A DESIGN ANALYSIS OF AN UNDERGRADUATE ROCKET TEST FACILITY

by

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INTRODUCTION

The Mechanical Engineering Department at Kansas State University was awarded a National Science Foundation grant for the purpose of designing and constructing an Undergraduate Rocket Test Facility (UGRTF). A rocket test facility which was used by the Convair Division of the General Dynamics Corporation for a variety of research projects was chosen as a prototype. Design of the Kansas State University Undergraduate Rocket Test Facility has now progressed to the system described and analyzed in this report.

The construction of such a facility posed many inherent hazards. The handling of such potentially explosive materials as gaseous hydrogen and oxygen and liquid RP-1 at high pressures demanded a most careful consideration of design safety requirements. The aim of the design was to reduce the hazards to a minimum and provide a safe and versitile system that would introduce the undergraduate engineer to the basic concepts of rocket engine design and testing.

The acquisition of the Forbes Atlas "E" Missile Squadron Site Number 7 near Wamego, Kansas, provided an ideal location for this rocket test facility. The area adjacent to the site was uninhabited; thus, the facility did not constitute a noise nuisance. The construction of the site would contain the worst conceivable accident which might befall the Undergraduate Rocket Test Facility and limit the damage to the immediate area. Excellent storage facilities existed, and much of the existing equipment at the site was suitable for conversion to the test facility's use.

DESCRIPTION OF SYSTEMS

Propellant Storage and Supply. The propellant storage and supply systems are those systems required to supply RP-1 or gaseous hydrogen (GH_2) and gaseous oxygen (GO_2) to the primary regulation console. The supply systems were located within the missile site underground area. The fuel storage systems were physically separated from the gaseous oxygen (GO_2) system to prevent any possible accidental mixing that might result from spillage while servicing or from rupture of component hardware during servicing or operation.

RP-1 System. A schematic diagram of the RP-1 system is shown in Figure 1 (Drawing UGRTF-01). The RP-1 fuel is stored in a twenty-five gallon high-pressure tank. The tank is equipped with a pressure relief valve and a rupture disk to prevent possible excessive pressurization. By adjusting a remotely controlled regulator in the nitrogen system associated with the fuel system (FN System), gaseous nitrogen is used to pressurize the fuel tank to approximately 2000 psig. Reservicing can be accomplished through a filler cap located in the top of the tank; fuel tank drainage is provided through a hand-operated drain valve located in the bottom of the tank. The RP-1 system is equipped with a filter in the supply line. Flow from the supply tank to the primary regulator is controlled by a solenoid valve which can be remotely operated by a switch on the engine operator's control console.

A sight gage has been incorporated into the RP-1 tank to facilitate the determination of propellant quantity prior to engine operation. The RP-1 tank is pressure and temperature instrumented to provide the data for density computations prior to operation and density monitoring during firing.

Gaseous Oxygen (GO_2) System. A schematic diagram of the gaseous oxygen (GO_2) system is shown in Figure 2 (Drawing UGRTF-04). Gaseous oxygen is procured and stored in bottles connected to a common manifold. The Airco

Figure 1. Schematic Diagram of the RP-1 System.

Figure 2. Schematic Diagram of the GO_2 System.

Company produces a manifold that meets the pressure and compatibility requirements of both the gaseous oxygen and gaseous hydrogen systems. Each manifold has inlet pigtails for twenty bottles with an end connection for the stacking of additional manifolds as necessary. The number of manifolds required was dependent on the desired firing time. For a five-minute-duration firing, only one manifold section is needed for the gaseous oxygen (GO_2) system. The manifold is supported by a floor mounting constructed of angle iron. The bottles are delivered in racks of twenty bottles each; thus, the need for the construction of a structure to contain the bottles has been avoided.

Although the design pressure of the system is 3000 psig and such a supply pressure was desired, local availability of gases limits the supply pressure to 2200 psig. This reduced initial supply pressure increased the number of bottles required with a subsequent increase in operating cost. Based upon the lower initial pressure of 2200 psig, a final pressure of 1100 psig, and a bottle volume of 260 cubic feet, a twenty bottle manifold of gaseous oxygen was specified in order to sustain a five-minute-duration firing. Had the 3000 psig initial pressure been available, thirteen bottles would have provided a sufficient gaseous oxygen supply.

The oxidizer manifold is connected to the main system through a hand-operated shut-off valve and a solenoid-operated shut-off valve. The solenoid valve is controlled by a switch on the engine operator's control console. When opened, the solenoid valve supplies the oxidizer to the primary regulation console.

The GO_2 system has been provided with vent lines to relieve the pressure that would otherwise persist between certain one-way check valves in the system. Cracking pressure for the check valves is 50-65 psig and, unrelieved, this would be sufficient pressure to cause a safety hazard should it become necessary to dismantle the system. The vent line for the manifold has been provided with a hand-operated shut-off valve, since it is unnecessary to vent the manifold between firing sequences. Downstream vent lines are provided with solenoid shut-off valves that can be operated by individual switches on the engine operator's control console.

Gaseous Hydrogen (GH_2) System. A schematic diagram of the gaseous hydrogen (GH_2) system is shown in Figure 3 (Drawing UGRTF-03). The gaseous hydrogen storage and supply system is similar to the gaseous oxygen system. The design pressure was 3000 psig but gaseous hydrogen is locally available only in 2000 psig, 195 cubic feet bottles. At the 2000 psig initial pressure and an 1100 psig final pressure, sixty bottles are required to sustain a five-minute-duration firing. Therefore, three Airco manifolds were required to form the gaseous hydrogen supply manifold for a five-minute-duration firing of a 160 pound thrust GH_2/GO_2 rocket engine. The manifold for the GH_2 system was floor-mounted in a manner similar to that employed for the GO_2 system.

Control and venting of the GH_2 system is also very similar to that incorporated into the design of the gaseous oxygen system.

Propellant Lines. The sizes of the propellant lines are given on the drawings for each system (see Figures 1, 2, and 3, respectively). All tubing selected was stainless steel with a wall thickness sufficient to give a safety factor of six for a design operating pressure of 3000 psig.

The tube fittings selected were standard 37° flare fittings throughout the design. The system was designed using straight lengths of tubing only. If, during construction, tube lengths are deemed sufficient, elbows shown on the blueprints may be omitted and appropriate tube bends may be

Figure 3. Schematic Diagram of the GH_2 System.

substituted.

Nitrogen (GN_2) Systems. Bottled gaseous nitrogen at 3000 psig was selected for use as the purging agent for the propellant systems and the control gas for the dome regulators. Two independent gaseous nitrogen (GN_2) systems were designed, one for the GO_2 system and one for the fuel systems. Two systems were used to preclude the possibility of mixing a fuel with the oxidizer through a common purge system. This was in keeping with the original safety requirements of complete isolation of the fuel and oxygen systems upstream of the injector face.

Fuel-Nitrogen (FN) System. A schematic diagram of the fuel-nitrogen (FN) system is shown in Figure 4 (Drawing UGRTF-02). As with oxygen and hydrogen, bottles of nitrogen were connected to a common manifold manufactured by Airco Company. A manually operated valve was incorporated to furnish a flow of gaseous nitrogen from the supply bottles to a three-way solenoid-operated shut-off valve. The normally open port of the shut-off valve was connected to a vent line. The "KEY" switch, located on the engine operator's control console, was connected so as to energize this valve and make nitrogen pressure available to the remotely controlled pressure regulators whenever power is turned on. This design feature was used to insure the availability of a supply of purging agent at all times that the facility was in use.

Nitrogen pressure was directed through a remotely controlled regulator to the RP-1 storage tank to produce a pressure differential of sufficient magnitude to supply the fuel to the rocket engine at the specified pressure. The nitrogen-filled void in the RP-1 tank serves the secondary function of providing an inert blanket over the RP-1 fluid. As a consequence, the formation of a combustible mixture inside the tank has been prevented.

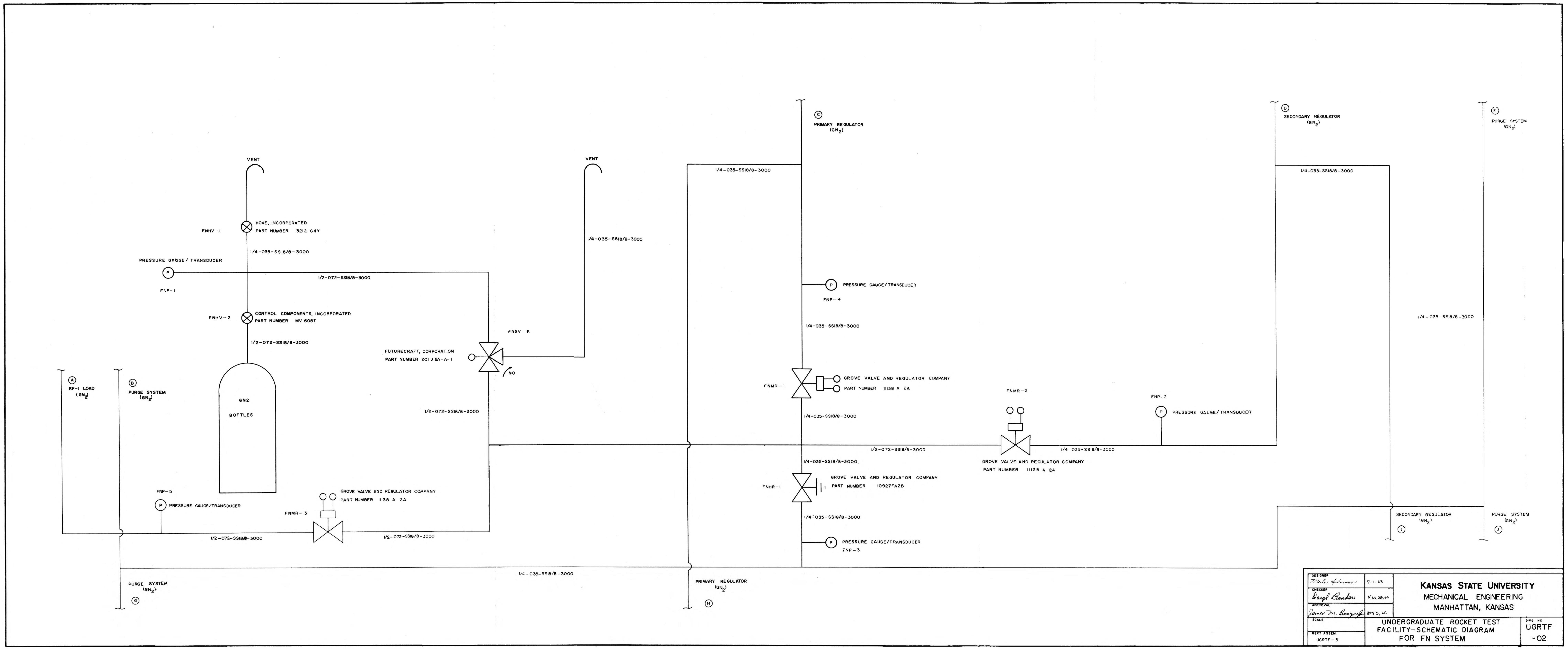

Figure 4. Schematic Diagram of the FN System.

Nitrogen was routed through a hand regulator, FNHR-1, to provide a supply of purging agent to two locations in the GH_2 system and two locations in the RP-1 system. Introduction of the purging agent at the upstream location in either system allows a complete purge of that system. Solenoid valves FNSV-2 in the RP-1 system and FNSV-4 in the gaseous hydrogen system are operated by the "FUEL PURGE" switch located on the engine operator's control console. The valves allow purging nitrogen to enter the fuel system upstream from the primary regulator, flow through the fuel lines, including the gaseous hydrogen pilot lines, and be expelled through the engine nozzle. This fuel purge mode was designed to clear all fuel or contaminants from the lines and is normally used before operation of the propellant system and after final shut-down.

The second possible route for the purging agent in either fuel system was provided to enable the purging of the rocket motor only. It furnishes a supply of nitrogen to a location just upstream from the motor. Solenoid valves FNSV-3 for the RP-1 system, or FNSV-5 for the GH_2 system are operated by the "MODEL PURGE" switch on the engine operator's control console. These valves allow nitrogen to enter the fuel lines and pass through the engine. This model purge mode was designed to clear all gases from the motor for brief inspections between firings.

A full system purge can also be initiated by the "PANIC" switch. As the name implies, this mode of operation was designed to be used in an emergency situation when it becomes urgent to discontinue operation of the facility. Upon closing the "PANIC" switch (located on the engine operator's control console) all propellant flow is shut off and the system automatically performs a complete purge of the propellant lines.

Nitrogen was used as the control gas for both the primary and secondary dome regulators of each system. Remote regulator FNMR-1 was employed to set the nitrogen pressure for the RP-1 and GH_2 primary regulators, while remote regulator FNMR-2 was employed to set the nitrogen pressure for the RP-1 and GH_2 secondary regulators.

Oxygen-Nitrogen (ON) System. A schematic diagram of the oxygen-nitrogen (ON) system is shown in Figure 5 (Drawing UGRTF-5). The operation and design of the oxygen-nitrogen system was identical to that of the fuel-nitrogen system with the following exception: The oxygen-nitrogen system was designed to support one oxidizer system only, whereas the fuel-nitrogen system was designed to support both the RP-1 and the GH_2 fuel systems. Thus, the lines and components used to pressurize the RP-1 fuel tank in the case of the fuel-nitrogen system have no counterpart in the oxygen-nitrogen system.

Metering and Regulation. The pressure regulation of all propellants was accomplished in two stages. The two-stage design permitted the use of higher storage pressures than would have been possible had a single pressure-regulating stage been employed.

Primary Regulation. The pressure of the propellants at the inlet of the primary regulating console is approximately the same as the pressure available in the storage system. The primary regulation console was designed to contain all of the necessary hardware for primary pressure regulation and is shown in Figure 6 (Drawing UGRTF-09). At the primary regulation console, the supply pressure is reduced to 1000 psig by remotely controlled dome regulators H2DR-1, RPDR-1, and O2DR-1. Nitrogen gas is used to load the domes of the regulators to the desired pressure as described under Nitrogen Systems of this report.

Metering and Secondary Regulation. The secondary regulation console

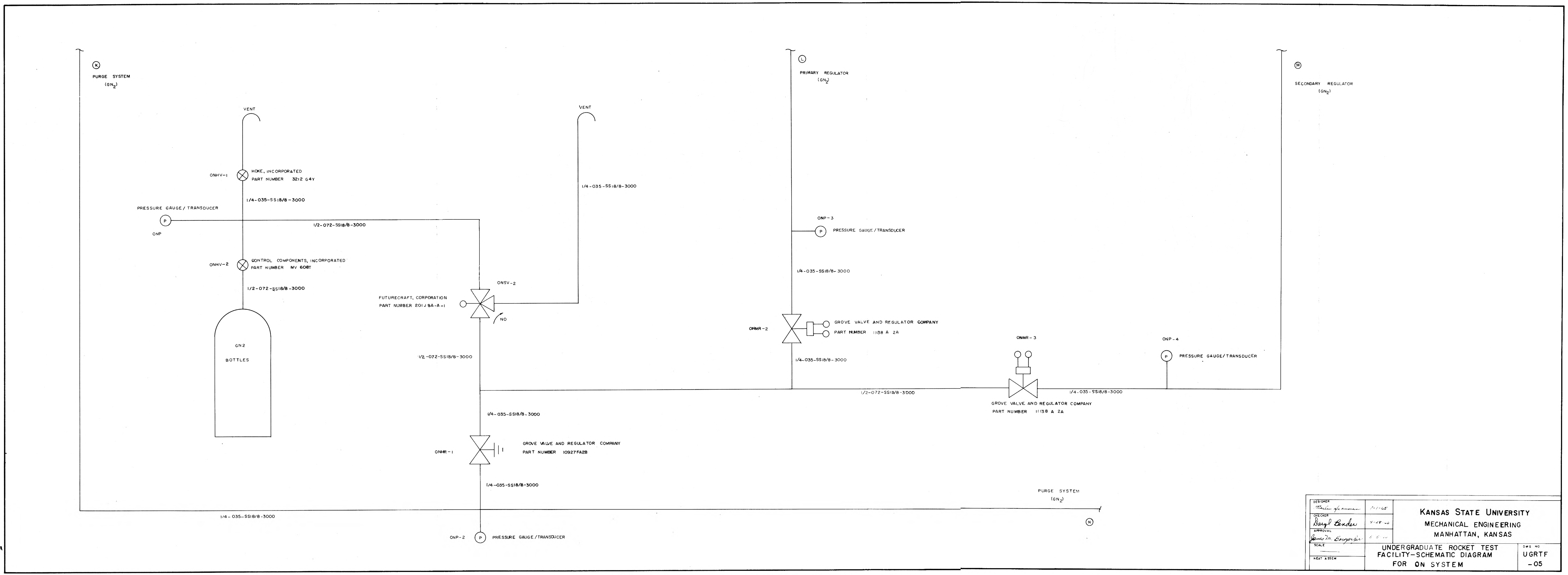

Figure 5. Schematic Diagram of the ON System.

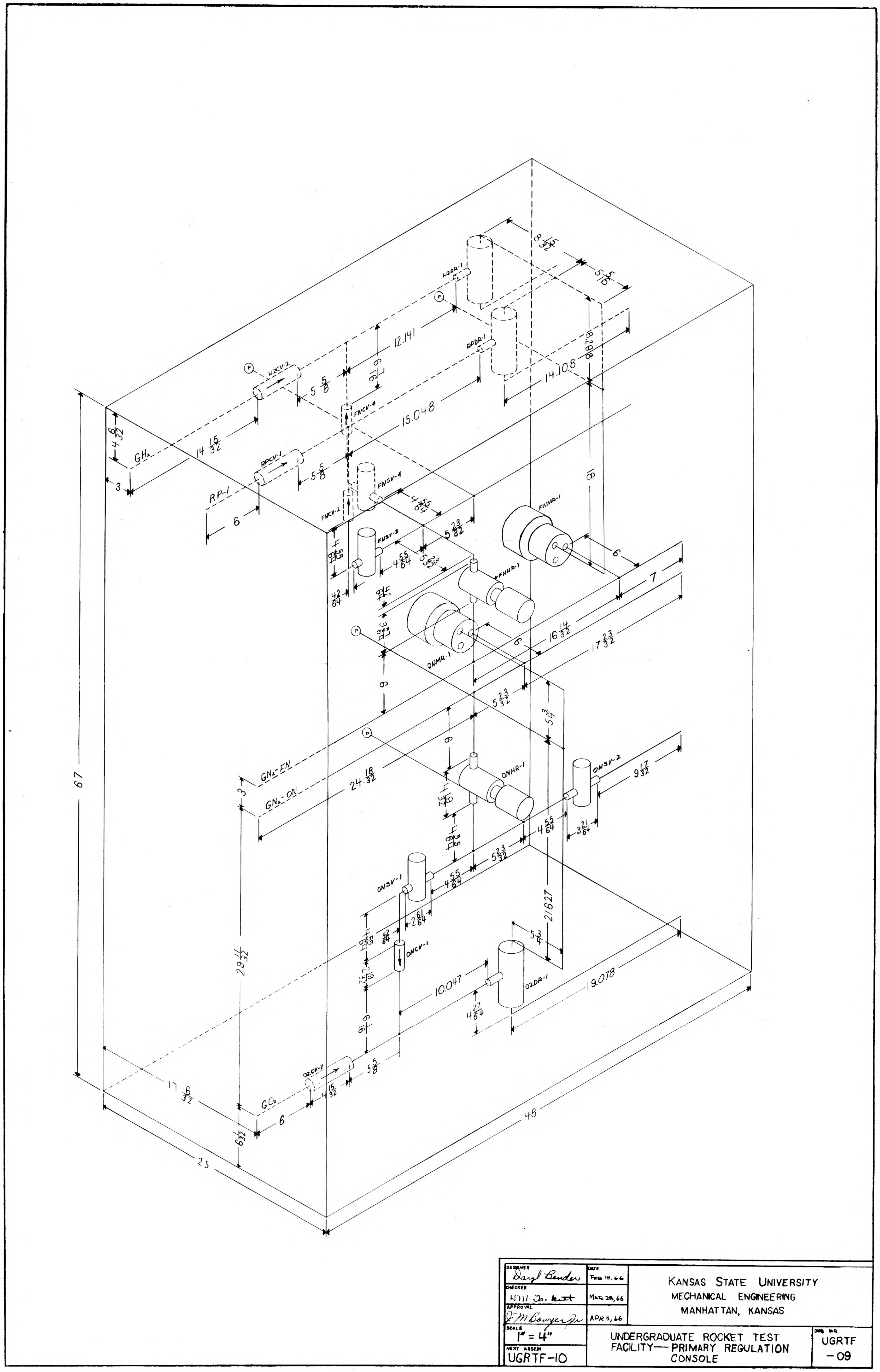

Figure 6. Isometric View of the Primary Regulation Console.

shown in Figure 7 (Drawing UGRTF-10), was composed of the necessary hardware and lines for propellant metering and secondary regulation as well as the valves and lines for the pilot system and the engine shut-off valves.

Metering of all propellants is accomplished by sharp-edged orfice assemblies. In the case of each propellant the pressure and temperature of the propellant are measured at the inlet to the orfice meter and used in conjunction with the measured pressure drop across the orfice to compute the mass flow rate of that propellant. The equation used to compute propellant flow rate is:

$$\dot{w} = C_d A \sqrt{2g\rho\,\Delta P}$$

where:

$\dot{w}$ = mass flow rate

C_d = calibrated discharge coefficient

A = throat area of the orfice plate

g = gravitational acceleration constant

ρ = density of the propellant

ΔP = differential pressure across the orfice plate.

Orfice meters available from Daniel Orfice Fitting Company were judged satisfactory for all three propellant systems. It was calculated that the orfice plate bore required was 0.4446 inches for GH_2, 0.4794 inches for GO_2, and 0.2537 inches for RP-1.

Secondary regulation was accomplished by employing remotely controlled dome regulators, using nitrogen as the control gas. The required pressure on a propellant as it leaves the secondary regulator is determined by the injection pressure required to maintain a design chamber pressure in the rocket motor at a given oxidizer-to-fuel (O/F) ratio. For design considerations a secondary regulation pressure of 700 psig was used in all cases.

<u>Coolant System</u>. A supply of water at or above a pressure of 500 psig and

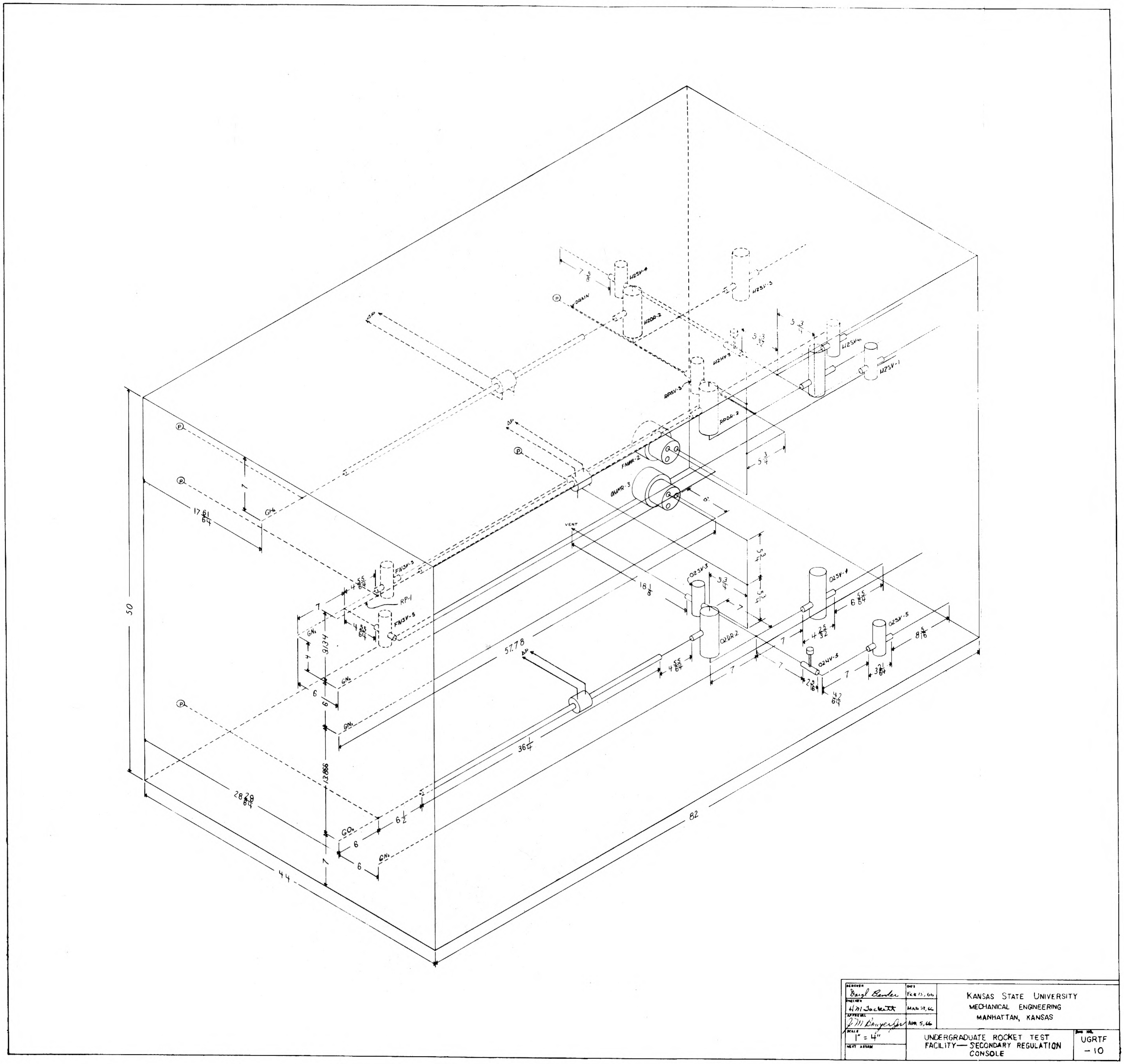

Figure 7. Isometric View of the Secondary Regulation Console.

at a flow rate of six pounds per second was needed for the engine cooling system. The design pressure in the coolant jacket was then greater than the design pressure in the combustion chamber of the motor. Such a pressure differential can serve to reduce the effects of a motor liner burn-out during a firing sequence by injecting water into the motor. At the same time, it would make the malfunction more noticeable and prevent combustible gas products from entering the water system.

A schematic diagram of the cooling system is shown in Figure 8 (Drawing UGRTF-11). The water is routed through a filter having a filtration rating of ten microns. A hand-operated shut-off valve was chosen to regulate the flow of water. To prevent low pressure operation, a pressure switch was located in the coolant inlet line of the motor. Sufficient pressure to actuate the pressure switch is required to provide electrical continuity of the firing circuits; without sufficient coolant supply pressure, the engine cannot be fired. High pressure flexible tubing was selected to serve as a connection between permanent coolant lines and the rocket motor. This design feature facilitates the assembly and disassembly of the motor. Thermocouples attached directly to the coolant supply lines of the motor were used to sense inlet and outlet water temperatures. To prevent operation in an overheated condition, a permissive switch was incorporated into a strip chart recorder which is used to monitor the inlet and outlet water temperatures. If the temperature difference should exceed a predetermined maximum, the permissive switch automatically would terminate the firing.

Engine Operator's Control Console. The schematic diagram for rocket engine control is shown in Figure 9 (UGRTF-06). The control system has been designed to regulate propellant flow, implement ignition, provide firing control and accomplish facility shut-down. Power to the console is furnished by

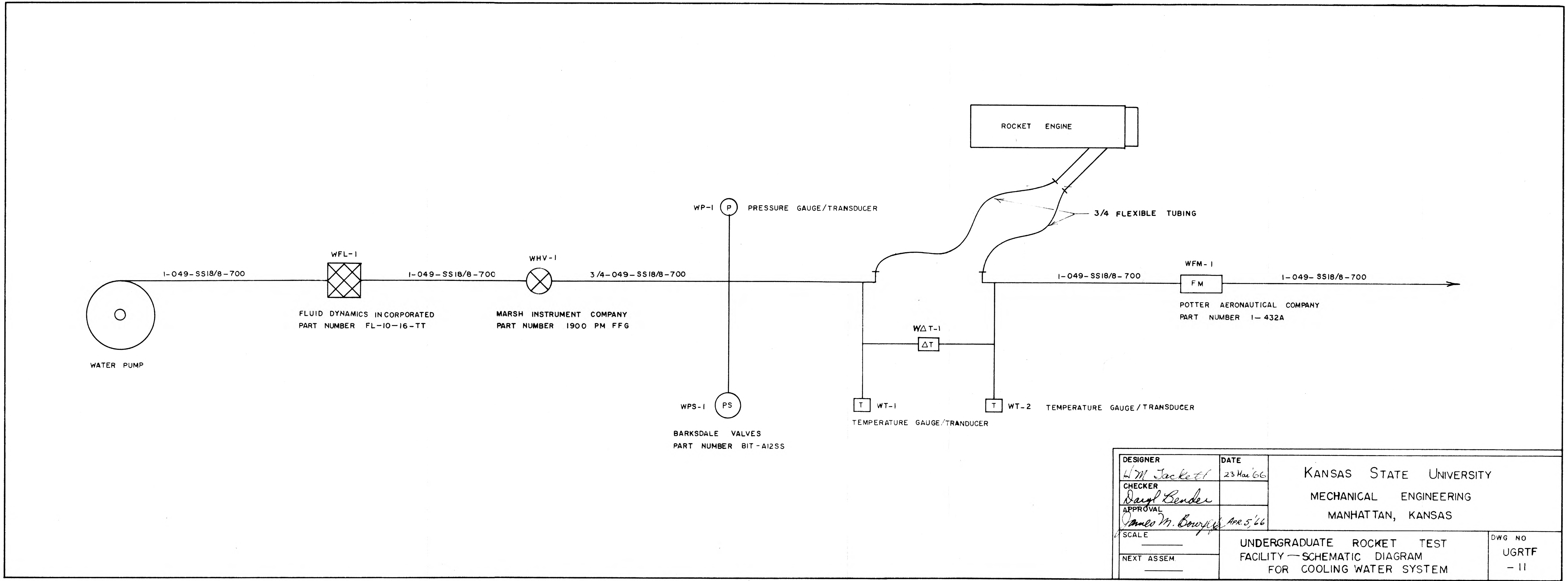

Figure 8. Schematic Diagram of the Coolant System.

Figure 9. Schematic Diagram of the Control System.

two 28v. DC buses protected by 15 amp fuses.

A "KEY" switch energizes both buses 1 and 2 through relays RA and RB, respectively. Relays RA and RB also energize the two three-way solenoid valves ONSV-2 and FNSV-6 to close off the vent lines and route nitrogen to the upstream side of each nitrogen loader. This makes nitrogen immediately available for purge and/or control.

A multiply ganged wafer switch is used to select the desired fuel system. A stacked wafer switch was incorporated to prevent inconsistent fuel system selection. The switch consists of four tiers, W1, W2, W3, and W4.

W1 selects the parts of the system to be purged if the model purge is activated. With switch W1 in the RP-1 position, relay R7 energizes solenoid valves FNSV-3 and ONSV-2. As a consequence, purge nitrogen is directed through the model via the RP-1 and GO_2 lines. With switch W1 in the H2 position, relay R8 energizes solenoid valves FNSV-5 and ONSV-2. Thus, nitrogen purge of the model is directed through the GH_2 and GO_2 lines.

Switch W2 selects the fuel system which is to be activated by the "FIRE" switch. With the switch in the RP-1 position, solenoid valve RPSV-4 is activated by the "FIRE" switch. Valve RPSV-4 allows RP-1 to enter the rocket motor. With the switch in the H2 position, solenoid valve H2SV-5 is actuated by the "FIRE" switch. Valve H2SV-5 allows GH_2 to enter the rocket motor. Control can be taken away from W2 by the activation of either relays R2 or R4 which are part of the "PANIC" control system.

Switch W3 selects the route of flow of GH_2 and GO_2 used for the pilot system. With switch W3 in the RP-1 position, the "PILOT FIRE" switch, SW15, energizes relay R10. Relay R10 supplies power to solenoid valves H2SV-1 and O2SV-5 which provide pilot propellants to the RP-1/GO_2 injector. With switch W3 in the H2 position, relay R11 is energized by closing the

"PILOT FIRE" switch. Relay R11 supplies power to solenoid valves H2SV-6 and O2SV-5 which provide pilot propellants to the GH_2/GO_2 injector. When either the RP-1 or the H2 mode is selected with switch W3, relay R16 is energized by the "PILOT FIRE" switch. Relay R16 supplies power to a latching relay. The latching relay provides continuity to the "FIRE" circuit and prevents the advancement of the system to the full thrust condition without first obtaining pilot ignition. The "PILOT FIRE" switch, SW15, supplies power directly to the ignition circuit.

Switch W4 selects the fuel system that is affected by activation of either the "PANIC" switch, SW2, or the "COMPLETE PURGE" switch, SW18. Its function will be explained more completely under the discussion of the "COMPLETE PURGE" switch.

The engine coolant system is activated by the "H20 PRESSURE" switch SW4. Relay R9 is used as a remote switch for power supply to the water pump.

The "H20 LIMIT" switch is controlled by three parameters: one from the water pressure switch, WPS-1; one from the water differential temperature switch, W ΔT-1; and one from the water flowmeter, WFM-1. All of the above parameters must be within prescribed limits to maintain electrical continuity to the "H20 LIMIT" switch circuit. When the switch is closed, the "H20 LIMIT" circuit enables the "PILOT FIRE" and "FIRE" switches. All firing operations are interrupted by the "H20 LIMIT" switch whenever any cooling water limit is exceeded.

For the purpose of system versitility, nitrogen purge is controlled by three modes of operation. Switch SW18 was included to allow initiation of a complete purge of the system, but it can also be used to purge either the fuel or the gaseous oxygen systems independently. The systems to be purged are selected by positioning the applicable enabling switches SW1 and/or SW3.

With the enabling switches closed, relays R1 and R6 are activated when the "COMPLETE PURGE" switch, SW18, is closed. The energized purge control relays, R1 and R6, interrupt the electrical continuity of the solenoid circuits associated with the normally closed propellant supply valves, O2SV-1, RPSV-1, and H2SV-2. Depending on the fuel system selected, relays R2 and R3 or relay R4 is activated when the "COMPLETE PURGE" switch, SW18, is closed. Relays R2 and R3 supply power to open solenoid valves RPSV-4, FNSV-2, FNSV-4 and H2SV-1. With the four valves open, purge nitrogen is directed through the RP-1 supply lines and the GH_2 pilot lines for the RP-1 fuel system. Relay R4 supplies power to open solenoid valves H2SV-5, H2SV-6 and FNSV-4. With the above valves open, purge nitrogen is directed through the GH_2 supply lines and the pilot lines for the GH_2 system. Relay R6 supplies power to solenoid valves O2SV-5, O2SV-4 and ONSV-1. With these three valves open, purge nitrogen is directed through the GO_2 supply and pilot lines.

The "PANIC" switch, SW2, was incorporated to provide a means of emergency shutdown of the rocket test system. By closing switch SW2, the enabling switches of the purge system can be by-passed and purge relays R1 and R6 can be energized through relay R5. The system in use at the time then will undergo a complete purge as described above. Activation of relay R5 also opens a normally closed relay in the electrical supply line to the "FIRE" switch, SW5. Upon loss of continuity, the latching relay R16, opens and thereby disables the fire circuit.

The system can also be advanced to the emergency shut-down mode if the chamber pressure, P_c, exceeds the prescribed upper or lower pressure limits. Limit switches were incorporated into the chamber pressure lines and these supply the activation power for relay R13. Relay R13 was used to supply electrical continuity to by-pass the "PANIC" switch. To allow start-up and the initial low-chamber-pressure mode of rocket engine operation, the chamber pressure lower limit switch is shunted at this stage of rocket engine operation. Activation of the chamber pressure lower limit circuit is established whenever the chamber pressure first exceeds the lower pressure limit. The chamber pressure lower limit switch is also connected through the "FIRE" switch, SW5, so that the system cannot proceed through the emergency shut-down sequence under a normal shut-down procedure.

The propellant supply valves O2SV-1, RPSV-1 and H2SV-2 are manually operated by switches SW6, SW7, and SW8, respectively. Line vent and drain valves O2SV-2, O2SV-3, RPSV-2, RPSV-3, H2SV-3 and FNSV-1 are manually operated by switches SW9, SW10, SW11, SW12, SW13 and SW14, respectively.

Dome loading for the remote regulators is accomplished by five two-way switches SW20, SW21, SW22, SW23 and SW24.

A timer to record firing durations has been included in the control system. Relay R12, which is energized by the "FIRE" switch, SW5, has been used to activate and deactivate the timer circuit.

Ignition System. A number of choices existed with regard to the selection of an ignition system. With propellant flowing, a pyrophoric chemical charge could be injected into the engine to cause spontaneous combustion of the resulting mixture. This system has been proven effective in other rocket test facilities, but it was judged to be unduly expensive, complex, and potentially hazardous in the present case.

Ignition could be produced by a hot wire technique. A fine nichrome wire might be stretched across the rocket engine exit area very near the exit plane. The ends of the wire would then be fastened to 28v. DC terminals through two all metal terminal blocks. In order to achieve ignition in this case, sufficient voltage is applied to the wire to cause resistive heating to the extent that the wire becomes "red hot". After ignition of the propellant mixture, the wire will burn through leaving the rocket exhaust plume free of obstruction.

It was finally decided to use spark ignition by allowing a spark to pass from a wire to the engine case at a point near the exit. For suitably chosen flow rates of the gaseous propellants, a combustible mixture is formed in the vicinity of the nozzle exit. This mixture can be ignited by the electrical spark and the flame front then can propagate to the injector face.

A pilot system was chosen whereby the propellants could be ignited aft of the engine nozzle exit. By limiting the initial propellant flow, the speed of the mixture of propellant gases was subsonic throughout the rocket engine and a flame front initiated at the nozzle exit of the motor can thus advance into the engine to the injector face. To achieve this initially reduced flow of the propellants, GO_2 and GH_2 were by-passed through pilot lines around the engine shut-off valves O2SV-4 and H2SV-6 and into the injector of the rocket motor. In the case of each of the propellants, the flow rate can be set by means of a hand-operated needle valve. Pilot flow is then initiated by opening the solenoid valves O2SV-5 and H2SV-1 (located in the pilot lines) for the RP-1 fuel or H2SV-6 for the GH_2 fuel, respectively. A "PILOT FIRE" switch located on the engine operator's control console is employed for this purpose. The function of the "PILOT FIRE" switch has been discussed in the preceeding section.

After pilot ignition has been obtained and stabilized, the rocket engine can be brought to full-thrust operation by actuating the "FIRE" switch, SW5, on the engine operator's control console. The "FIRE" switch energizes a relay which opens the engine shut-off valves as described in the preceeding section. In this configuration, the fuel and oxygen flow is controlled by the remote dome loaders until design pressures are established. This allows a smooth transition from the "PILOT FIRE" mode of operation to the full-thrust mode of operation. After thrust has stabilized at its rated level, the pilot system is deactivated by setting the "PILOT FIRE" switch to the "OFF" position.

Rocket Engine. In the late 1950's the Convair Division of General Dynamics constructed and tested a number of model rocket engines (1), (2). The particular engine selected and purchased for the Kansas State University UGRTF was a 160 pound thrust engine designed for two uses: (1) to provide a uniform, shock free jet for use in infrared radiation research under sea-level test conditions ($A/A^* = 5.26$) and (2) to provide an exhaust jet simulating the exhaust jet of a full-scale ICBM boost stage rocket engine ($A/A^* = 8$).

A sectional drawing of the 160 pound-thrust rocket engine is shown in Figure 10. The engine is composed of four basic parts: 1) the propellant injector, 2) the combustion chamber-expansion nozzle, 3) the coolant nozzle block and 4) the coolant jacket. Each part will be discussed briefly.

Propellant Injector. The propellant injector was constructed in two parts, the injector face and the injector body, from electrolytic tough-pitch copper. The injector face contains the propellant manifolds and two concentric rings of injector ports. The inner ring is for the fuel injection and the outer ring is for oxidizer injection. The ports are drilled at angles such that the fuel and oxidizer impinge upon a small circular area approximately .1 inches in front of the injector face.

The injector face of the GH_2/GO_2 engine is cooled by circulating water through a water passageway located directly behind the injector face. The injector face of the $RP\text{-}1/GO_2$ engine is cooled by the liquid RP-1 propellant

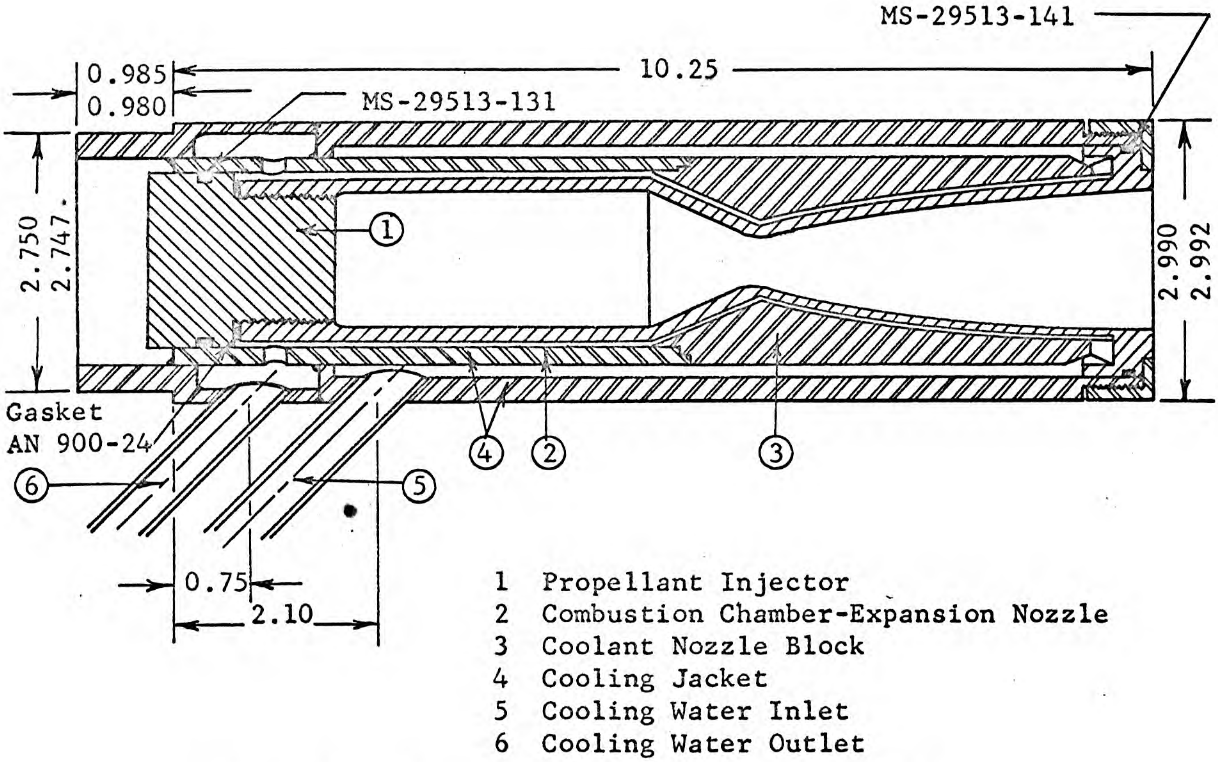

Figure 10. Convair 160-lb. Engine. A/A*=8. (3)

as it passes through the fuel manifold and injector ports. To allow monitoring of the combustion chamber pressure, a pressure tap is located in the injector face.

The injector body contains the propellant passages which lead from the propellant manifolds to the injector ports and a water passageway. The injector assembly is attached to the combustion chamber by screw threads and is sealed by a crush gasket.

Combustion Chamber-Expansion Nozzle. The combustion chamber-expansion nozzle is also fabricated from electrolytic copper. Three of these subassemblies were purchased from Convair. One of these 'liners' has a nozzle whose exit-to-throat area ratio is 8 and the other two are identical and have an exit-to-throat area ratio of 5.26 allowing exhaust at sea level ambient pressure under normal operating conditions. Any liner may be used with either injector sub-assembly. In all of these 'liners', the combustion chamber is 1.5 inches in diameter and 3.47 inches in length with a downstream conical section converging to the throat of the nozzle.

The expansion nozzle is a contoured nozzle; the nozzle coordinates were chosen to provide uniform axial parallel flow at the nozzle exit.

The combustion chamber-expansion nozzle is cooled by a counter-current axial flow of cooling water at 500 psia and a flow rate of approximately forty gallons per minute. The cooling water exits near the injector face.

Nozzle Block. The nozzle block is constructed from 304 stainless steel in two semi-cylindrical sections which fit around the nozzle 'liner'. Its function is to direct the water downstream over its exterior surface and provide an annular region for a water return upstream between the contoured interior surface of the block and the contoured exterior surface of the nozzle. Such an arrangement forces the water to flow in a counter-current direction with respect to the gases while cooling the nozzle.

Coolant Jacket. The coolant jacket is also constructed of 304 stainless steel. It forms the exterior casing for the rocket motor assembly. The coolant jacket is connected to the other engine components by a threaded cap at the exit plane of the nozzle. It is staticaly sealed at either end by "O" rings. The cooling water inlet and outlet tubes are welded to the side of the cooling jacket near its forward end.

Engine Mounting. The engine mounting was required to support the engine and to serve as a thrust indicator during power-on operation. Consideration was given to the construction of a thrust platform suspended on roller or ball bearings. The thrust could then have been measured by use of a load cell attached to the thrust platform. The advantages offered by this design were rigid confinement of the rocket engine, which was attached by two clamps, and accurate thrust indication. The fabrication of the thrust platform was judged to be unduly complex and expensive for the present application.

The engine mounting shown in Figure 11 was chosen as a more satisfactory mounting for the system under consideration. It is constructed from a 2024-T4 aluminum plate three-fourths inch thick. The plate can be attached to a rigid platform by four one-half inch bolts passing through matching holes at the base of the plate. The thrust plate was designed as two sections fastened together by two one-fourth inch bolts. The sections form a clamp when fitted around the rocket engine. The engine was designed with a 0.98 inch wide depression at the injector end around which the mounting plate can be attached. Instrumentation for the thrust indication is provided by the attachment of strain-guage transducers to sense the bending strain at the surface of the beam. The strain-guage transducers have been located on the centerline of the plate three inches from the base on either side of the plate.

Instrumentation. The primary parameters for which it was necessary to provide instrumentation were temperature and pressure. Chromel-Alumel thermocouples have been specified for temperature pick-ups. The temperature tap used at the inlet to each of the flow meters was designed to meet ASME codes. All other propellant line temperature indications were not deemed to be as critical as the temperature used in the flow rate calculations and have been obtained by connecting the thermocouples directly to the propellant lines. Temperature of the RP-1 fuel is sensed by a thermocouple immersed in the fluid. All temperatures except the coolant water temperature will be recorded on a Heiland multi-channel data recorder. The cooling water temperature difference across the rocket engine assembly is monitored by electrical

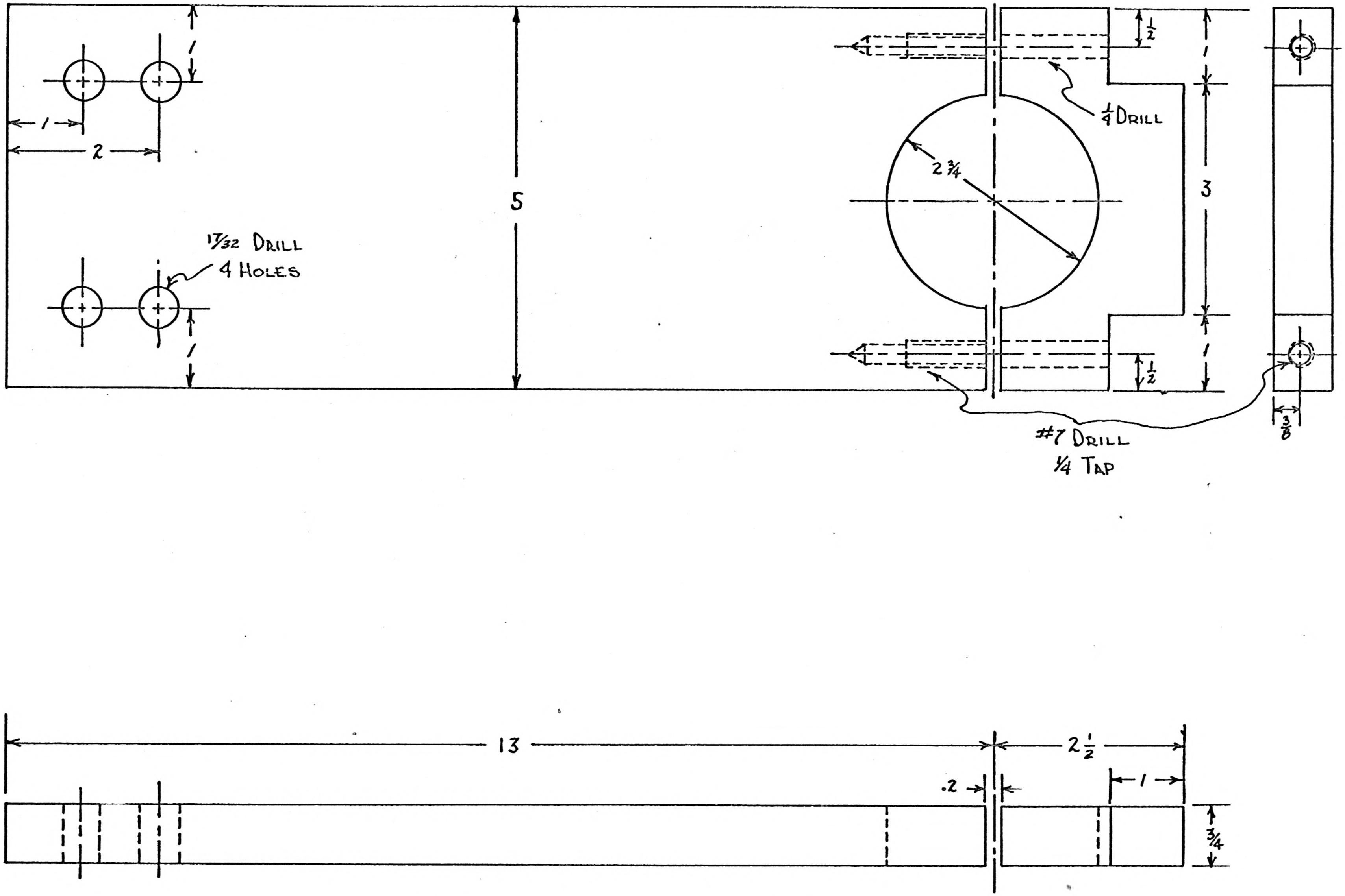

Figure 11. Engine Mounting Stand, 160 lb Rocket Engine.

contacts attached to the pins of a dual-channel strip chart recorder. The contacts are adjusted such that a signal is sent to the "H20 LIMIT" switch of the control system whenever the temperature difference between the inlet and the outlet water becomes excessive.

Two means of pressure measurement were considered for the rocket test facility design: pressure transducers and pressure gages. Pressure gages were more attractive from an economic point of view but appeared to be potentially troublesome from the point of view of safety. It would have been necessary to attach numerous lines containing high pressure fluids to gages located near the control console. A rupture of any of these lines could have caused considerable damage to equipment and possible injury to personnel and could also have posed a serious fire hazard in the event that the rupture occurred in a propellant line.

In view of the above it was decided to employ pressure transducers for pressure measurements. The Edcliff model 2-8-6 pressure transducer was selected for use to indicate the initial supply pressures and the available purge pressures. Consolidated Electrodynamics Corporation type 4-326, 0-1500 psia pressure transducers were chosen for indications of the primary regulator pressures in both the propellant and control gas lines. Consolidated Electrodynamics Corporation type 4-326, 0-850 psig pressure transducers were selected for pressure indications from the secondary regulators, the secondary control gas lines, the combustion chamber of the rocket engine and the inlet cooling water line. No final selection of pressure transducers has yet been made for use to measure the pressure difference across the orifice plates. The pressure data are to be recorded on a Heiland multi-channel data recorder.

A one-inch-diameter Potter flowmeter was selected to measure the outlet engine cooling water flow rate. The Potter meter's electrical output is connected to a frequency meter. The frequency meter is then used to control a permissive switch in the firing circuit.

Closed-circuit television equipment will be used to monitor the model test area. This arrangement will allow visual observation of the firing proceedings without exposing the observers to possible danger.

Safety Features. The primary guideline for the system design was personnel and facility safety. The major structural material is stainless steel 316, and a minimum safety factor of three was specified for any burst pressure.

Four safety features were incorporated into the coolant system to insure proper operation and, thereby, to protect the rocket motors. A ΔT switch, which measures the temperature differential between the inlet and outlet cooling water has been included in the design to prevent engine operation above a predetermined critical temperature differential. A pressure switch on the water inlet side of the rocket motor cooling system insures that proper water supply pressure will be maintained by the coolant system during firing. The flow rate of coolant is monitored by a Pottermeter on the outlet side of the rocket motor. The firing sequence would be terminated should the coolant flow rate fall below a critical value. The coolant pressure outside the rocket motor combustion chamber liner was specified to be greater than the chamber pressure. This specification prevents propellants from contaminating the coolant system in the event of a rocket motor 'liner' failure.

Due to the high chemical activity of the rocket propellants, an inherent danger exists in their presence. To minimize this danger, the oxidizer system was completely isolated from either of the fuel systems. Thus, the oxidizer and the selected fuel cannot mix until after they are separately injected into the combustion chamber.

Separate purge systems were used to preclude inadvertant mixing of propellants through the purge system. If a common purge system had been used, some possible combinations of malfunctions in the propellant supply systems and the common purge system would have allowed mixing of the propellants. Storage facilities have been segregated. Each area has been provided with an electronic warning system to detect the presence of any propellant should leakage occur. Critical components have been protected by pressure relief valves. One-way check valves preclude the possibility of oxidizer flow into a fuel line and vice versa. The possibility of the installation of the wrong piece of hardware into the oxygen system was guarded against by imposing the condition of oxygen and fuel compatibility upon the selection of all component hardware used in the design.

Each of the above features was covered in greater detail under its respective section of this report.

CONCLUSIONS

The Undergraduate Rocket Test Facility (UGRTF) described and analyzed in this report will provide a safe and versitile assembly by which elementary rocket propulsion principles may be demonstrated. The basic guidelines pertaining to component selection, overall construction, and control were found to be technically sound. However, the design presented should not be accepted as final. As operating experience is gained, valid modifications may become evident and should be incorporated into the system.

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APPENDIX

Flow Rate and Pressure Drop Calculations. The following calculations were used to estimate the storage volume and line size requirements for the propellant systems. The storage volume needed was determined by the mass flow rate of the propellants and the desired firing duration. The line sizes used were determined by trial and error approximations of the associated pressure drops incurred during propellant transfer.

Flow Rate Calculations.

GO_2/GH_2 System. The stoichiometric equation for hydrogen combustion is:

$$\tfrac{1}{2}O_2 + H_2 \rightarrow H_2O \tag{1}$$

The corresponding mass balance is:

$$16 + 2 \rightarrow 18 \tag{2}$$

The mass ratio at stoichiometric conditions is 16/2 or 8. The design parameters for the mass ratio were chosen to be between 4 and 16. The thrust, T, for a rocket engine is given by the relationship

$$T = \dot{w}_T I_{sp} \tag{3}$$

where,

$\dot{w}_T$ = the total mass flow rate

I_{sp} = the specific impulse.

The specific impulse for hydrogen and oxygen is approximately 300 seconds and the thrust produced by the engine is approximately 150 pounds. Solving Equation (3) for the total mass flow rate, $\dot{w}_T$ was found to be 0.5 pounds per second.

The maximum gaseous oxygen (GO_2) flow rate will occur at a mass ratio of 16, from which

$$MR = \dot{w}_{O_2}/\dot{w}_{H_2} \tag{4}$$

or $$\dot{w}_{O_2}/\dot{w}_{H_2} = 16$$

and $$\dot{w}_{H_2} = 1/16\ \dot{w}_{O_2}. \qquad (5)$$

The total flow rate $\dot{w}_T$ is given by,

$$\dot{w}_T = \dot{w}_{O_2} + \dot{w}_{H_2}. \qquad (6)$$

Substituting the value for $\dot{w}_T$ and Equation (5) into Equation (6)

$$\dot{w}_{O_2} = 8/17 \text{ lb/sec}. \qquad (7)$$

The density of oxygen (GO_2) at 700 psi and 80^oF, assuming that the perfect gas law applies, is given by

$$\rho_{O_2}(700 \text{ psia}) = P/RT$$

or $$\rho_{O_2}(700 \text{ psia}) = 3.87 \text{ lb/ft}^3. \qquad (8)$$

The volumetric flow rate, Q, is given by

$$Q = \dot{w}/\rho$$

from which

$$Q_{O_2\,(max)} = 7.296 \text{ ft}^3/\text{min} \qquad (9)$$

and is the maximum flow rate of gaseous oxygen at a pressure of 700 psia. The maximum flow rate of oxygen at standard conditions was found from the value given in Equation (9) to be 347.44 SCFM. The maximum GO_2 flow rate is set by maximum oxidizer to fuel ratio operation of the GO_2/GH_2 system.

A similar treatment for the hydrogen flow rate at mass ratio of 4 will result in

$$\dot{w}_{H_2} = 0.1 \text{ lb/sec} \qquad (10)$$

$$Q_{H_2} = 24.79 \text{ ft}^3/\text{min} \qquad (11)$$

at a pressure of 700 psia and 80^oF or, at standard conditions, a volumetric flow rate of 1,180.5 SCFM was calculated.

Using the above values and isothermal expansion in the tanks, the volumes reported in the <u>Propellant Supply and Storage</u> section were calculated. Actually the flow will be somewhere between isothermal and adiabatic under normal operating conditions. Calculations using the assumptions of reversible adiabatic flow from the tanks indicate an additional gas supply requirement of 15% for a five-minute-duration firing. However, this additional volume is recoverable after sufficient time has elapsed to allow the temperature to return to ambient.

GO_2/RP-1 System. The stoichiometric equation for RP-1 combustion, assuming that the composition of RP-1 is CH_2, is:

$$CH_2 + 3/2O_2 \rightarrow CO_2 + H_2O. \tag{12}$$

The corresponding mass balance is:

$$14 + 48 \rightarrow 62. \tag{13}$$

The mass ratio at stoichiometric conditions is 48/14 or 3.428. The specific impulse of gaseous oxygen and RP-1 is approximately 275 sec. Therefore, the total mass flow, $\dot{w}_T$, for the GO_2/RP-1 system is approximately the same as for the GO_2/GH_2 system or 0.5 pounds per second. The minimum oxidizer to fuel ratio was assumed to be 0.5. Therefore,

$$\dot{w}_{O_2}/\dot{w}_{RP-1} = 0.5 \tag{14}$$

$$\dot{w}_T = \tfrac{1}{2} = \dot{w}_{O_2} + \dot{w}_{RP-1} \tag{15}$$

or

$$\dot{w}_{RP-1} = 1/3 \text{ lb/sec.} \tag{16}$$

The density of RP-1 is given by

$$\rho_{RP-1} = (62.4)\ (0.8) = 50 \text{ lb/ft}^3. \tag{17}$$

The maximum flow rate of RP-1 was calculated to be 0.4 ft^3/min.

Pressure Drop Calculations. As long as the velocity of the propellants does not exceed 200 ft/sec, they may be considered to be incompressible fluids. This assumption was employed throughout the pressure drop calculations. The pressure drop for an incompressible fluid is given by the relationship,

$$\Delta P/L = (-4f/D)(\tfrac{1}{2}\rho V^2). \tag{1}$$

For turbulent flow, Blasius (4) suggested the following expression for the friction factor, f:

$$f = 0.079(Re)^{-\frac{1}{4}}. \tag{2}$$

Substituting Equation (2) and the relationship for the Reynold's number into Equation (1).

$$\Delta P/L = -7.62 \times 10^{-4} \left[\frac{\mu \rho^3 V^7}{D^5}\right]^{\frac{1}{4}}. \tag{3}$$

where the dimensions are

D = inches

P = lb/in^2

L = feet

V = ft/sec

ρ = lb/ft^3

μ = lb/ft-sec.

From the continuity equation

$$AV = \dot{w}/\rho = \pi D^2 V/4(144)$$

or

$$V = 183.2(\dot{w}/\rho)(1/D^2). \tag{4}$$

Substituting Equation (4) into Equation (3) and rearranging,

$$(\Delta P/L)(D)^{4.75} = (7.0/\rho)(\dot{w})^{\frac{7}{4}} (\mu)^{\frac{1}{4}}. \tag{5}$$

For gaseous oxygen at 700 psia,

$$\mu_{O_2} = 1.34 \times 10^{-5} \text{ lb/ft-sec} \tag{6a}$$

$$\rho_{O_2} = 3.87 \text{ lb/ft}^3 \tag{6b}$$

$$\dot{w}_{O_2} = 8/17 \text{ lb/sec.} \tag{6c}$$

Substituting Equations (6a), (6b), and (6c) into Equation (5),

$$(\Delta P/L)_{O_2} (D_{O_2})^{4.75} = 0.0293 \tag{7}$$

For ½-inch tubing, of the required burst strength, the inside diameter is 0.356 inches resulting in

$$(\Delta P/L)_{O_2} = 4.0 \text{ (lb/in}^2\text{)(1/ft)}. \tag{8}$$

For 3/4-inch tubing, of the required burst strength, the inside diameter is 0.56 inches resulting in

$$(\Delta P/L)_{O_2} = 0.392 \ (\text{lb/in}^2)(1/\text{ft}). \qquad (9)$$

For gaseous hydrogen at 700 psia,

$$\mu_{H_2} = 5.35 \times 10^{-6} \ \text{lb/ft-sec} \qquad (10a)$$

$$\rho_{H_2} = 0.242 \ \text{lb/ft}^3 \qquad (10b)$$

$$\dot{w}_{H_2} = 0.10 \ \text{lb/sec}. \qquad (10c)$$

Substituting Equations (10a), (10b), and (10c) into Equation (5),

$$(\Delta P/L)_{H_2}(D_{H_2})^{4.75} = 0.248. \qquad (11)$$

For ½-inch tubing with an inside diameter of 0.356 inches,

$$(\Delta P/L)_{H_2} = 3.38 \ (\text{lb/in}^2)(1/\text{ft}). \qquad (12)$$

For 3/4-inch tubing with an inside diameter of 0.56 inches,

$$(\Delta P/L)_{H_2} = 0.333 \ (\text{lb/in}^2)(1/\text{ft}). \qquad (13)$$

For RP-1,

$$\mu_{RP-1} = 0.0674 \ \text{lb/ft-sec} \qquad (14a)$$

$$\rho_{RP-1} = 50 \ \text{lb/ft}^3 \qquad (14b)$$

$$\dot{w}_{RP-1} = 0.333 \ \text{lb/sec}. \qquad (14c)$$

Substituting Equations (14a), (14b), and (14c) into Equation (5),

$$(\Delta P/L)_{RP-1}(D_{RP-1})^{4.75} = 0.0104. \qquad (15)$$

For ½-inch tubing with an inside diameter of 0.356 inches,

$$(\Delta P/L)_{RP-1} = 1.42 \ (\text{lb/in}^2)(1/\text{ft}). \qquad (16)$$

For 3/4-inch tubing with an inside diameter of 0.56 inches,

$$(\Delta P/L)_{RP-1} = 0.14 \ (\text{lb/in}^2)(1/\text{ft}). \qquad (17)$$

A DESIGN ANALYSIS OF AN UNDERGRADUATE
ROCKET TEST FACILITY

by

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AN ABSTRACT OF A MASTER'S REPORT

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ABSTRACT

A description and analysis of the Undergraduate Rocket Test Facility designed by the Mechanical Engineering Department at Kansas State University was presented in this report. The purpose of the analysis was to examine the overall design concept and the safety features of the design.

The facility used a 160-pound thrust rocket motor that was designed and fabricated by the Convair Division of General Dynamics. The rocket fuels considered in the design were gaseous hydrogen and RP-1. Gaseous oxygen was used as the oxidizer. The design pressure of the system was 3000 psi. Because of the high pressure and the chemical nature of the propellants, the primary construction material was stainless steel. Gaseous nitrogen was used as the control gas for the two-stage pressure regulation of each propellant and as a purging agent for the propellant lines.