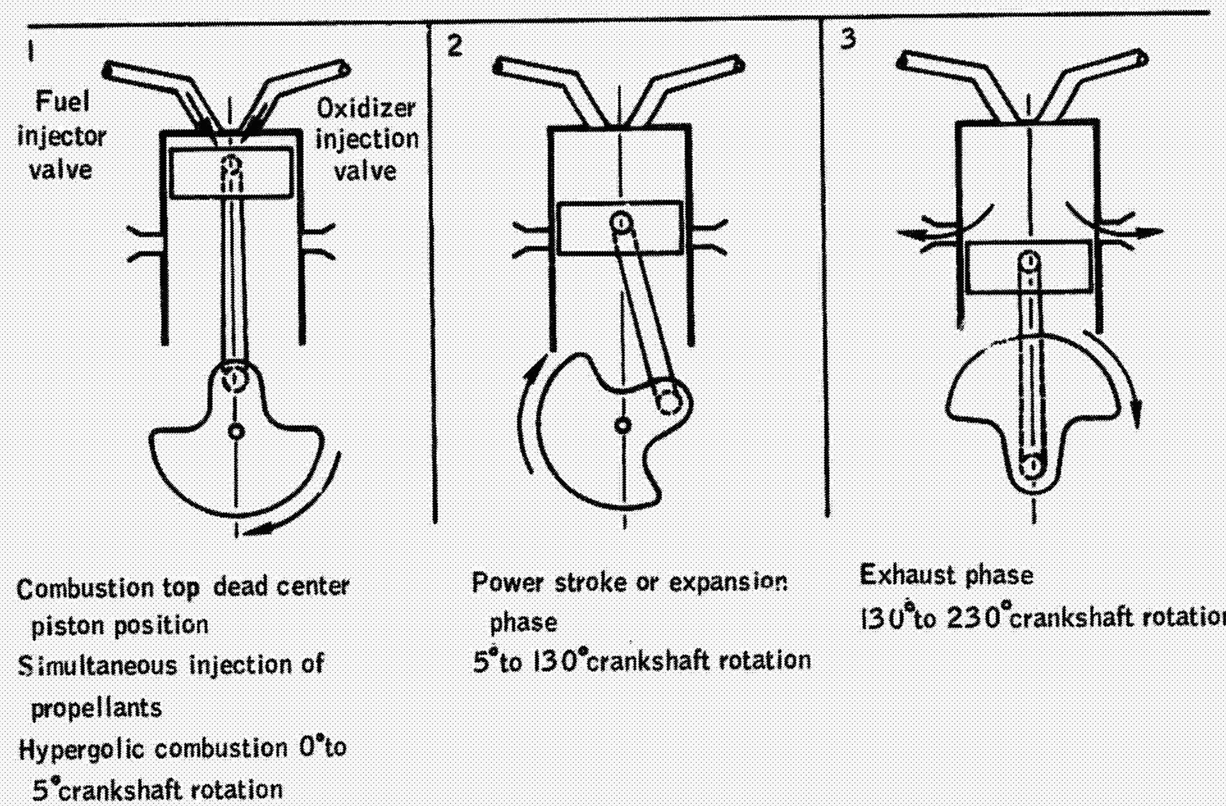


Figure 3.- Hypergolic ignition engine operating cycle.

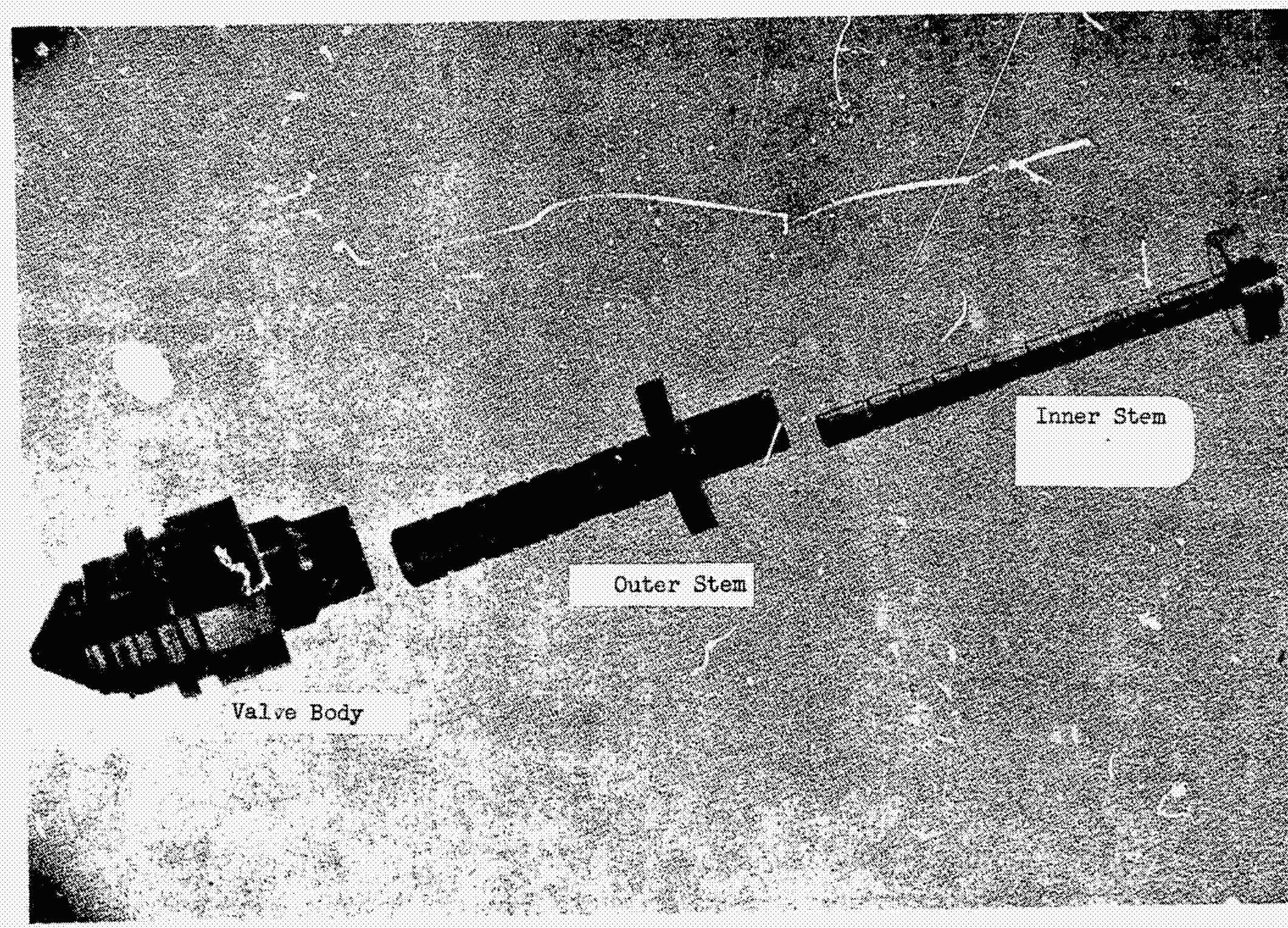


Figure 4.- Poppet valve assembly.

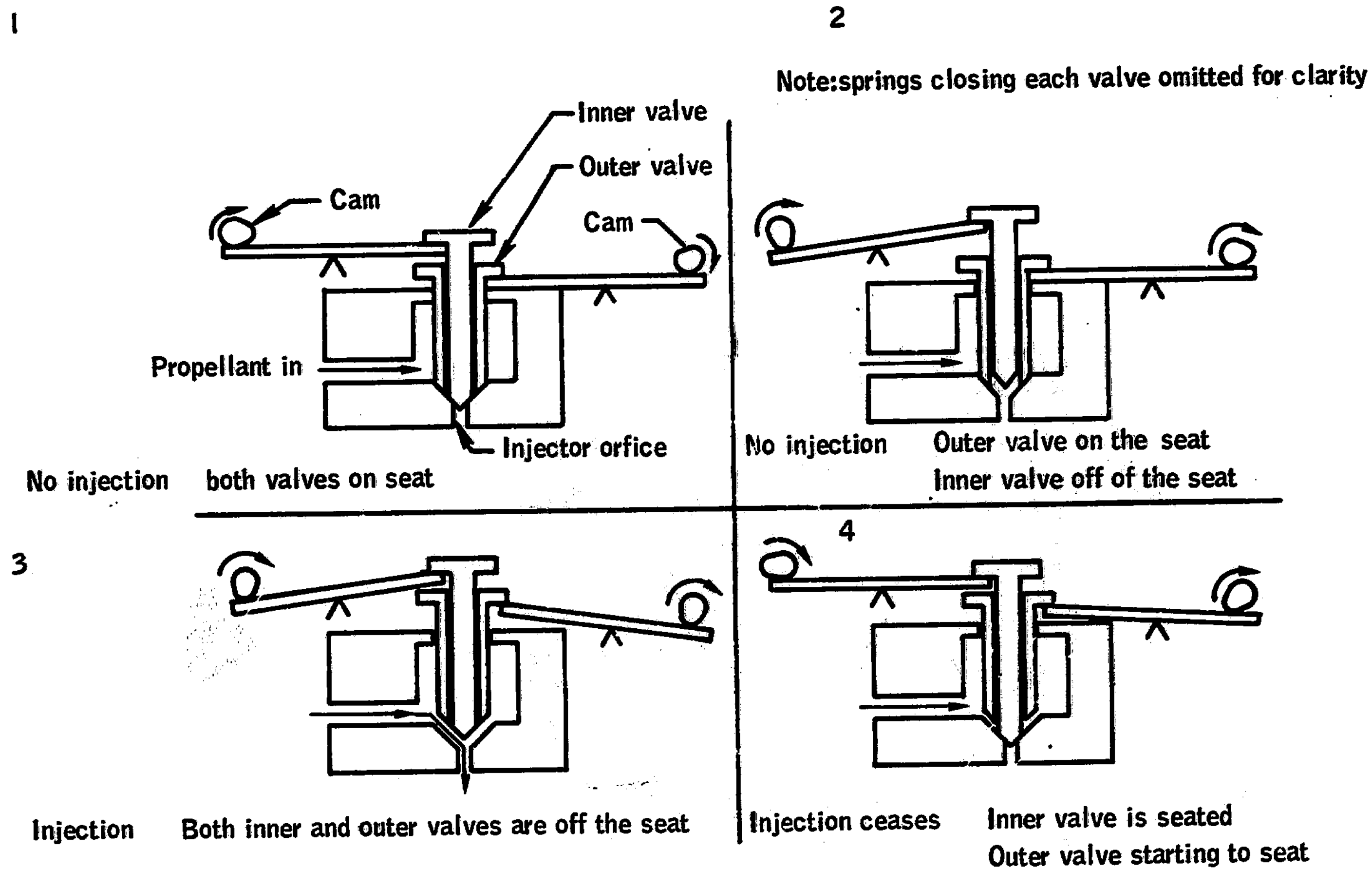


Figure 5.- Schematic-short dwell injection system.

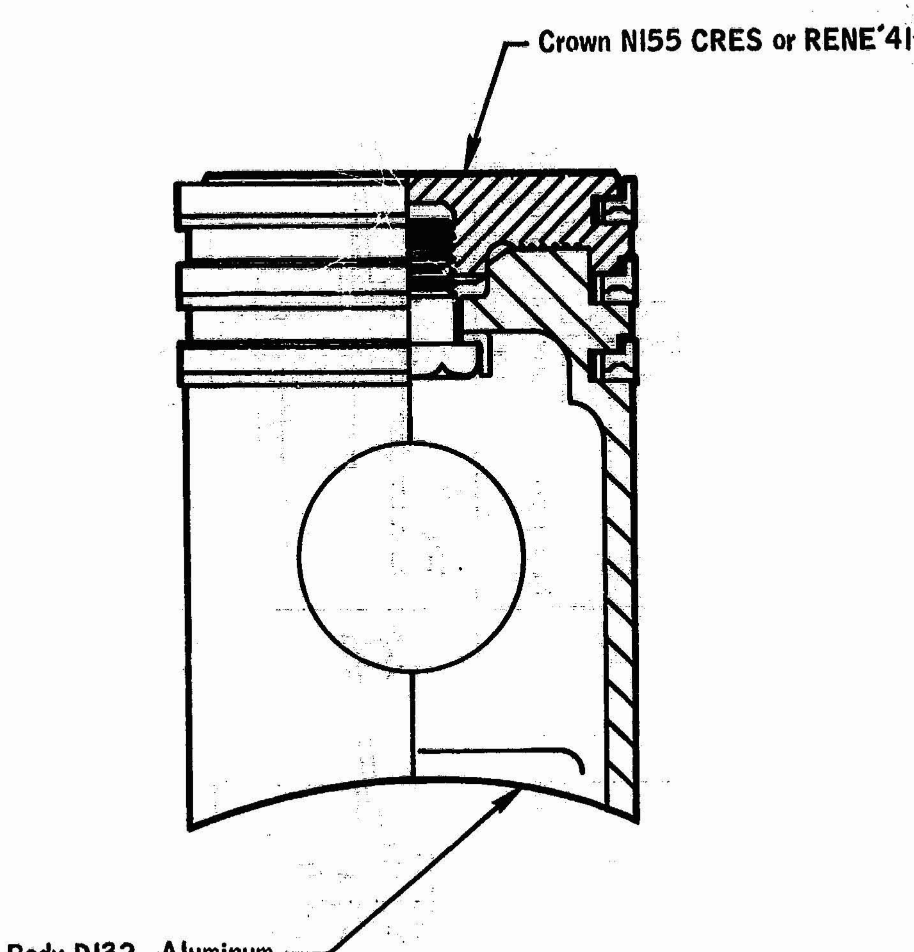


Figure 6.- Composite piston.

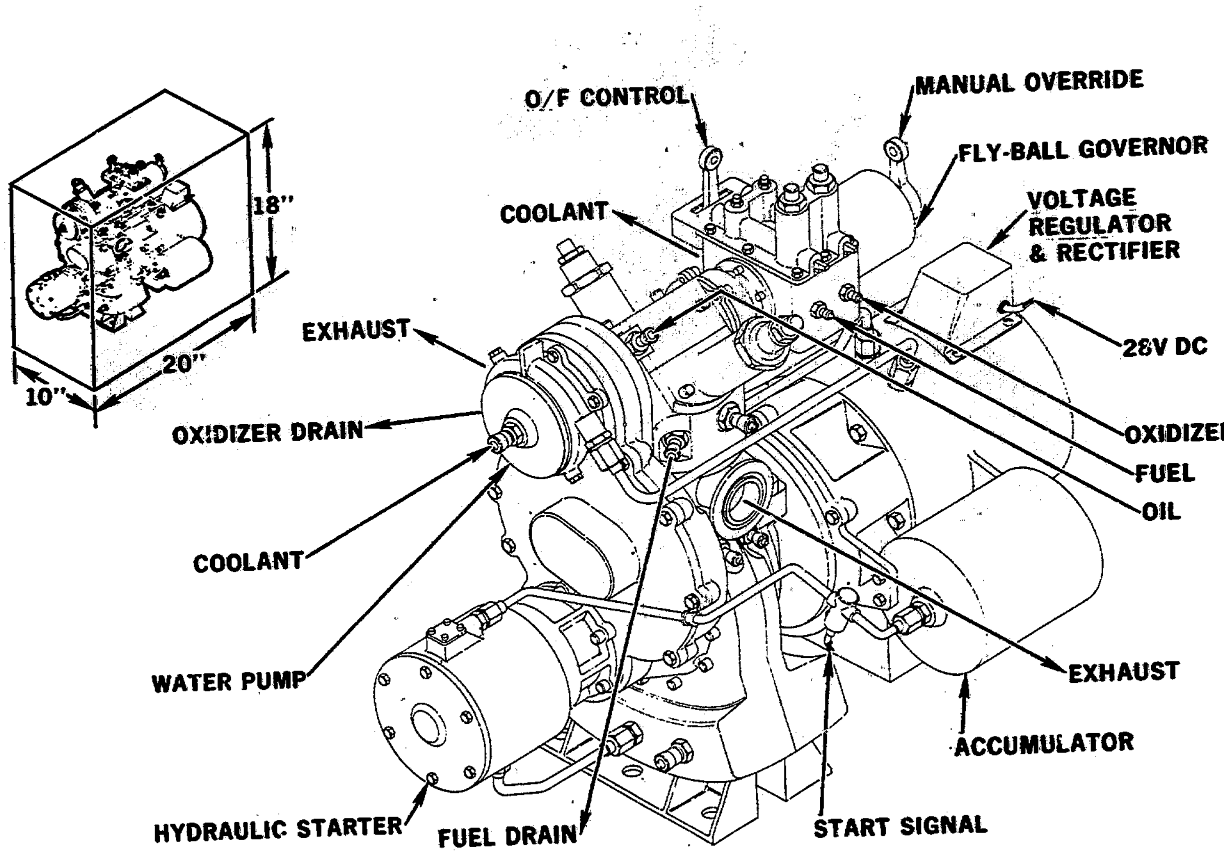


Figure 7.- Hypergolic reciprocator overall system.

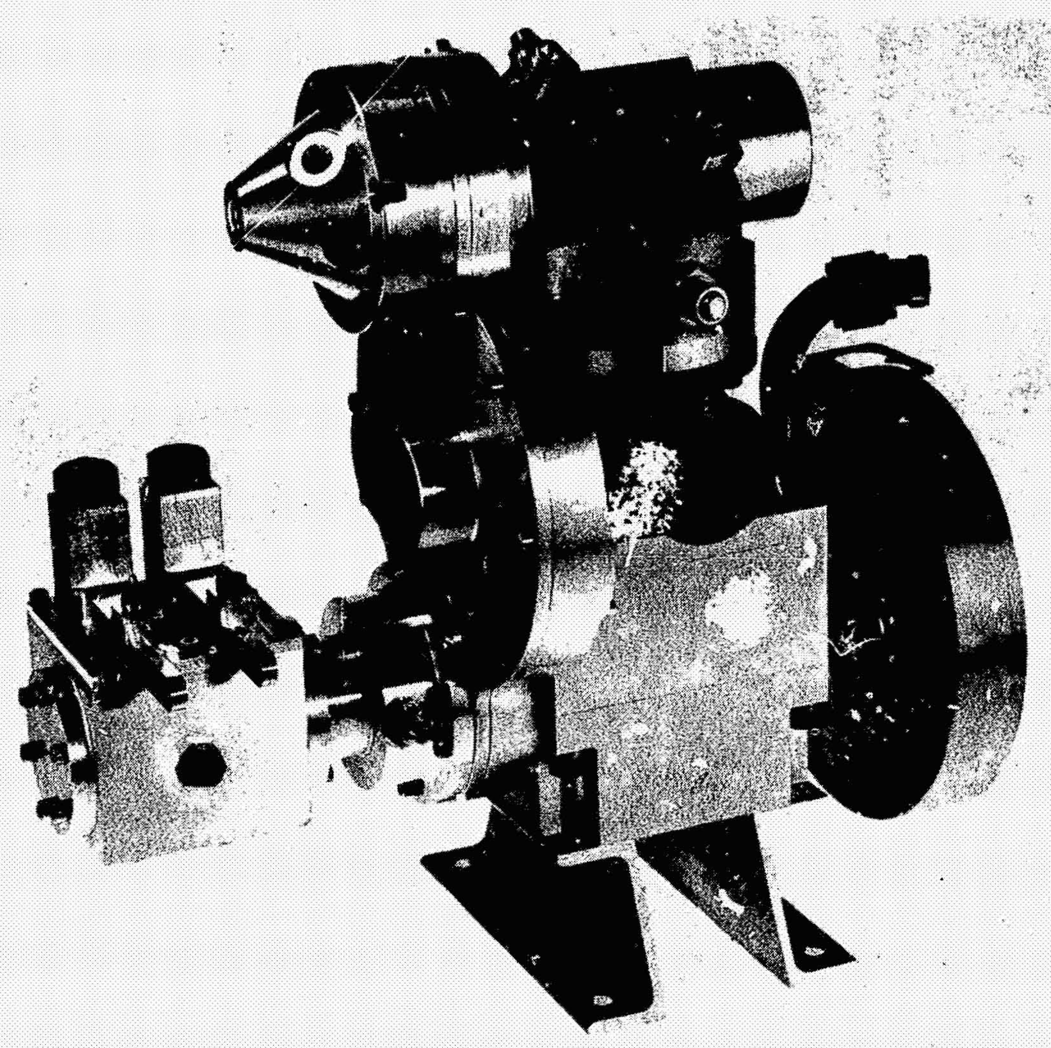
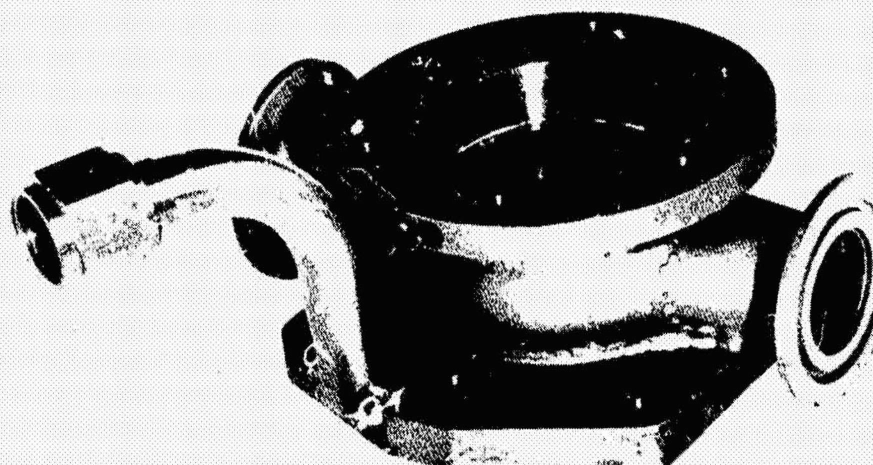
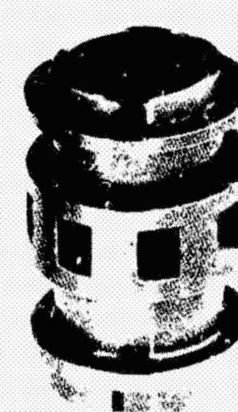


Figure 8 - SPU-3 engine and pump.



Cylinder Body



Cylinder Liner

Figure 9.- SPU-2A-2 cylinder assembly.

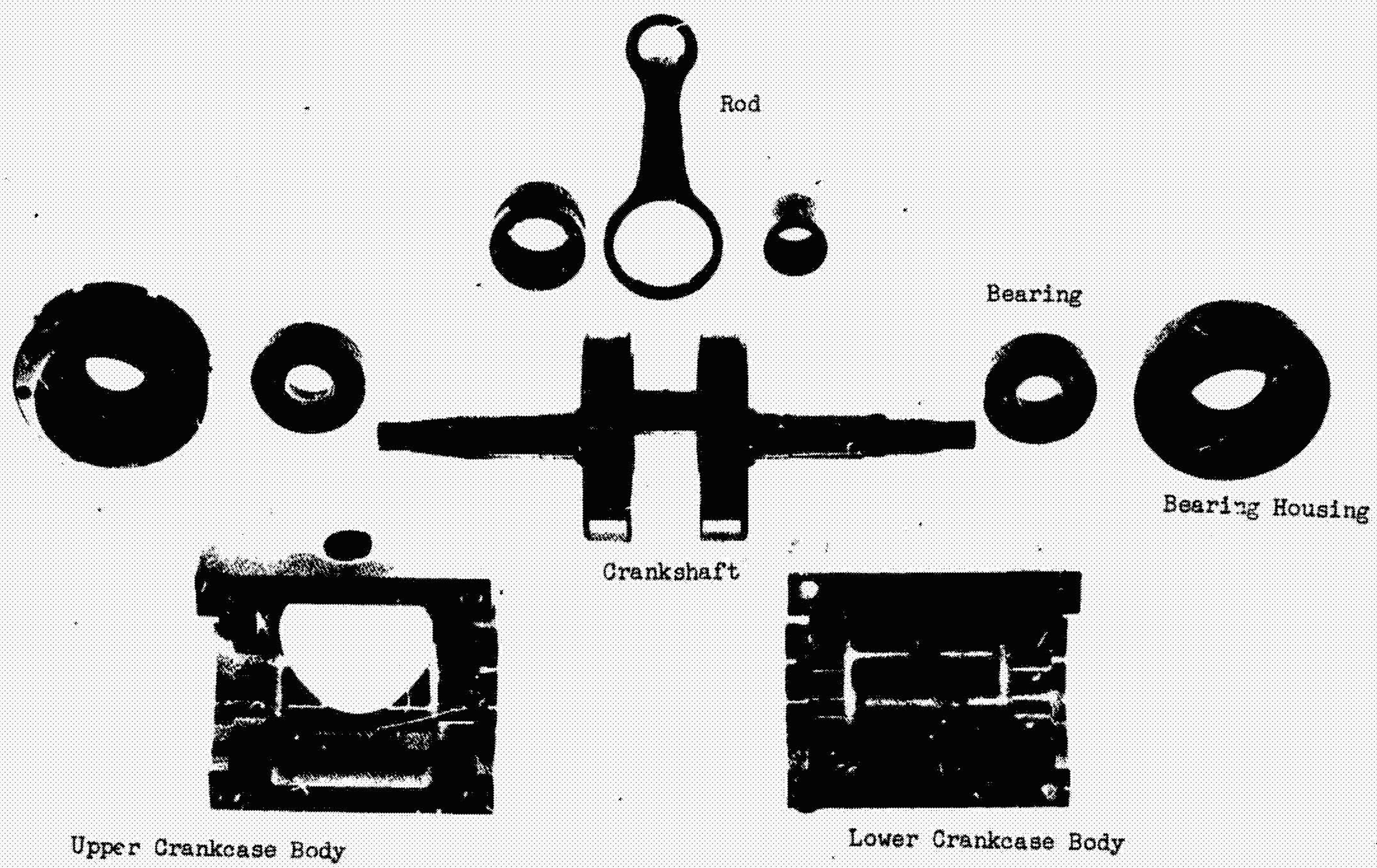


Figure 10.- SPU-2A-2 crankcase assembly.

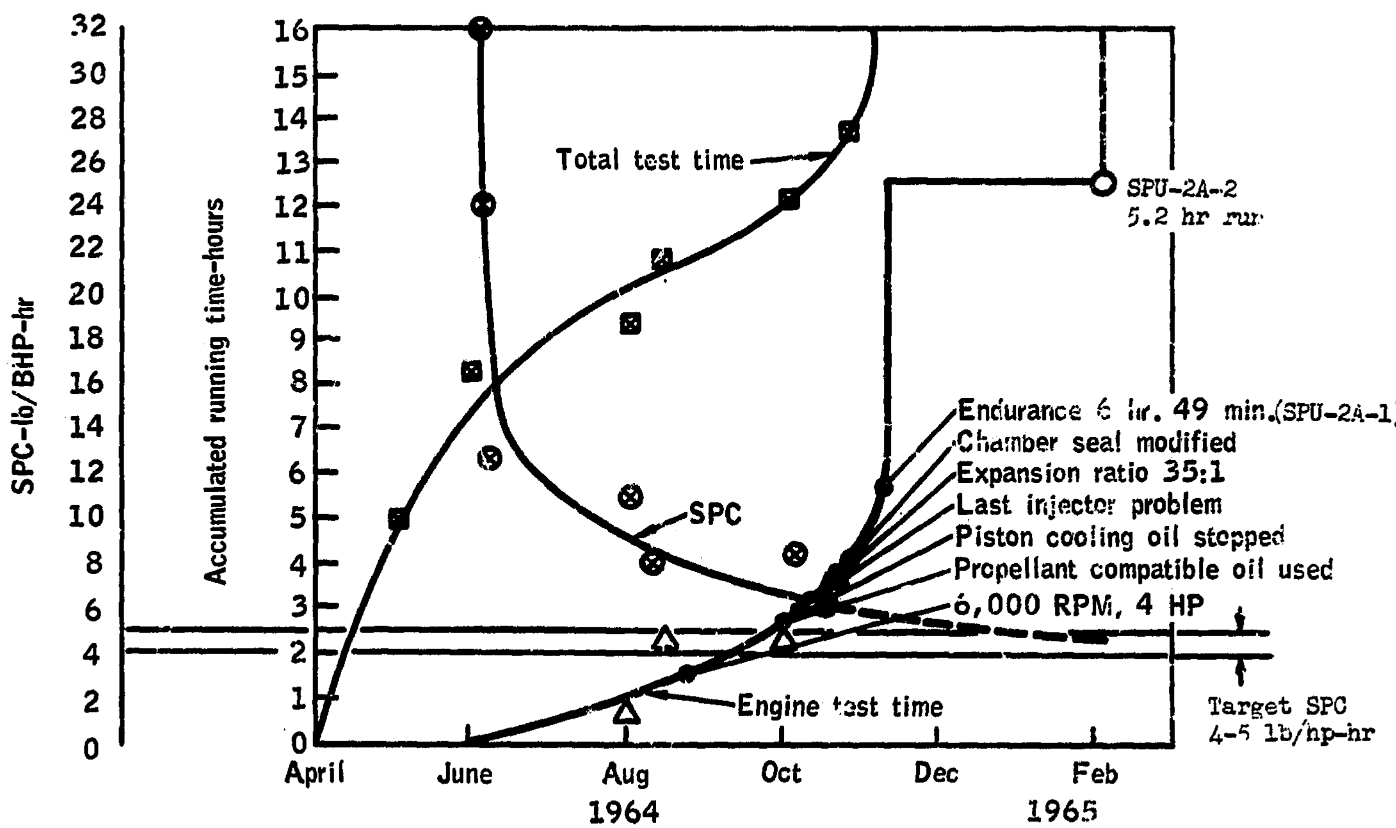


Figure 11.- Engine milestone-history chart - Phase II through Feb. 1965.

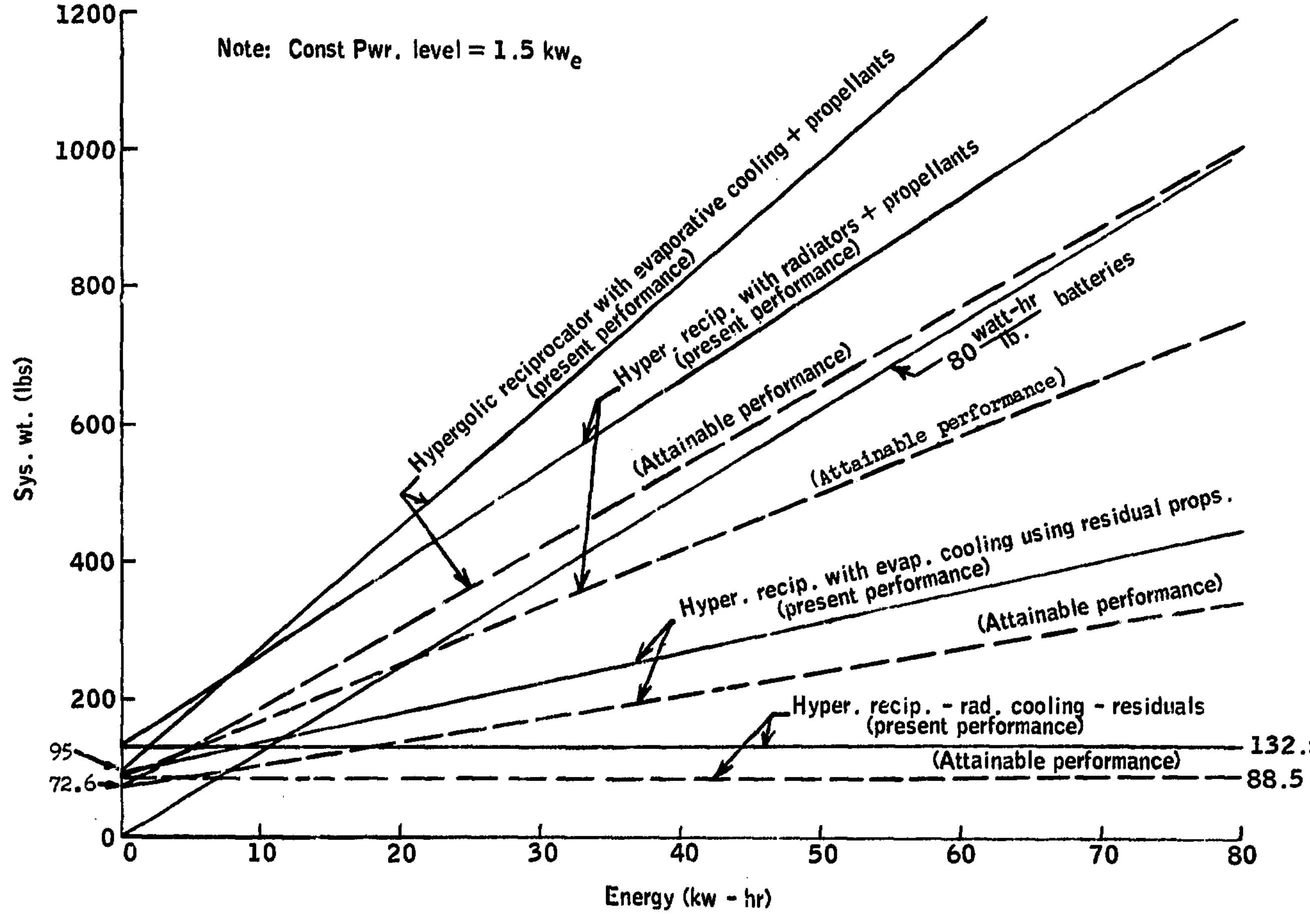


Figure 12.- System weight.

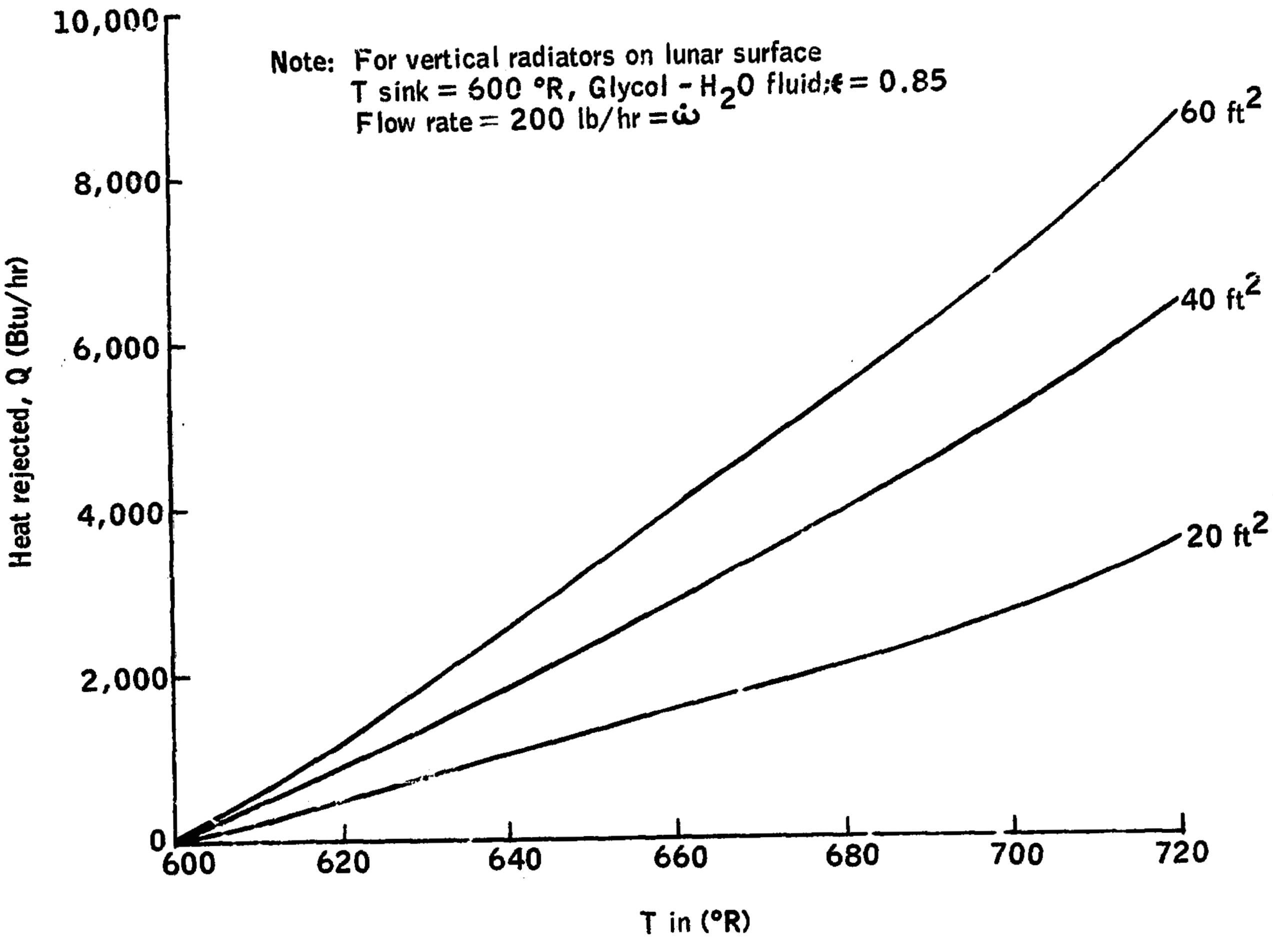


Figure 13.- Radiator area requirements for 200 lb/hr coolant flow.

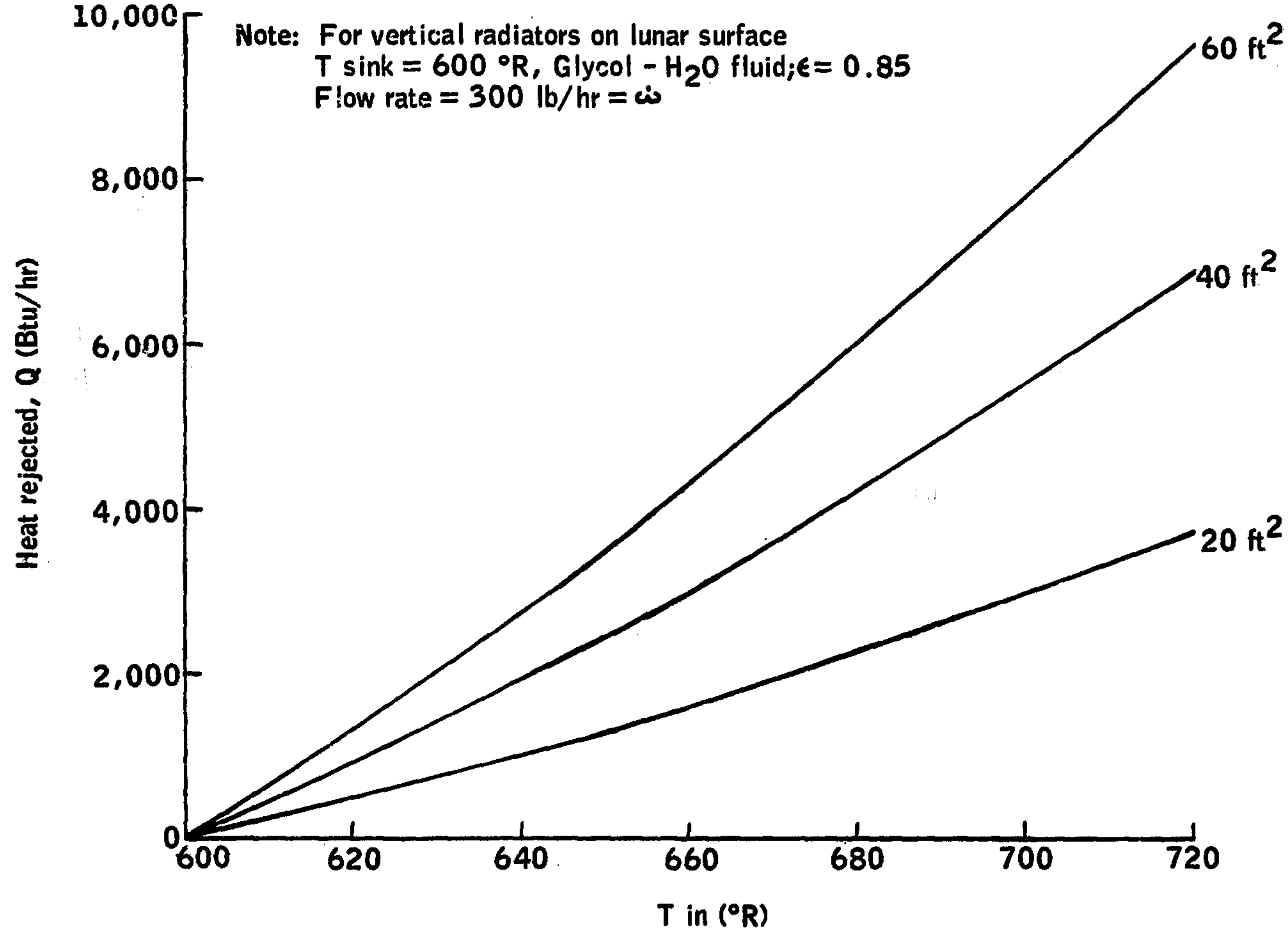


Figure 14.- Radiator area requirements for 300 lb/hr coolant flow.

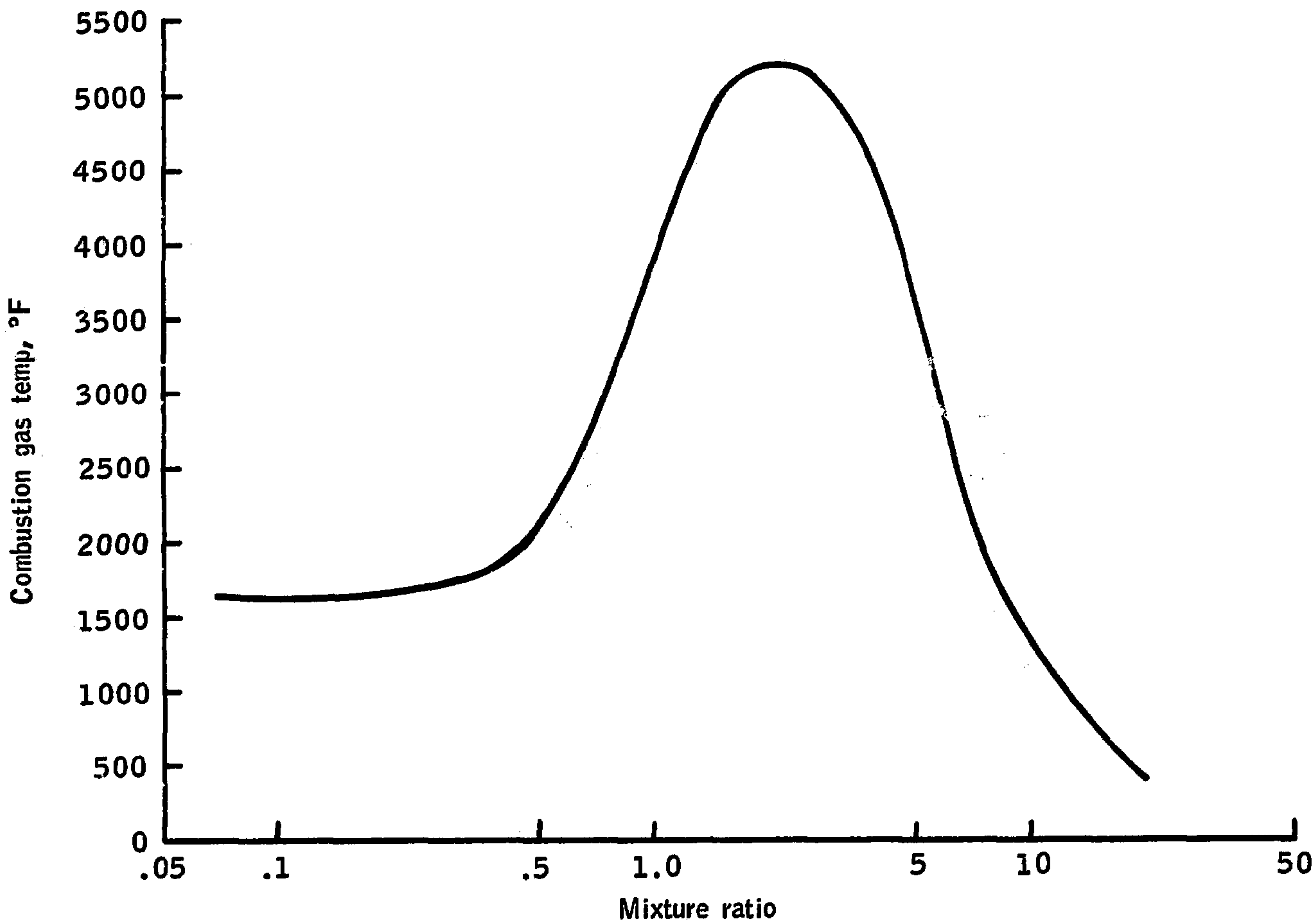


Figure A1.- Combustion gas temp versus mixture ratio (A-50/N_2O_4).

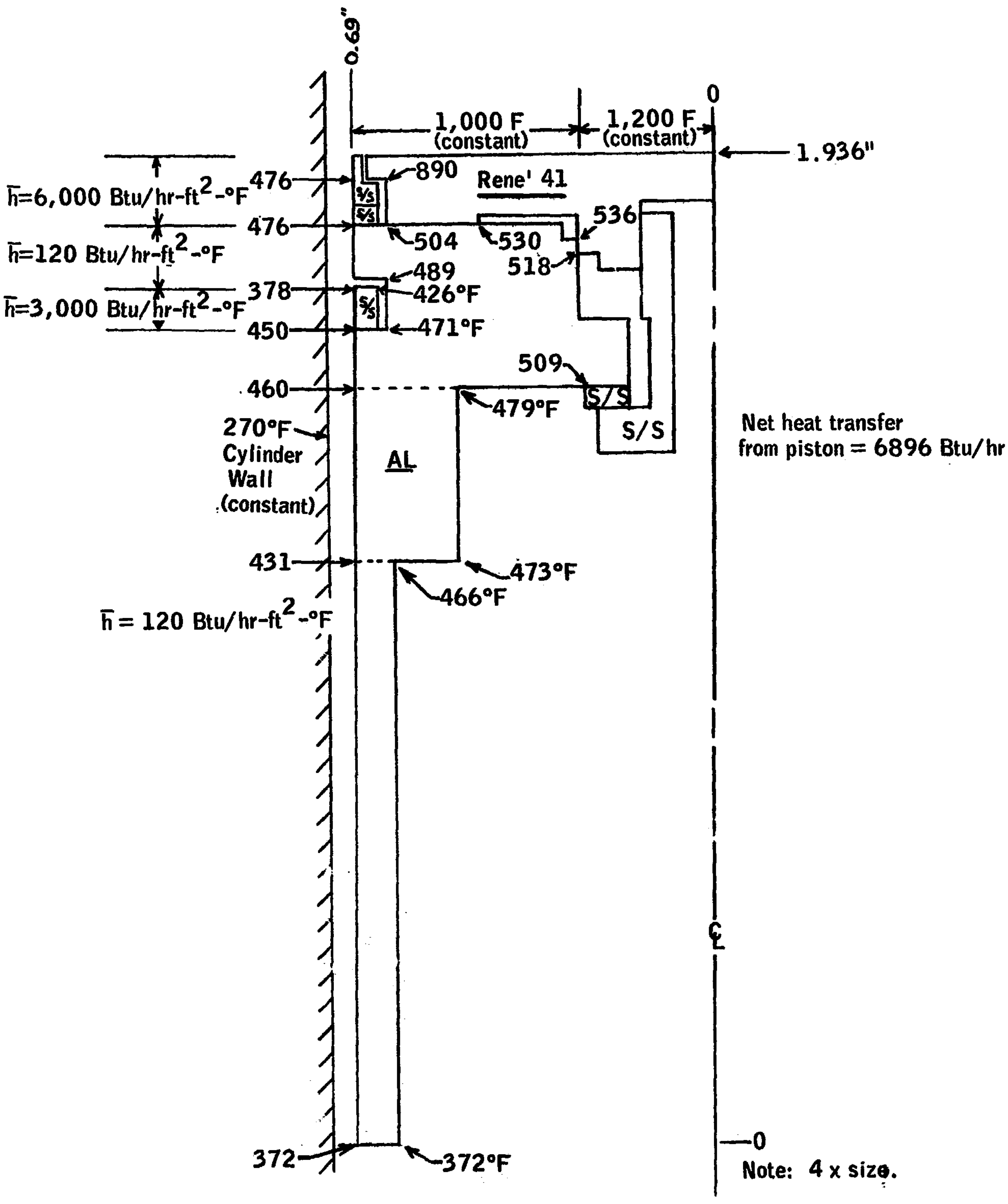


Figure D1.- Baseline piston.

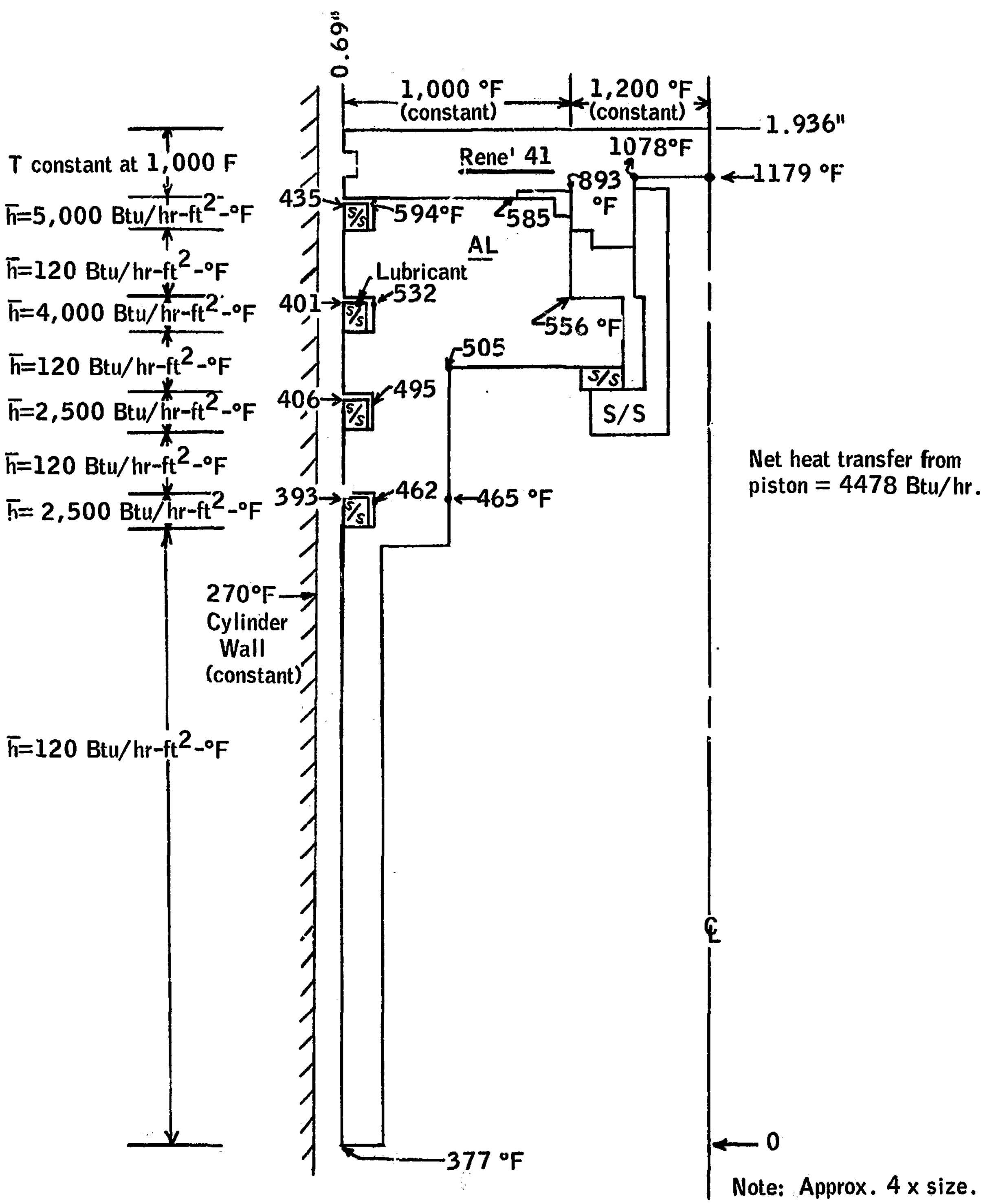


Figure D2.- Number 1 piston.

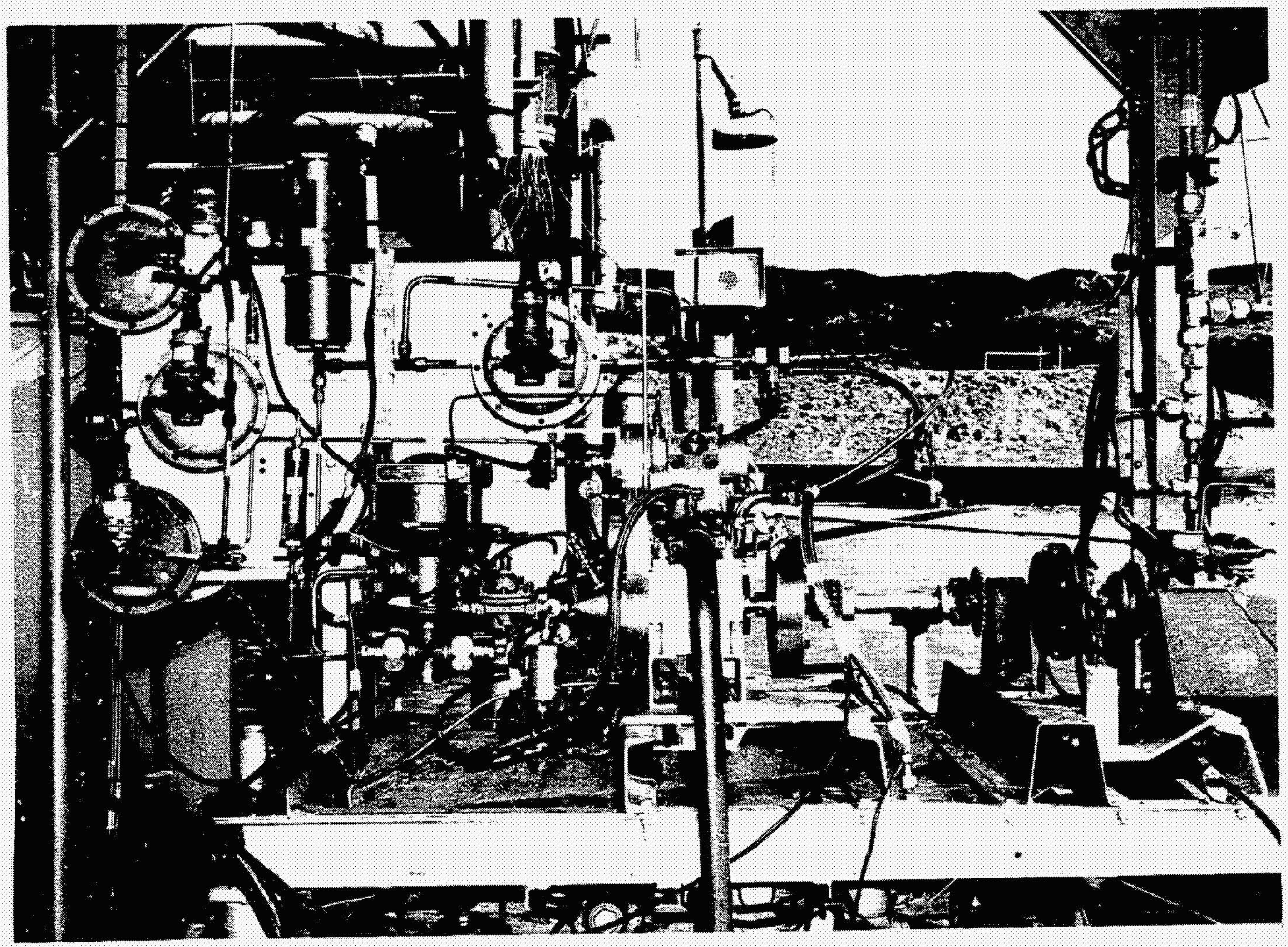


Figure E1.- Engine mounted in test facility.