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SUBLIMING SOLID CONTROL ROCKET

PHASE I

Prepared by
ROCKET RESEARCH CORPORATION
Seattle, Wash.
for Goddard Space Flight Center

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Prepared under Contract No. NAS 5-3599 by
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ABSTRACT

The Subliming Solid Control Rocket is an attitude control propulsion device employing a solid propellant which slowly sublimes to a low molecular weight vapor. This vapor is exhausted by means of propellant valves through nozzles to produce reaction thrust. The principal advantage of the Subliming Solid Control Rocket is that its propellant is stored at high bulk density and low pressure. This permits the use of small, low weight propellant tanks and minimizes leakage.

A development program sponsored by Goddard Space Flight Center, NASA had as its principal goals the examination of the fundamental operating principles of the Subliming Solid Control Rocket and to study optimization of components.

This Phase I final report is divided into three parts. The first deals with the concept of operation of the Subliming Solid Control Rocket and is a detailed examination of its operating principles and important components. The second part of the report describes the experimental and analytical results of the developmental program. The third section of the report systematically discusses the equations and design procedures for the Subliming Solid Control Rocket.

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1.0 SUBLIMING SOLID REACTION CONTROL SYSTEM DESCRIPTION

1.1 Introduction

1.1.1 Contract Scope

Reaction control jets are required on most space vehicles flown today to perform such functions as: Flight trajectory control, orbit control, maintaining vehicle attitude and spin rate. In many cases, the total impulse requirements for such reaction control systems are moderate. In these cases, in the interest of simplicity and reliability, a class of reaction jets known as cold gas reaction control systems is applied. In general, such cold gas reaction control systems take the form of a compressed gas (usually nitrogen) which is placed in a high pressure propellant container at 3,000 to 6,000 psi. A regulator reduces the gas pressure to a useful working level and the gas is then exhausted by a means of conventional solenoid valves through nozzles to produce thrust when required. Propellant tank volume is reduced by storing the gas under high pressure. This high pressure propellant storage has three disadvantages:

a. A high pressure gas reaction control system leaks excessively when operating life exceeds six months.

b. Weight of the propellant tank is excessive.

c. A loaded high pressure cylinder is dangerous and requires special handling.

The objective of Contract NAS 5-3599 was to examine a new cold gas reaction control system which does not have the disadvantages of the high pressure system. The reaction control system studied under this contract was the Subliming Solid Control System (SSRCS).

1.1.2 Subliming Solid Reaction Control System Development

Rocket Research Corporation experimentally demonstrated the operation of a SSRCS in the Spring of 1962. The first laboratory test hardware of the Subliming Solid Control Rocket was delivered to NASA Goddard Space Flight Center in May, 1963. Contract NAS 5-3599 was then awarded to Rocket Research Corporation by Goddard Space Flight Center to begin development of the Subliming Solid Control Rocket for flight application. Contract NAS 5-3599 was the first phase in system development, and dealt primarily with the investigation of propellant characteristics and components. Specific areas to be studied were as follows:

a. Propellant Properties

b. Thermal Conditioning

c. Migration and Recondensation

d. Propellant Packaging

e. Valves and Filters

f. System Testing

This document incorporates the results obtained under Contract NAS 5-3599. It is written in the form of a design manual which will allow utilization of the information obtained for preliminary design purposes. Section 1.0 contains a discussion of all aspects of the Subliming Solid Reaction Control System. Section 2.0 contains the specific results of experiments performed as a part of this contract. Section 3.0 outlines the procedure to be followed in designing a typical SSRCS. A specific example, using all required equations, is also included in Section 3.0.

This report will be supplemented from information obtained from Phase 2 of the overall development program for the SSRCS which deals with the design and testing of a prototype and flight Subliming Solid Control Rocket aboard the OV2-1 satellite.

1.1.3 Concept of Operation

The operation of the Subliming Solid Reaction Control Rocket is exceedingly simple. A typical unit is shown in Figure 1-1. The propellant is stored as a solid crystalline mass in a thin wall propellant tank of any shape. A simple filter keeps the solid in the tank and out of the propellant lines and valves while solid-vapor equilibrium maintains the vapor pressure in the tank. When a propellant valve is opened, the vapor escapes through a nozzle producing thrust. The solid propellant in the tank then sublimes rapidly to replace the vapor which has been released. No combustion, ignition, or catalytic decomposition is required.

1.1.4 General Characteristics

The following general characteristics may be used for preliminary design application studies:

Vapor Pressure:	SUBLEX A	7.0 psia at room temperature
	SUBLEX B	1.4 psia at room temperature

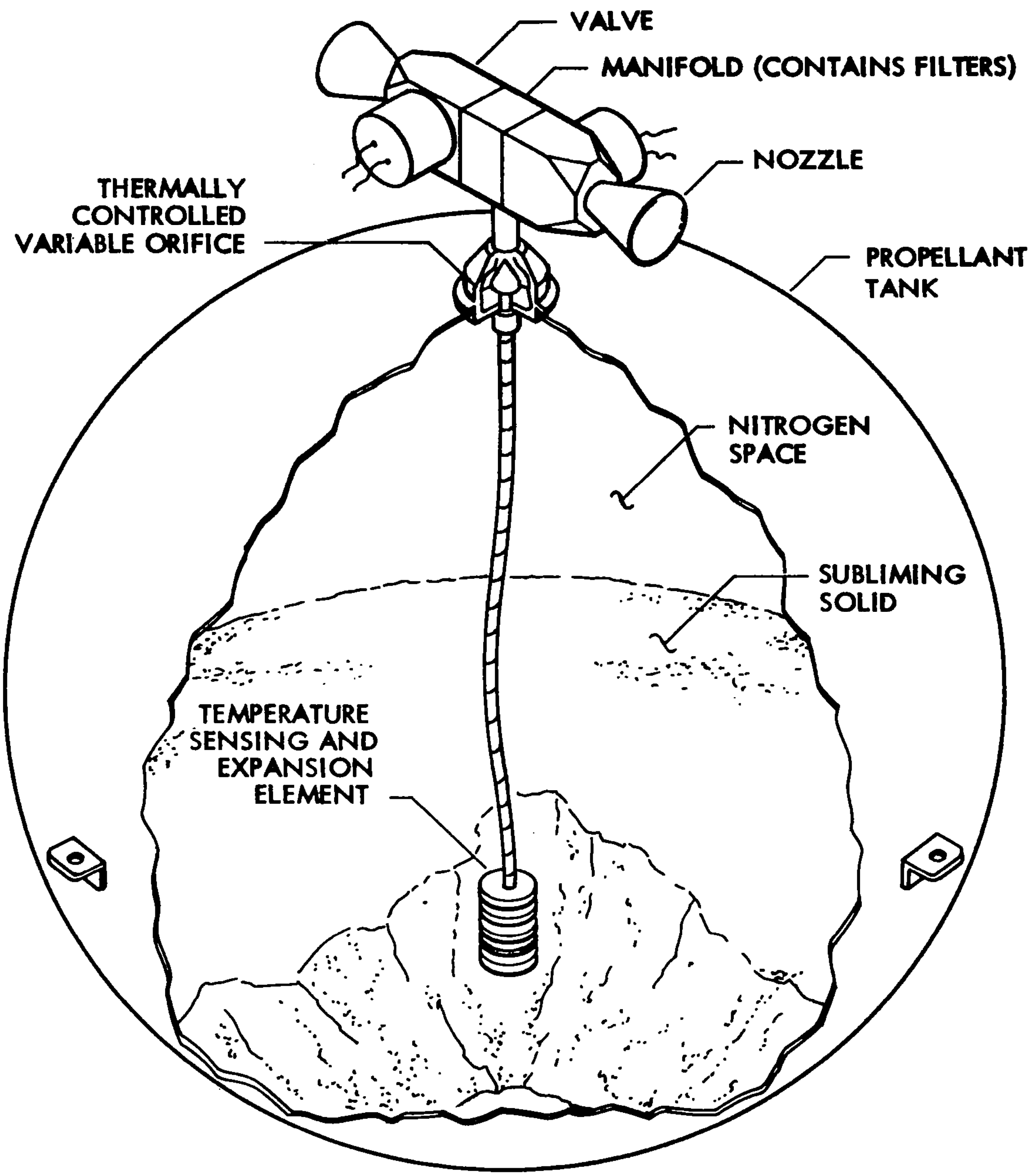


FIGURE 1-1. TYPICAL SUBLIMING SOLID CONTROL ROCKET

Change in Vapor Pressure With Temperature:	Approximately a factor of 2 for every 20°F
Heat of Sublimation:	10 watts per 0.001 pound thrust
Vapor Molecular Weight:	SUBLEX A 25.5
	SUBLEX B 24.0
Propellant Density:	Minimum 0.027 pound per cubic inch
	Maximum 0.04 to 0.05 pound per cubic inch
Specific Impulse:	84 to 85 seconds theoretical vacuum at 100 to 1 area ratio
Propellant Tank Weight Factor:	0.1 pound or less per pound propellant stored
Typical Valve Weight:	0.1 to 0.25 pound
Typical Nozzle Weight:	0.05 pound
Typical Weight of Variable Thermal Control Orifice:	0.25 to 0.5 pound

1.1.5 Subliming Solid Reaction Control System Advantages

A SSRCS has the following advantages over a high pressure cold gas system:

a. Lower Overall Weight

Only 0.1 pound of propellant tank weight is required for each pound of propellant stored. A high pressure cold gas system normally requires at least one pound of tank for each pound of propellant stored. This reduced tank weight is due to the lower storage pressure. (Cold gas nitrogen is stored at 3,000 to 6,000 psia, while the Subliming Solid Propellant is stored at approximately 7 psia. The total system weight for the subliming solid is approximately 15.3 pounds which is less than half that of a cold nitrogen system.

b. Use of Odd Shaped Tanks Possible

Because of low operating pressure, odd shaped tanks may be used when required to fit available cavities within a space vehicle while still maintaining a low weight.

c. Lower Inherent Leakage

The lower inherent leakage is due to the fact that the subliming propellant is stored as a solid at 7 psia and the cold gas (usually nitrogen) is stored at between 3,000 and 6,000 psia.

d. Lower Propellant Volume

Propellant tank volume is reduced in the SSRCS because the propellant is stored as a solid providing a far higher density in the tank.

e. Greater Specific Impulse

Comparing the subliming propellant with cold nitrogen shows that the theoretical vacuum specific impulse at 70°F is 84 to 85 seconds for the subliming propellant at an expansion ratio of 100 to 1 while the cold nitrogen gas has a vacuum specific impulse of 76 seconds.

In addition, the SSRCS operation does not require combustion, ignition, or catalytic decomposition, as is necessary with other high energy systems.

1.2 Propellant

1.2.1 Propellant Requirements

The propellant is the key to a successful Subliming Solid Reaction Control System. There are many subliming compounds which could be considered as possible candidates for subliming solid rocket propellant. However, in order to be a practical propulsion fluid, the propellant must have certain important characteristics. These characteristics are:

a. A useful vapor pressure

b. A reasonably low molecular weight

c. Good stability

1.2.1.1 Useful Vapor Pressure

A useful vapor pressure is necessary to produce the required thrust level within the limits of available hardware size. A low exhaust fluid molecular weight is desirable in order to achieve the highest possible specific impulse so that the SSRCS will be competitive with cold nitrogen gas. The propellant must have good stability to permit long duration missions and long storage periods. In addition, the propellant should be noncorrosive, easy to handle and store, and have no subliming rate limitations.

The propellant choice for a specific subliming solid reaction control system is discussed in further detail in the System Design Section, Paragraph 3.3.2. It has been found, however, that SUBLEX A and SUBLEX B have the best combination of characteristics for the most applications. Of the two, SUBLEX A is the most widely applied. For this reason, the test program concentrated on SUBLEX A thoroughly and as a result, SUBLEX A is discussed most often in this report. Some additional work on SUBLEX B was also performed.

The desired vapor pressure for a subliming solid propellant depends primarily on the thrust level to be achieved. A low vapor pressure keeps the propellant tank weight to a minimum. In addition, where very low thrust levels are anticipated, a low vapor pressure permits the use of a sufficiently large exhaust orifice to prevent the possibility of plugging. However, vapor pressure must be high enough at any given thrust level so that size of propellant valves and lines may be reasonably small, in order to minimize hardware volume and weight. At thrust levels below one (1) pound, a vapor pressure of below 10 psia is desirable. At thrust levels below 10^{-3} pounds, a vapor pressure of below one (1) psia is generally desirable.

The propellant choice which yields the best vapor pressure in any given application, will also depend on operating temperature. Figure 1-2 includes the room temperature vapor pressures for various candidate Subliming Solid Propellants. Of the propellants listed, SUBLEX A is generally the most useful and widely applied for thrust levels between 10^{-3} and 0.1 pound. This is due to its vapor pressure of 7 psia at room temperature and its high volitility. Figure 1-2 is a graph of vapor pressure versus temperature for SUBLEX A and SUBLEX B propellants. Tests conducted to determine propellant vapor pressure are discussed in Paragraph 2.7.

1.2.1.2 Molecular Weight

The equation which governs the specific impulse, I_{sp}, of a rocket exhaust fluid is as follows:

$$I_{sp} = \left[\left(\frac{2}{g}\right)\left(\frac{k}{k-1}\right)\left(\frac{R' \; T}{M}\right)\eta\right]^{1/2}$$

Where:

g	=	32.2 ft/sec^2
k	=	specific heat ratio for exhaust fluid
R'	=	universal gas constant, 1,544 ft-lb/mole °R
T	=	upstream stagnation temperature °R

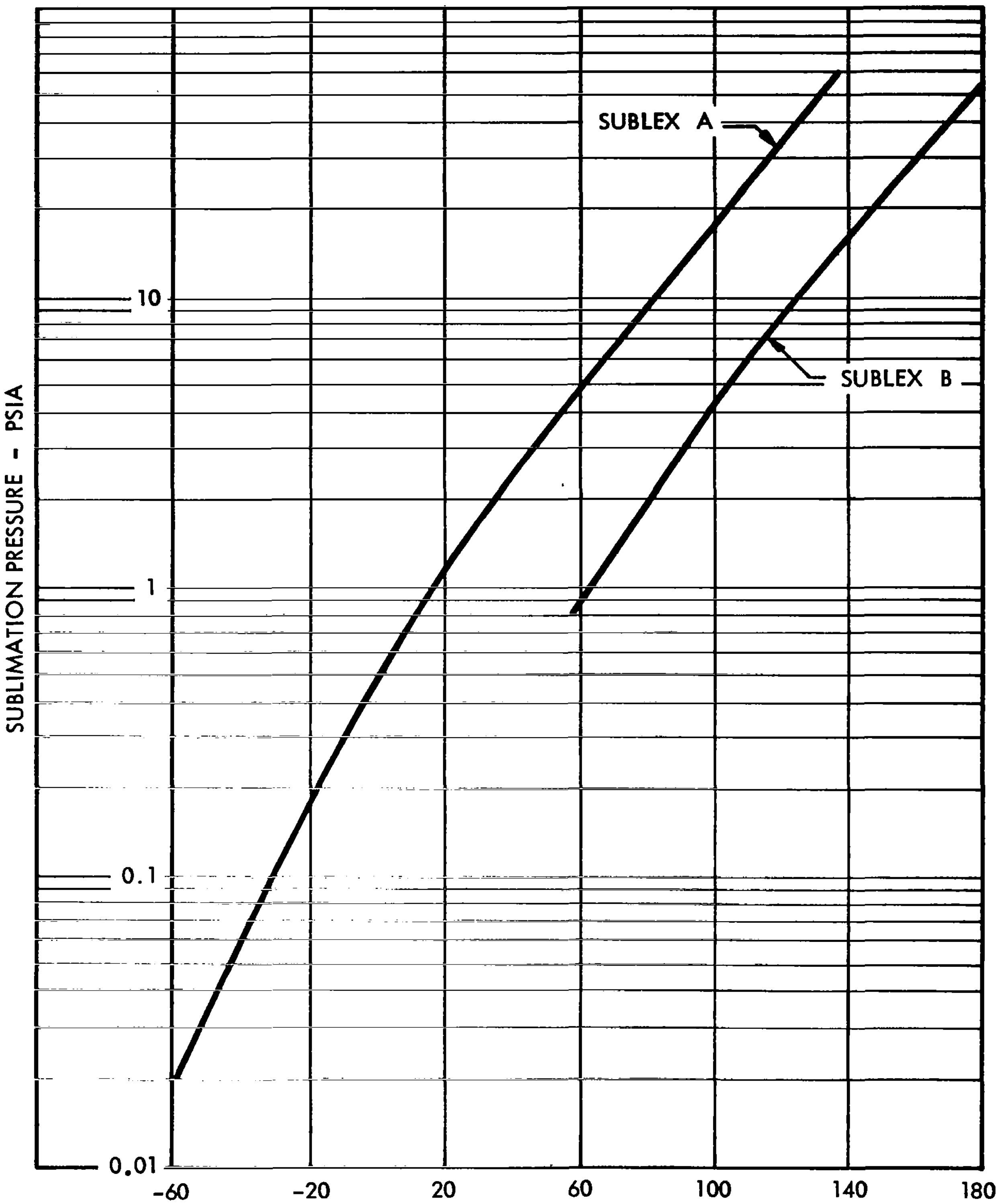


FIGURE 1-2 SUBLIMATION PRESSURE VS. TEMPERATURE FOR SUBLEX A AND SUBLEX B

M	=	exhaust fluid average molecular weight, lb/mole
η	=	rocket cycle efficiency, (a function of the nozzle area ratio and expansion pressure ratio)

It can be seen from this equation that the specific impulse of the propellant will be inversely proportional to the square root of its molecular weight. Therefore, for any given operating temperature, it is desirable to have a propellant with as low a molecular weight as possible. The high pressure nitrogen cold gas reaction jet system has a molecular weight of 28. The SSRCS, in order to be competitive, must have specific impulse on the same order as that of nitrogen. The amount of propellant to perform a specific mission would, therefore, be the same; and a significant savings in weight could be obtained with the SSRCS due to reduced propellant tank weight. For this reason, only those subliming solid materials that had vapor phased products with a molecular weight lower than 35 were considered to be usable propulsion fluids. The average exhaust molecular weight of SUBLEX A is 25.5 and 24 for SUBLEX B.

The second factor, which influences the specific impulse of the propellant, is the specific heat ratio, k. SUBLEX A has a specific heat ratio of approximately 1:31. Nitrogen gas has a specific heat ratio of 1:40. The combined effects of specific heat ratio and molecular weight result in a specific impulse of 85 seconds for SUBLEX A propellant as compared with 76 seconds for a cold nitrogen gas system under vacuum conditions with an expansion ratio of 100:1.

1.2.1.3 Propellant Stability

Propellant stability which is required in order to perform repeatable and reliable missions, is attained when the following conditions are met:

a. The propellant must not slowly dissociate or decompose and thereby build up gas pressure within the propellant storage tank.

b. The propellant must not slowly polymerize in such a way that the propellant becomes less volatile with time.

c. A constant vapor pressure for any given temperature must be maintained for a period far longer than the duration of the required mission.

d. The propellant must be nonexplosive and not overly sensitive to air and moisture to allow easier handling and low cost.

e. The propellant must not be too reactive or corrosive. See Table 2-3 for compatibility of SUBLEX A with various materials.

The specific tests performed on SUBLEX A are described in detail in the propellant testing section. (See Paragraph 2.5.) SUBLEX A was found to be completely stable with time. Jars of SUBLEX A have been stored for periods of over two years without any noticeable change in vapor pressure or physical appearance. It has been found, however, that SUBLEX A propellant exposed at length to air and moisture will degrade, as evidenced by the formation of a yellow discoloration on the surface of the propellant. Momentary exposure of samples of SUBLEX A propellant to air for 30 seconds to 1 minute will degrade less than one percent of the propellant exposed. However, prolonged exposure will seriously affect the bulk of the SUBLEX A exposed. Sublimation of the degraded propellant will occur at a lower rate than with pure SUBLEX A propellant. Degradation of SUBLEX A propellant has not been noticed when the propellant is exposed only to dry nitrogen and does not come in contact with air.

No spontaneous decomposition of SUBLEX A propellant (releasing gradual quantities of gas which build up on the propellant tank) has ever been observed. SUBLEX B is also exceedingly stable. The only effect observed is a gradual change of the compound under prolonged exposure to moisture.

1.2.2 Subliming Rate

Subliming rate is another important characteristic related to the vapor pressure of the subliming solid propellant. Some propellants, even under flow conditions, will sublime rapidly enough to maintain equilibrium vapor pressure, while other propellants have a significant subliming rate limitation. Under flow conditions these propellants will not sublime rapidly enough to maintain equilibrium pressure in the propellant tank. In cases of subliming rate limitations, the thrust produced by a SSRCS will be lower than predicted by the equilibrium vapor pressure for any given temperature. Furthermore, thrust will tend to decrease as the amount of propellant remaining in the tank decreases.

In any specific case, subliming rate limitation may be due to one of two causes. One cause of rate limitation is that when limited in area, the propellant surface chills rapidly due to sublimation which is caused by low thermal conductivity through the subliming solid crystal. Therefore, although sufficient heat is retained within the solid crystal to cause sublimation, the surface may become sufficiently colder than the bulk of the crystal. This results in a lower vapor pressure on the surface than might be indicated for the temperature of the bulk of the crystal. In finely divided granular form, surface chilling is not nearly so severe due to small crystal dimensions which results in a high ratio of surface to total mass. This permits heat transfer to occur more readily from the total mass available.

The second cause of subliming rate limitation deals with the kinetics of the subliming process. In this case, vapor leaves the solid surface more slowly than is theoretically possible, in spite of the fact that sufficient heat and temperature is available to cause sublimation. This process is not clearly understood, but seems to occur in certain classes of compounds.

SUBLEX A, in granular form, has not exhibited any rate limitations at thrust levels up to 5×10^{-2} pounds. The recondensed form of SUBLEX A may exhibit a small rate reduction due to the small surface area. SUBLEX B, on the other hand, has exhibited subliming rate limitation in certain instances as discussed in Paragraph 2.8.

From the experimental data gathered thus far, it appears that SUBLEX A may be used at virtually any thrust level which is practical for a Subliming Solid Control Rocket. SUBLEX B, on the other hand, may only be useful at the lower thrust level; in those cases where appreciable quantities of propellant are carried, or where some gradual decrease in thrust may be tolerated as the propellant is consumed.

1.2.3 Handling and Storage

A subliming solid propellant must be easy to handle and store. The ability to use conventional procedures for propellant handling reduces system cost and results in maximum flexibility. SUBLEX propellant, if exposed to air for sufficient periods of time, will absorb moisture. Therefore, SUBLEX propellants must be handled in a dry box, which is flushed with dry nitrogen. The handling procedures for propellants in a dry box are not difficult and may be performed routinely.

SUBLEX A must be stored in a dry nitrogen atmosphere if prolonged storage is anticipated. Propellant tanks loaded with SUBLEX A are generally shipped with a dry nitrogen topping atmosphere in the tank, at positive pressure, to prevent leakage of air into the tank.

Aside from handling in a dry box, and a dry atmosphere, both SUBLEX A and SUBLEX B require no special handling procedures. They are completely safe with respect to shock and sensitivity, and since they are not reactive chemicals, no danger of explosion exists. SUBLEX propellants may be stored for long periods of time without pressure buildup within the storage vessel.

1.2.4 Materials Compatibility

The subliming solid propellant should be compatible with the usual spacecraft construction materials. However, some care must be exercised in the choice of materials to be used in the construction of the Subliming Solid Control Rocket. A detailed list of materials which are compatible with SUBLEX A is shown on Table 2-3. Paragraph 2.5 summarizes the materials compatibility testing.

In general, the only serious compatibility problem is with copper or copper alloys. Copper will react with SUBLEX B to form nonvolatile compounds. SUBLEX A will corrode copper severely in the presence of air and moisture. Even without air and moisture, continued exposure to high density SUBLEX A vapor will slowly corrode copper. For this reason, copper and any copper alloy, or any copper underplating, should be completely avoided on Subliming Solid Control Rocket components. Aluminum and stainless steel are completely compatible with the propellants and may be used on any component. However, there is evidence that corrosion of copper and silver does not occur in a space vacuum. This is apparently due to the fact that under the pressure found in and around a satellite in space, there is insufficient SUBLEX propellant vapor to cause corrosion. Corrosive tests on material, including copper and brass, were conducted by NASA in the Goddard Space Flight Center vacuum chamber. Only copper, of the materials placed near the exhaust nozzle of the SSRCS, showed slight discoloration after several hours of continuous exposure. It is felt, that due to the rapid dissipation of exhaust vapor from a flight vehicle in space, that SUBLEX A should not present any serious problem or potential hazard to satellite components.

1.3 Propellant Tanks

1.3.1 Requirements

The basic requirements for a subliming solid propellant tank are:

a. Minimum weight

b. Minimum volume

c. Ease of packaging and installation in the spacecraft

d. Efficient heat transfer

1.3.2 Propellant Tank Construction

Thin wall aluminum is best suited for a subliming solid propellant tank and has been used almost exclusively in the construction of experimental and flight prototype subliming solid systems.

Aluminum offers light weight, structural integrity, relative ease of construction, and where necessary, good heat transfer capability. Aluminum is also easily spun to obtain any desired shape for unusual packing situations and may be anodized or painted for thermal control.

Although aluminum is the preferred material, others such as fiberglass, stainless steel, certain plastics, and in special cases polymers like Teflon or Buna-N rubber, can be used. The only requirement is compatibility with the propellant, which will exclude materials such as copper and brass or any other copper compound. See Table 2-3 for materials compatibility information.

The low vapor pressure (generally 15 psia or less) of the subliming solid propellant permits the use of minimum gauge materials. Tank wall thickness is a function of the specific propellant tank design. For aluminum, 0.020 inches is recommended. Somewhat higher wall thicknesses are required if the Subliming Solid Control Rocket is to operate at high temperatures. This requirement is due to the increase in vapor pressure as temperature rises. Standard pressure vessel equations may be used to calculate the specific wall thickness required for propellant tanks, but in no case should the wall thickness be below 0.020 inch.

The low pressure of the subliming solid propellant allows a great deal of flexibility in the design or configuration of the propellant tank. Standard aluminum welding and brazing assembly techniques are applied in construction.

Under certain conditions, epoxy materials may also be used to join parts of the propellant tank. However, before a specific type of epoxy is used, propellant compatibility tests should be performed. Lightweight flanges, bolts, and certain clamps may also be used in propellant tank construction when required or desired. All standard rules, governing high pressure vessels, need not be followed in the design of subliming propellant tanks if the operating temperature of the subliming solid propellant is such that only low pressures will be achieved. Conventional "O"-ring seals may be used to attach components to the propellant tank. Buna-N "O"-ring material is generally considered the best for this application.

A definite consideration in certain applications of the SSRCS is rapid transfer of heat to the propellant which is closely related to propellant tank design. Use of aluminum for tanks provides efficient and effective heat transfer from the surroundings to the propellant surface. In cases where the propellant tank will be exposed to the sun, or near other heat producing satellite components, it may be desirable to build the propellant tank in a pancake configuration which will expose a maximum surface area to sunlight or other heat source. It is also desirable to maintain good thermal contact with the satellite structure if heat is to be obtained. In this case, the propellant tank should be designed with a large contact area to a mounting panel aboard the satellite. Such a propellant tank may have a large flat bottom if it is of cylindrical shape. A large contact area greatly improves thermal transfer from the surroundings to the propellant tank; as described in more detail in Paragraph 1.9 which discusses heat transfer, and Paragraph 2.11 which discusses experiments and analyses in heat transfer.

1.4 Propellant Packaging

The subliming solid propellant can be packaged in one of three basic forms:

a. Solid crystalline mass

b. Highly pressed powder

c. Loose granular powder

1.4.1 Solid Crystalline Mass

The solid crystalline form of subliming solid propellant is produced by recondensing the propellant within a propellant tank of any shape or size. The propellant tank to be filled is partially immersed in a dry ice alcohol solution; and connected to a container of the subliming solid at, or slightly above, room temperature. The line connecting the two tanks is heated to prevent recondensation in the line which would plug the flow. The tank to be filled is gradually immersed deeper into the cold liquid solution until a solid crystalline mass is built up inside. Slow recondensation results in a very high density which may be, as high as, 0.05 pound per cubic inch. Rapid recondensation would produce voids in the propellant, and as a result, propellant density would be lower than with slow recondensation. The recondensation process may take several days to fill a moderate size propellant tank, as described in more detail in Paragraph 2.15.

The main advantage in using a recondensed cake is that it yields the minimum required tank volume. Basically, the disadvantage in using a recondensed cake is a relatively low available surface area for sublimation. In the case of SUBLEX B, the reduction in surface area will lead to subliming rate limitations which lowers the thrust level.

1.4.2 Highly Pressed Powder

The pressed subliming solid powder propellant form is produced by placing granular propellant in a die and applying extremely high pressure. Approximately 10 to 50 tons pressure per square inch is required to produce structurally integral grains. Theoretical propellant density up to 90 percent may be obtained by using this process.

The basic advantages of a pressed grain are high density and minimum propellant tank volume. The pressed grain is somewhat porous and, therefore, has a higher sublimation surface area than the recondensed cake. However, since a special die is required to make a propellant grain, specific tank configuration is not as flexible as either the recondensed grain or the granular powder form. SUBLEX A is also more difficult to produce in a pressed form, since exposure to air while pressing the propellant must be avoided.

1.4.3 Loose Granular Powder

The loose granular form of propellant is the most commonly used of the three forms, because of handling and pouring ease. This form may be easily packaged in any propellant tank configuration and has a high available sublimation surface. However, density is only 1/2 to 2/3 that of the theoretical density which requires a higher propellant tank volume than the pressed powder form. Both SUBLEX A and SUBLEX B are available in the granular form.

1.5 Valves

1.5.1 Requirements

The basic requirements for a subliming solid propellant valve are:

a. Low leakage

b. Minimum size

c. Minimum weight

d. Minimum operating power

Since the SSRCS is a low thrust rocket, small solenoid valves may be used to control propellant flow. These valves use either direct pull solenoids or electric torque motors to open and close propellant ports, and normally require an operating voltage of 18 to 30 volts D.C. which is the standard voltage range for a typical spacecraft. The actual physical size of the valve is primarily dependent upon the flow rate desired, and the available vapor pressure of the subliming solid propellant. These factors are, in turn, dependent upon the desired thrust level and the temperature at which the subliming solid propellant will operate. A further discussion on valve size is found in Paragraph 2.14.

1.5.2 Configuration

The most common solenoid valve configurations are the coaxial, poppet, and shear seal types.

1.5.2.1 Coaxial Valve

The coaxial valve is the most common solenoid valve used to control propellant flow in both liquid and gaseous rocket systems because of its low weight, small volume, and minimum pressure drop. They are manufactured in a wide variety of sizes by many different manufacturers. A coaxial valve is shown schematically on Figure 1-3. When energized, the solenoid coil creates a magnetic field that pulls back the plunger, which in turn, pulls the poppet away from the seat. The term coaxial valve is derived from the fact that the vapor or gas flows, coaxially or symmetrically, straight through the valve body.

Coaxial valves are constructed in both hard and soft seat configurations. Both the poppet and seat material are nonresilient in the hard seat valve. The materials often used are tungsten carbide, anodized aluminum, saphire, and stainless steel. The soft seat valve usually provides a tighter seal under conditions where slight contamination exists. The hard seat valves are particularly useful where extremely long life is desired, or where corrosive fluid is being handled which deteriorates either Teflon or rubber. Hard seat valves must be carefully lapped to prevent leakage.

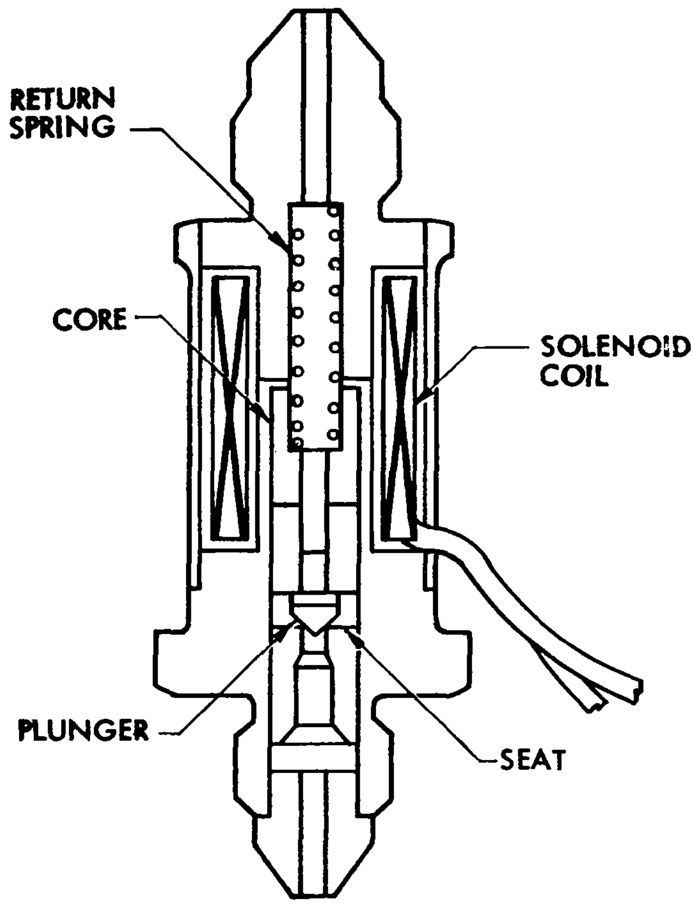


FIGURE 1-3. TYPICAL COAXIAL VALVE

1.5.2.2 Poppet Valve

The poppet valve, shown on Figure 1-4, is similar to the coaxial type in its poppet and seat arrangements except it is built in a right angle configuration. The orifice area in the poppet valve, which is capable of being opened, is somewhat larger than in the coaxial type. Poppet valves are also constructed of both soft and hard seat configurations. The basic advantage of the poppet valve is its large orifice area; however, under conditions of contaminated flow, the large area allows a greater degree of fouling.

1.5.2.3 Shear Seal Valve

The shear seal valve, illustrated on Figure 1-5, is essentially a gate valve where the flow seal is accomplished by means of a sliding plate rather than a plunger which pulls away from a seat. This type of valve is especially attractive under conditions of flow contamination, since the seat wipes clean with every actuation and particles cannot lodge in the seat area. Very careful lapping of the shear seal is required to obtain a tight seal. The shear seal valve is generally not as compact as the coaxial valve, and there is some limitation to the orifice area which may be opened at low electrical power.

1.5.3 Valve Choice

There are several SSRCS characteristics which affect the choice of a propellant valve. The first of these is particulate matter in the exhaust stream. The subliming solid propellant tank is fitted with a filter assembly, discussed in detail in Paragraph 1.6, which is designed to separate the vapor from the solid particles. The degree of filtration is usually on the order of 40 microns, which has proven to be very reliable, and will not plug under either pulsed or continuous flow conditions; however, some solid particles may find their way into the valve. As a result, the valve must be capable of ingesting particles up to 40 microns in size without failure or excessive leakage. In general, soft seat coaxial and shear seal valves are capable of particulate flow of this size without detrimental effects.

The second characteristic of subliming solid operation, which influences valve choice, is the possibility of recondensation in or around the valve seat during off periods. Such recondensation, as described in Paragraph 1.10, is due to slow cyclic temperature variations of the valve during normal storage or flight conditions. Recondensation can be detrimental to coaxial and poppet type valves due to

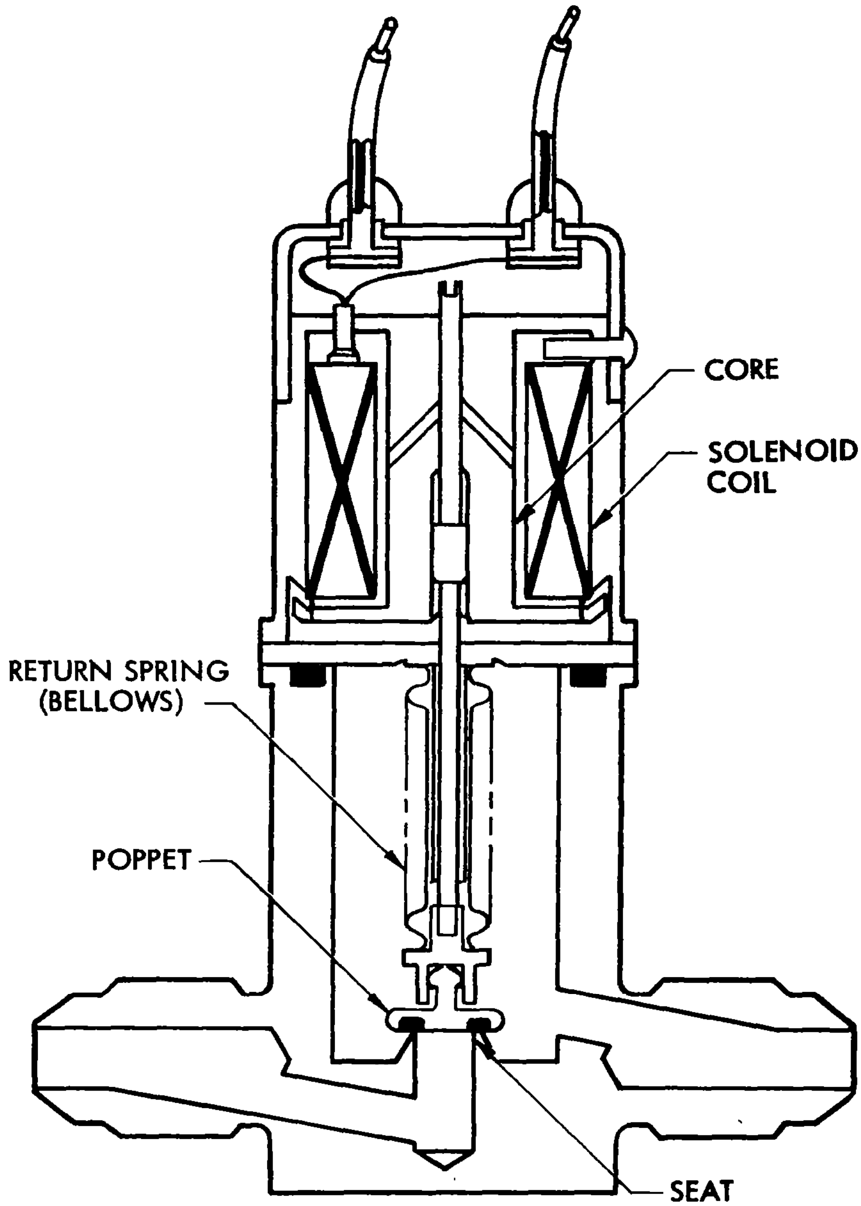


FIGURE 1-4. TYPICAL POPPET VALVE

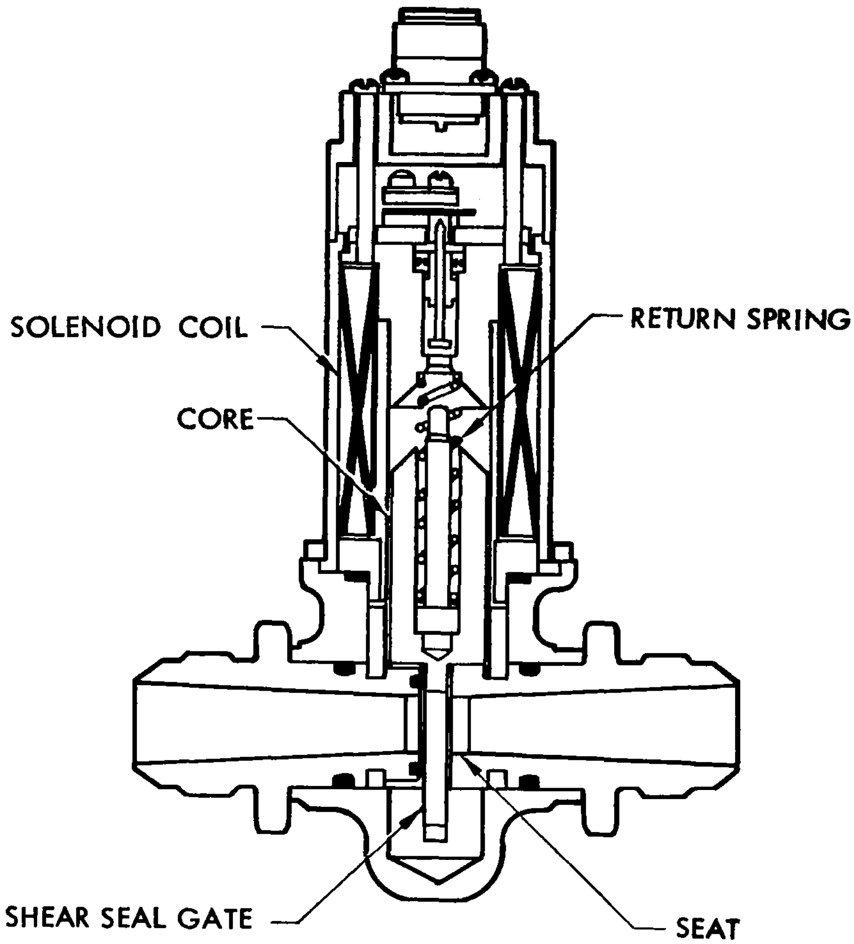


FIGURE 1-5 TYPICAL SHEAR SEAL VALVE

formation of small slugs of recondensed propellant which, upon valve actuation, may lodge in the valve seat area during the initial on period, preventing effective closing. Such recondensation apparently does not affect the shear seal valve as severely, due to the self wiping motion of the seat. Any small recondensation film is wiped off and carried away by the flow and thereby removed from the sealing area. Small particles that might remain in the valve are also wiped aside during the closing action of the valve and, therefore, do not interfere with the seal.

A third important characteristic is the relatively low pressure at which the propellant valves operate. This is one of the important SSRCS characteristics, which is a distinct advantage, resulting in reliable and efficient valve operation. Not only does the valve open against smaller forces, and therefore at lower electric power; but the leakage through the valve, when it is closed, is considerably lower than typical cold nitrogen gas systems. Valves leak on a gas volume per unit time basis; therefore, mass flow loss from such a leak will be dependent upon the density of the gas, which is a direct function of gas pressure. A subliming solid system operating at 15 psia will have a mass leakage rate of one two hundredths (1/200) of that of a cold nitrogen system operating at 3,000 psia. Thus, for long operating duration, the propellant loss from a subliming solid system will be better than two orders of magnitude less than a comparable nitrogen system. This is particularly important in missions of over six months duration.

1.5.4 Valve Materials

Material incompatibility problems are the most likely cause of valve failure in a Subliming Solid Reaction Control Rocket. If any corrosion of the internal valve parts occurs, buildup of corroded metal could find its way into close fitting areas or to the valve seat, causing valve seizing or leakage. The only instances of valve failure in experimental subliming solid systems have always been traced to valve corrosion. The choice of materials in the valve is, therefore, exceedingly important and critical. Material compatibility with propellant is discussed in detail in Paragraphs 1.2.4 and 2.5. However, a few general rules can be discussed with regard to the application of materials in valves. In particular, copper and copper bearing metals must be avoided as copper reacts rapidly with both SUBLEX A and SUBLEX B. Materials considered suitable for the construction of internal parts of the subliming solid valve are: Stainless Steel No's. 303, 304, 316, and 321; Aluminum; Buna-N; and Teflon. See Table 2-3 for complete tabulation of materials compatible with subliming solid propellant.

1.5.5 Valves Suitable for SSRCS

There are a number of small electrically operated valves available which can be applied to the SSRCS. A listing of these valves, with their characteristics and comments as to applicability and availability, are shown on Table 1-1.

An Eckel all welded, stainless steel valve (Part Number AF 77C-A119) has been chosen for flight on a SSRCS. This is a two watt, coaxial solenoid valve weighing 0.15 pound with orifice diameter of 0.05 inch. A Parker valve, coaxial type, (Part Number PTS-5640053) has been chosen for flight aboard a hybrid hydrazine subliming solid system. This is an alphatized stainless steel valve weighing 0.1 pound. It operates at one watt with an orifice diameter of 0.03 inch. A Whittaker valve (Part Number 221075-2) performed satisfactorily on a subliming solid laboratory test model. This valve is a bidirectional, shear seal valve with a 10 watt torque motor actuator. The valve weighs 0.35 pound and has an effective orifice diameter of 0.060 inch. Whittaker Part Number 221315 was also delivered on a subliming solid laboratory test model. This is a two watt, coaxial valve weighing 0.35 pound, with an orifice diameter of 0.1 inch. This valve was unsatisfactory in its operation, due to a copper underplating in the interior surfaces of the valve which corroded and seized the valve during a long storage period. Whittaker (Part Number 127675) is a low power shear seal valve currently under development by NASA. This valve has an effective orifice area of 7.5×10^{-4} square inch with a discharge coefficient of 1. It should be an excellent choice once development is complete.

Three valves were tested during the experimental program. Details of these tests are described in Paragraph 2.1.4. The valves tested were: An Eckel coaxial type, Part Number AF 42-562; a Valcor shear seal type, Part Number V-5000; and a Carleton poppet type, Part Number 1755001-3. The valves were tested under severe conditions, with large particle flow purposely forced into the valve inlet. The results of these tests indicated that both the coaxial and shear seal valves are capable of withstanding severe particulate flow without catastrophic failure. The poppet valve did not perform satisfactorily in this severe test due to excessive leakage early in the test.

1.5.6 Valve Application

The choice of a valve for a SSRCS is dependent upon many factors. Four of the main factors are:

VALVE TYPE	WEIGHT (POUNDS)	POWER REQUIRED (WATT)	MATERIAL	ORIFICE DIAMETER (INCHES)	AVAILABILITY	RECOMMENDATION
COAXIAL						
Eckel P/N AF77C-A119	0.15	2.0	All welded, Stainless Steel	0.05	8 Wks. ARO	Acceptable for Flight Service
Parker P/N PT5-5640053	0.10	1.0	Alphatized Stainless Steel	0.03	12 Wks.	Testing with SUBLEX A required
Whittaker P/N 221315	0.35	2.0	Nickle plated Mld.Steel Copper under-plating	0.10		Valve failed in test
Eckel P/N AF42-562	0.15	7	Mild Steel, Stainless Steel, Brazed construction	0.05	6 Wks. ARO	Materials not compatable after long storage

TABLE 1-1

VALVE CHARACTERISTICS AND AVAILABILITY

Sheet 1 of 2

VALVE TYPE	WEIGHT (POUNDS)	POWER REQUIRED (WATT)	MATERIAL	ORIFICE DIAMETER (INCHES)	AVAILABILITY	RECOMMENDATION
POPPET						
Carleton P/N 1755001-3	0.20	2.2	Stainless Steel	.050		Valve failed in test
SHEAR SEAL						
Whittaker P/N 221075-2	0.35	10	Stainless Steel	0.060		Good performance after 15 month storage on Lab. Test model
Whittaker P/N 127675	0.25	2	Stainless Steel	.03	8 Wks.	Testing required
Valcor P/N V-5000	0.50	10	Aluminum and mild steel	.10	4 Wks.	High leakage

TABLE 1-1

VALVE CHARACTERISTICS AND AVAILABILITY

Sheet 2 of 2

a. Availability

b. Orifice Size

c. Operating Power

d. Volume and Weight

1.5.6.1 Availability

In most cases, a valve should be chosen which may be purchased as an off-the-shelf item, resulting in lower cost and faster delivery time. Most of these valves will have had previous usage and testing (perhaps on flight vehicles) on which to judge reliability and conformance to individual requirements. Valve development should only be undertaken when an existing valve cannot be used without degrading mission objectives.

1.5.6.2 Orifice Size

The primary requirement in the sizing of a valve is the determination of orifice size. Orifice size is established by defining the desired thrust level, the propellant to be used, and the expected operating temperature range for the control rocket. The temperature range, in turn, is determined not only by the environmental temperature, but also by the pulse duration of the SSRCS duty cycle. See Paragraph 3.3.7 for a detailed discussion of valve sizing.

1.5.6.3 Operating Power, Volume and Weight

Once the valve orifice size has been determined, the specific valve can be chosen. In the case of relatively large orifice sizes, the field is usually considerably narrowed. Electrical operating power, volume, and weight of the valve are then optimized for any given application. Optimization of operating power, volume, and weight are dependent on the criticality of each parameter.

1.5.7 Valve Type

The type of valve chosen is dependent upon specific characteristics of the mission to be undertaken and its duration. The degree of expected particulate flow and the probability of recondensation in the valve are the most important characteristics with regard to valve choice. In most applications, the coaxial valve can be used with a high expected reliability, however, it may be that the shear seal type is inherently better for the subliming solid operation. Further valve evaluation will be required before this is definitely established. The type of valve seat material, whether soft seat or metal-to-metal seat, is the main determination of valve life. In general, soft seat valves will give better seals under contaminate flow, but are somewhat prone to deformation with age.

1.6 Filters

1.6.1 Requirements

The key to SSRCS operation is the efficient and complete separation, performed by the filter assembly, of the solid from the vapor. The filter assembly is generally located at the outlet of the propellant tank and prevents any small particles of subliming solid material from entering propellant lines, valves, and exhaust nozzles.

The degree of filtration must be fine enough to prevent any possible seizure of the propellant valve or plugging of small critical orifices. However, the filtration must not be so fine as to cause excessive pressure drops across the filter assembly or be prone to plugging. The filter assembly should be small enough to allow easy packaging into the propellant tank but still have sufficient flow area or diameter so that compacting of powder particles does not totally block off the flow. For example, a filter or primary filter should not be located in a small diameter propellant line at some distance from the propellant tank because small subliming solid particles could build up on the line and compact to form a flow restriction.

1.6.2 Configuration

The subliming solid filter can take many different forms depending on the degree of filtration required which, in turn, is dependent upon the specific mission. The filters most commonly used in laboratory and flight prototype SSRCS thus far have been either 200 mesh or 40 micron final filtration; and the valves which have been exposed to filtration of this degree have functioned properly without excessive leakage, seizure, or failure, of any type. It can be concluded from this experimental evidence that, at least for the coaxial and shear seal valves tested, 40 micron filtration is satisfactory. However, a series filter arrangement was empirically designed and tested to prevent massive packing at the screen surface. A typical series filter consists of a 50 or 100 mesh screen, a spacer followed by a 200 mesh screen, and a second spacer followed by a 40 micron filter screen. The purpose of the larger mesh screens is to keep the bulk of the propellant particles (particularly in the case of loose granular packing) away from the fine filter screen. The relatively large pore sizes of the 50, 100, and

200 mesh screens prevent dense packing and, therefore, flow obstruction. Tests using such filter arrangements are described in detail in Paragraph 2.12.

1.6.3 Design

The design of the filter assembly is primarily dependent upon the design of the propellant tank outlet and the available space. The outside surface area of the filter should be as large as possible to reduce the possibility of packing the propellant around the filter assembly inlet, and to minimize any pressure drop across the filter assembly. Filter screens are available in a wide variety of sizes and shapes.

Disc type filter assemblies were used in SSRCS development. These assemblies may be cut from conventional stainless steel or certain commercially available cylindrical elements, however, stainless steel screen is preferred due to its excellent compatibility with both SUBLEX A and SUBLEX B propellant. The 40 micron stainless steel screen also comes in a variety of shapes. A screen configuration with the maximum flow area should be used since the passage area in a 40 micron screen is considerably less than coarser screens. An example of this type filter is the Bendix series Poromesh, which is a disc type filter, made from a rippled stainless steel screen providing a very large flow surface area for any given flow tube diameter. This type of filter has been used extensively in SSRCS laboratory test models and flight prototypes.

The pressure drop across a standard filter assembly can be calculated based upon the desired flow. The filter assembly for any given application must be tested to assure that excessive pressure drop does not occur, either initially or during expulsion of the propellant, during the mission. A typical filter design which has been applied to a flight prototype system is shown on Figure 1-6.

1.7 Nozzles

1.7.1 Requirements

The function of the exhaust nozzle is to accelerate the exhaust vapor to the highest possible velocity. There are many texts available which treat the theory of exhaust nozzle operation, such as, Rocket Propulsion Elements, Third Edition, by G. E. Sutton, published by John Wiley and Sons, New York.

The equations governing nozzle design are relatively straightforward since the SSRCS is a cold gas system requiring no chemical reaction in the exhaust nozzle. The specific impulse of the SSRCS operating in a vacuum (infinite pressure ratio) is a

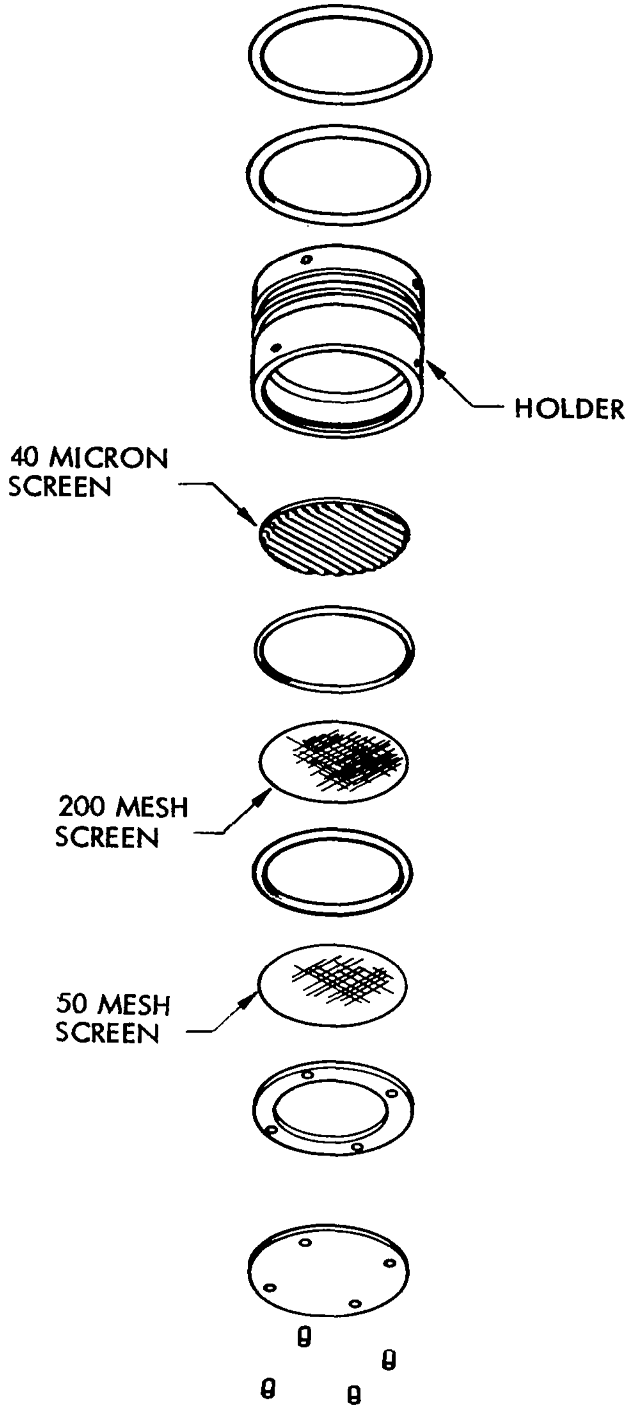


FIGURE 1-6. TYPICAL FILTER DESIGN

function of nozzle area ratio, vapor specific heat ratio, molecular weight, and temperature. Figure 1-7 is a typical plot of the specific impulse of SUBLEX A vs. nozzle area ratio.

Since the molecular weight and the specific heat ratio for the subliming solid fluid are fixed, it can be seen that the specific impulse is dependent only upon the fluid exhaust temperature, the area ratio of the nozzle, and the pressure ratio of expansion. Since most SSRCS are operated in the vacuum of space, the pressure ratio is infinite. Therefore, in most instances, the exhaust temperature is also fixed; the only variable is the area ratio.

From Figure 1-7 it can be seen that the area ratio should be as large as possible, and typical space nozzles are designed with area ratios of 100:1 or more. There is one factor which is not covered on Figure 1-7 -- momentum loss due to boundary layer buildup which occurs in all rocket nozzles and can be a severe problem for small nozzles.

Boundary layer buildup is due to gas friction on the nozzle walls. In large nozzles, due to the fact that they generally operate at high pressures and are of large diameter, the boundary layer effect is small and can be ignored. However, in small nozzles, the dimensions of the nozzle may be such that the boundary layer thickness becomes relatively large compared with a typical dimension in the nozzle. In addition, at low vapor pressure or low operating pressures, the Reynolds number of the fluid is low. The combination of these factors results in appreciable momentum loss and, therefore, decrease in nozzle efficiency in small exhaust nozzles used on SSRCS. A good deal of experimental work must still be done to define the magnitude of this loss. However, the limited amount of experimental work which has been performed thus far indicates that nozzle efficiencies of as low as 70 percent can occur at thrust levels down to 10^{-4} pounds. At thrust levels of 10^{-2} pounds, nozzle efficiency is approximately 90 percent. With higher thrust levels (10 pounds of thrust or more) nozzle efficiency is generally 98 percent with regard to momentum loss.

The conventional nozzle configuration is a convergent/divergent section with a conical divergent section of 30 degrees included angle. It has been experimentally demonstrated on large rocket systems that in most cases this is an optimum angle-- the larger the angle of the divergent section, the greater the momentum losses due to side velocity vector as the gas escapes from the nozzle. However, excessively

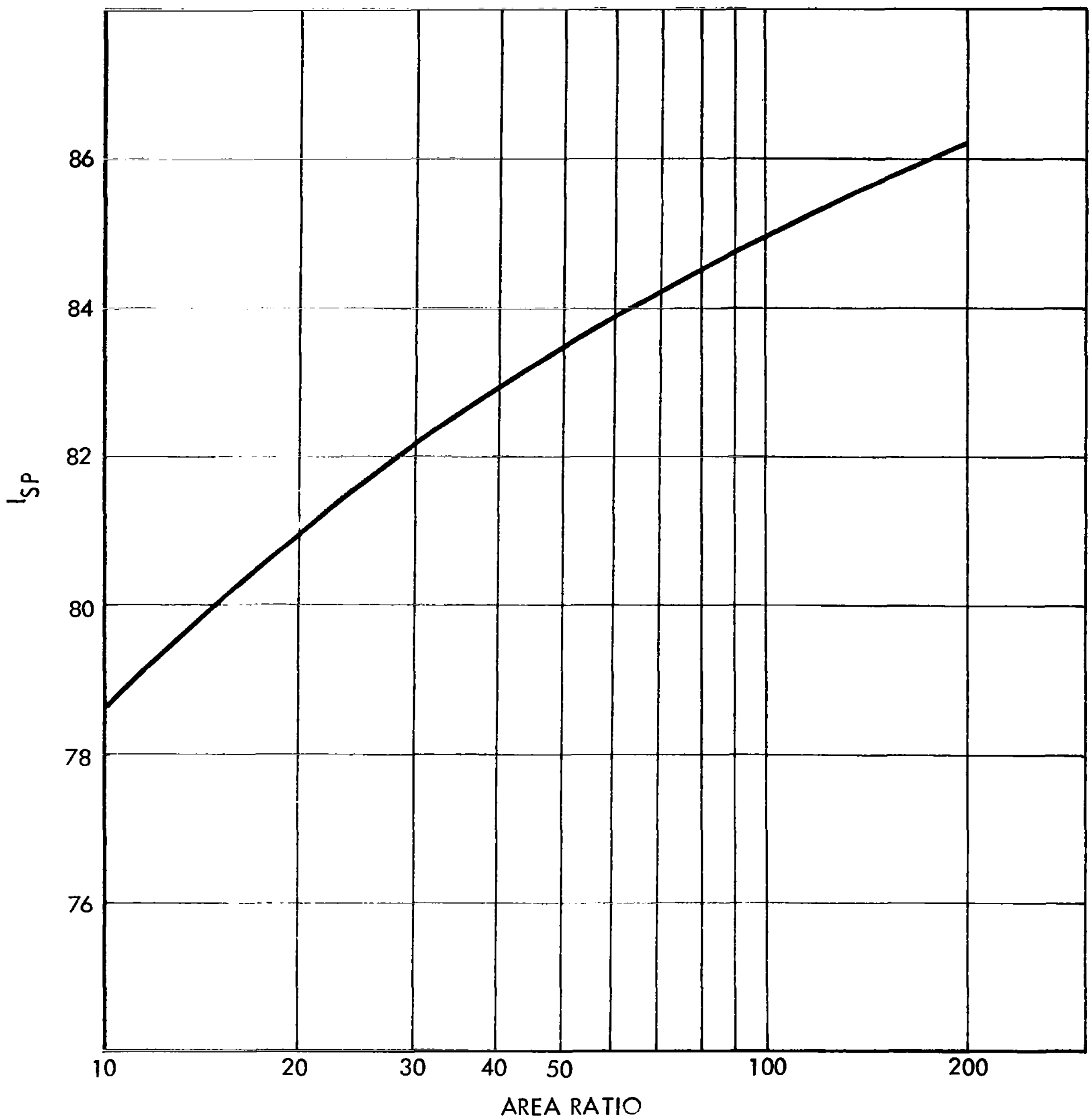


FIGURE 1-7 SPECIFIC IMPULSE AS A FUNCTION OF NOZZLE DESIGN

small nozzle angles lead to undesirably long nozzles with associated momentum boundary layer loss. Since momentum boundary layer is more significant in the case of small nozzles than in large nozzles, a larger divergent angle is desired. Considerable experimental work must be done before optimum divergence angles can be defined. At the present time, a divergence angle of between 30 and 40 degrees included may be considered desirable.

1.7.2 Nozzle Design

A typical nozzle is shown on Figure 1-8. The nozzle material may be of any type compatible with the exhaust vapor. In general, exhaust nozzles are machined from aluminum because of machining ease, compatibility, and low weight. The nozzle is generally machined with a relatively thin wall down to approximately the throat section and a threaded section beyond as illustrated on Figure 1-8. The nozzle is usually threaded into some sort of a mounting block which is in turn directly connected to the SSRCS, the propellant valve, or to a propellant line. The area ratio of the exhaust nozzle is usually from 50:1 to 100:1. The approach section of the nozzle is not particularly critical. Conical approach sections of 90 degrees to 120 degrees have been used. It is generally desirable to have the approach section to the nozzle a factor of two or more larger in diameter than the nozzle throat, but the nozzle throat itself should be well rounded.

The sizing of a nozzle is dependent upon the desired thrust level and the vapor pressure of the subliming solid propellant. The fluid pressure at the nozzle inlet is, in turn, a function of the temperature of the propellant during operation and any pressure drops which may occur or be built into the SSRCS. The pressure drop through the filter, propellant valve, any restricting orifices, and the exhaust line must be considered. The sizing of the nozzle throat is discussed in further detail in Paragraph 3.3.8.

1.8 Miscellaneous Parts

In addition to the major subliming solid components discussed thus far, there are a number of auxiliary parts which are required to assemble a typical SSRCS. The most important of these are:

a. Propellant lines

b. "O"-rings

c. Joints

d. Heaters (See Paragraph 1.9)

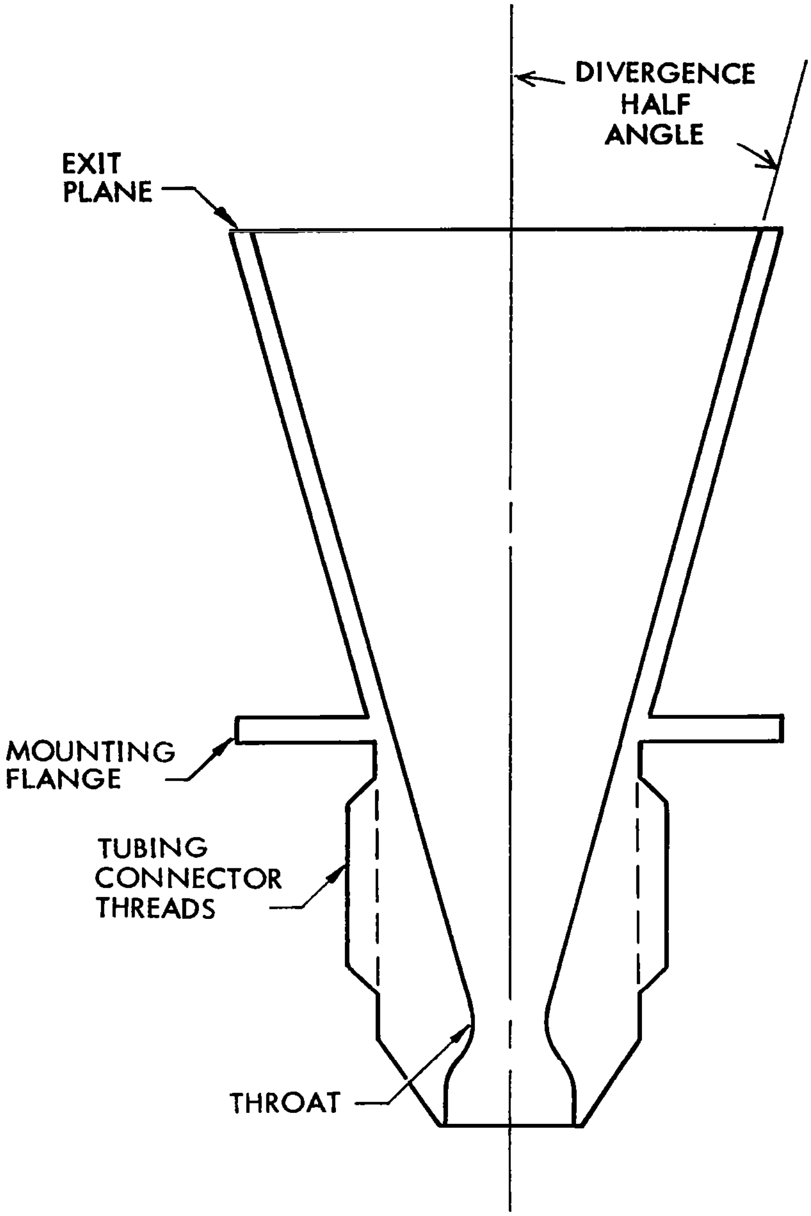


FIGURE 1-8. TYPICAL SUBLIMING SOLID CONTROL ROCKET NOZZLE

1.8.1 Propellant Lines

In some SSRCS designs, a propellant line, up to several feet in length, is used to connect the propellant tank to the valve and/or the valve to the exhaust nozzle. The only basic requirement for the propellant line is that it be of a sufficiently large diameter in order to prevent excessive pressure drops which restrict the propellant flow, and that line material be compatible with the subliming solid exhaust vapor. Many different materials can be considered; such as, plastic, rubber, and aluminum and stainless steel metals. Aluminum lines are most commonly used in SSRCS because the material is lightweight, readily available, easily formed into desired shapes, and easily flared for connection.

If the exhaust nozzle is located remotely from the propellant valve, the rise time and tail-off of the thrust pulse delivered from the nozzle will be affected. This is due to the fact that the interconnecting propellant line has a finite volume which must be filled and emptied. In certain cases where rapid pulse response is required, such delay in rise time and tail-off could be detrimental. When this is the case, it is desirable to size the exhaust line diameter at absolute minimum. The line must still be of sufficient diameter to prevent flow restriction but should be sufficiently small to minimize hold up volume. The equations which govern rise time and tail-off are as follows:

Rise Time

$$t_{90\%} = \frac{2.3}{c}$$

$$c = \frac{12g\, R'\, T_c\, A_t}{c^*\, M\, V}$$

Where:

g = $32.2\ \text{ft/sec}^2$

R' = universal gas constant, $1545 \dfrac{\text{ft-lb}}{\text{lb mole }^\circ\text{R}}$

T_c = vapor temperature, °R

A_t = throat area, in^2

c^* = characteristic velocity, ft/sec

M	=	vapor molecular weight, lb/lb-mole
V	=	line or chamber volume, in^3

The above equation assumes choked flow into and out of the chamber.

<u>Tail-Off Time</u>

$$t_T = \frac{V}{A_t} \; \frac{\left(\frac{2}{K-1}\right)\left(\frac{K+1}{2}\right)^{\frac{K+1}{2(K-1)}}\left[\left(\frac{P_o}{P_1}\right)^{\frac{K-1}{2K}} -1\right]}{12\left(g\,K\,\frac{R'}{M}\,T_o\right)^{1/2}}$$

Where:

t	=	tail-off time after valve closure, seconds
V	=	line or chamber volume, in^3
A_t	=	choked discharge orifice throat area, in^2
g	=	32.2 ft/sec^2
K	=	specific heat ratio of gas or vapor
R'	=	universal gas constant, 1545 ft/lb/lb-mole °R
M	=	gas or vapor molecular weight, lb/lb-mole
T_o	=	gas or vapor temperature, °R
P_o	=	Initial pressure of gas or vapor in line or chamber, psia
P_1	=	Instantaneous pressure of gas or vapor in line or chamber, psia

The above equation neglects heat transfer, friction pressure drop, and recondensation either before or during discharge.

These equations can be used to determine the characteristic rise time and tail-off for any system with a finite volume between the propellant valve outlet and the nozzle inlet.

A critical factor in the use of propellant lines in the SSRCS is the elimination of propellant recondensation. If a propellant line several feet long is routed through a space vehicle, it may pass through regions in which the environmental temperature is significantly below the temperature of the propellant in the propellant tank during operation. In these cases, the vapor pressure in the line must either be sufficiently low to prevent recondensation, or the propellant line must be heated. This problem of recondensation is discussed in more detail in Paragraph 2.10.

1.8.2 "O"-Rings

"O"-rings, which are used whenever two parts must be bolted together, are the standard method of connecting parts in a pressurized system. The "O"-ring provides sealing against a smooth surface to prevent gas leakage past small deformities in the two surfaces. The proper application of "O"-rings can be obtained from standard documents which are published by "O"-ring manufacturers. A typical document is:

Parker "O"-Ring Handbook
Catalog 5700
Parker Seal Company
Cleveland, Ohio

"O"-rings have been used extensively in the fabrication of typical laboratory tests and flight prototype SSRCS. They are used, for example, to seal the filter assemblies into the propellant tanks, the valves into the filter assemblies, and the nozzles into their mounting blocks. The basic requirement for an "O"-ring, aside from standard application and usage, is that the "O"-ring material be compatible with both SUBLEX A and SUBLEX B. Teflon is also completely compatible with the subliming solid propellants and may be used as an "O"-ring or seal material. The basic difficulty with Teflon is that it is not as flexible as rubber and, therefore, will not crush as well to make a good static seal. Teflon will cold flow with time and a seal can gradually open, thus allowing a small amount of leakage. However, it is more resistant to wear. For this reason, rubber "O"-rings are desirable in the moderate temperature environments usually encountered for SSRCS static seals. Teflon is used for SSRCS seal material (due to its durability and inert qualities) only in valves where seating and unseating can cause wear.

1.8.3 Joints

In addition to "O"-ring seals at joints, several other joint types (generally metal-to-metal) are also applicable to the SSRCS. The most common of these is the conventional AN standard nut and flare type joint. This joint is especially dependable when aluminum tubing is used, since aluminum is sufficiently soft to make a good seal on the connecting fitting. These joints present no problem as the subliming solid operates at low pressure. Standard parts of flared or fitted shapes are used to make the joint following standard practices for tube flaring. Other standard joining techniques such as swage lock, welding, and brazing may also be used for the SSRCS.

1.9 Heat Transfer

1.9.1 Requirement

The transition from a solid to a vapor state requires absorption of heat by a molecule in order to increase molecular energy. Generally, compounds which sublime or vaporize to a low molecular weight vapor have relatively high latent heat. SUBLEX A requires 782 BTU per pound, and SUBLEX B requires 877 BTU per pound.

Therefore, the successful operation of a SSRCS must include some provision for transferring the heat required for sublimation to the subliming solid propellant. The source of heat may be either passive or active. Passive sources are classified as heat from the surrounding structure of components, a radioisotope heat slug, or solar heating. Active heat is electrical resistance heating which is either provided on command or by means of a thermostat system.

Approximately ten thermal watts of heat are required to maintain a thrust level of 0.001 pound. The relationship between heat required and thrust is linear. In other words, a thrust level of 10^{-2} pounds requires 100 thermal watts of heat to cause sublimitation. This heat transfer process may occur, in part, after the motor is turned off, in which case the heat warms the propellant tank back to equilibrium temperature. During sublimation, heat required at the propellant surface is provided by:

a. Heat transfer from another solid in contact with the propellant

b. From the surrounding gas

c. From within the subliming particle.

Therefore, the specific heat of the subliming solid and/or the propellant tank instantaneously provides the heat of sublimation for the propellant during flow conditions. As the temperature of the propellant tank and the subliming solid particle drops slightly, due to sublimation, heat transfer then begins from the surroundings or from the heat source. Thus, a SSRCS may operate for several seconds at a moderately low thrust level before any appreciable heat transfer occurs from the surroundings.

1.9.2 Heat Transfer Methods

In most cases, the thrust level/duty cycle relationship for a SSRCS is sufficiently low so that heat may be readily obtained from the vehicle structure or components surrounding the propellant tank. This is the most desirable case as it does not involve a special design with regard to solar radiation, electrical heat, or the use of a radioisotope heat slug.

The heat transfer process can be divided into two steps. The first involves the transfer of heat from the surroundings to the propellant tank; the second involves the transfer of heat from the propellant tank to the propellant. In both cases, good physical contact must be made to realize the maximum thermal conductivity of the system.

Tank mounting brackets, which provide good thermal shorts to the surroundings, are desirable to maintain good heat transfer from the surroundings to the propellant tank. Such mounting surfaces might take the form of flat-bottomed, cylindrical propellant tanks or special large area mounting brackets which can be used as heat conductors. The propellant tank should be constructed of a good thermal conductor to assist in distributing the heat around and through the propellant tank, thus providing maximum heat conduction from the tank to the propellant itself. When good thermal contact cannot be made with the surroundings, it is necessary to rely strictly upon transfer by radiation. In these cases, the propellant tank surface is generally blackened to increase heat absorption.

Heat transfer between the propellant tank surface and the propellant is accomplished by radiation, convection, and conduction. Of the three, direct conduction is the most efficient. Therefore, it is desirable to keep the propellant in direct contact with the propellant tank, which, in spinning satellites, is accomplished automatically. In cases where no artifical force field is available, spring loaded devices may be considered. It is also possible to package the propellant in a honeycomb structure which provides the maximum propellant contact with metal surfaces. Convective heat transfer primarily occurs during the SSRCS on periods when propellant vapor is flowing through the propellant tank and out through the valve. Propellant tank designs with large convective flows will assist in heat transfer. Heat transfer by radiation is the least efficient of the three methods. In certain cases, when the thrust level or duty cycle is sufficiently low, heat transfer by radiation alone may be sufficient. It is desirable to construct the propellant tank in such a way that maximum heat transfer results from convection and, most importantly, from direct conduction.

Propellant form also plays a part in heat transfer. In general, a propellant form with a high sublimation surface area will yield the highest possible propellant flow. This is because heat transfer from the interior of the propellant particle to the surface is more rapid in numerous small particles than in one large propellant particle. In addition, the numerous small particles will maintain better contact with the propellant tank wall than will one large propellant crystal.

Heat transfer in a typical SSRCS configuration with analysis and supporting experiments is described in detail in Paragraph 2.11. The heat transfer characteristics of any system will depend upon the specific conditions of the mission. However, the experiments did show that, for radiation alone to a spherical tank six inches in diameter, a thrust level/duty cycle relationship as high as 10^{-3} pounds was typical. (10^{-3} pounds requires an average heat input of 10 to 11 thermal watts.) This means that it is possible to operate at 0.01 pound thrust level on a 10 percent duty cycle or at 0.1 pound thrust level on a one percent duty cycle, etc. The thrust level/duty cycle relationship may be increased if a good conductive heat path is provided through the mounting brackets to the propellant tank. Under certain conditions, a thrust level/duty cycle relationship as high as 3×10^{-3} pounds is possible with approximately a 20°F temperature difference between the propellant and the surroundings.

Higher thrust levels may also be obtained on a continuous basis if unusually large sources of heat are available. In certain instances, radioisotope heating slugs may be placed in contact with or inside the propellant tank. Such slugs emit approximately 100 watts allowing a continuous thrust level of 10^{-2} pounds to be obtained. Heat sinks may be attached to the propellant tank to allow relatively large or long pulses to be achieved at thrust levels between 10^{-2} and 10^{-3} pounds. It is also possible to locate the propellant tank where it receives a large quantity of solar radiation. In this case, the propellant tank is designed with a large exposed surface area which has a high absorption coefficient. The propellant tank may also be located near some heat producing system aboard the satellite or space vehicle. Momentary electric heating at high power levels may be considered in those cases where relatively short pulses at high thrust level are necessary.

In certain special cases, it is possible to use the SSRCS aboard the space vehicle as a heat sink or refrigerator for electronic components which are constantly dissipating certain amounts of thermal energy. If the SSRCS is used in this manner, it must operate on a continuous duty cycle.

1.9.3 Heat Transfer System Considerations

Heat transfer plays an important part in SSRCS sizing for any specific mission. In particular, a knowledge of the heat transfer characteristics of a specific system and the thermal environment is important in choosing the proper orifice size for a propellant valve. The nozzle throat diameter will be similarly sized. Once the thrust level/duty cycle and maximum pulse duration is established, an estimate can be made as to the temperature drop the propellant will experience during operation. The minimum temperature will establish the minimum vapor pressure to be expected during operation which is in turn used to calculate the required valve orifice size and nozzle size for the minimum allowable thrust set for the system. A further discussion of this relationship is contained in Paragraphs 3.3.7 and 3.3.8.

The major limitation to obtaining high thrust from the SSRCS is the availability of sufficient heat. However, there may be certain instances in which a high thrust and relatively long pulse may be required which sublimation alone cannot provide. A specific example is the use of the SSRCS to provide initial vehicle acquisition. In such cases, it is possible to load the propellant tank with nitrogen at some moderate pressure of 100 to 300 psia and use this topping nitrogen to achieve a high thrust level/duty cycle relationship. The nitrogen operates in a blow-down mode and may provide continuously high thrust until the nitrogen is depleted and subliming solid vapor pressure is reached.

1.10 Recondensation

The sublimation of subliming solid propellant is a reversible process. Any stable subliming solid material which is vaporized upon heating will recondense when recooled. Therefore, when components or lines, which are attached to the propellant tank and are in open communication with the propellant vapor, are cooled below the propellant temperature, the propellant vapor may recondense to a solid at these points. The degree, speed, and density of the recondensation will vary depending on specific conditions and the temperature differences. The tests performed by Rocket Research Corporation are outlined in Paragraph 2.10. The effect of recondensation on various system components is presented in the following paragraphs.

1.10.1 Filter Recondensation

As a result of filter and recondensation tests, it has been determined that a series of three screens (a 50 mesh screen in contact with the bulk of the propellant, an intermediate 200 mesh screen and a 40 micron fine filter) offers the best filtering for the subliming solid propellant. The 50 and 200 mesh screens are 1.25 inch diameter discs and the 40 micron screen is a folded screen assembly with an effective flow area of 6.6 square inches. The tests showed that the filter assemblies using screen material as the filtering medium were not prone to plugging even under conditions of severe recondensation. The filter assemblies were chilled to well below the temperature of the subliming solid propellant. Normally, under these conditions, thorough recondensation and plugging does occur in small diameter tubes. In the case of the filter assemblies, recondensation did occur around the edges of the filter assembly holder--not enough to completely coat or plate the screen which would result in plugging. The lack of filter recondensation is believed to be due to the high available flow area, the low heat conduction from the screen, and the porous nature of the granular propellant that comes in contact with the screens and of any recondensed propellant which may adhere to the screens. Approximately 20 systems have been constructed and tested by Rocket Research Corporation using this type of screen filter assembly and tests of up to one month in duration have been conducted with no flow stoppage or limitation.

Rapid plugging does occur under recondensation conditions when using other filtering materials such as porous metal discs. Some plugging with porous metal has been observed even without recondensation conditions. Therefore, porous metal filters should not be considered for SSRCS filters. It is believed that the problem with porous metal discs is that they offer a greater resistance to vapor and any small particles striking the filter. This results in the small subliming solid particles being trapped and packed into the filter causing eventual plugging. This is not the case with a conventional screen. The porous metal also offers a better heat transfer path and, therefore, serves as a better recondensation surface than does a single flat screen.

1.10.2 Valve Recondensation

The installation of propellant valves on a SSRCS is specifically arranged to prevent any adverse effects due to recondensation. Coaxial propellant valves are oriented reverse to normal flow direction so that recondensation cannot occur on the moving surfaces around the plunger. The only possible point at which recondensation could render the valve inoperative is in the inlet orifice and seat. Under severe adverse temperature differentials, it is possible to build up a sufficient deposit of recondensed propellant which could inhibit the flow of propellant vapor through the seat and orifice area. Such an adverse temperature gradient condition will generally occur only if the valve is located in an area colder than the propellant tank. For one test, a propellant tank and close coupled valve were placed in an environment shroud for several weeks. No stoppage of the valve due to recondensation was observed.

1.10.3 Propellant Line Recondensation

Recondensation can occur readily in propellant lines if the line temperature drops below the equilibrium temperature for the propellant pressure in the lines. However, tests have shown that if the propellant pressure in the lines is decreased by pre-choking, the recondensation temperature may be lowered to any desired level. Tests conducted by Rocket Research Corporation have shown that this technique can operate on a continuous basis without any recondensation.

1.10.4 Nozzle Recondensation

It has been postulated that expansion of the subliming solid vapor through an exhaust nozzle produces a sufficient drop in vapor to cause recondensation in the nozzle. Specific tests have been performed with transparent nozzles to examine this effect. Theoretically, such recondensation, due to expansion of the gas, may be possible; this was never observed during exhaust nozzle testing of the SSRCS. Lack of recondensation is probably due to the very short vapor stay-time for small nozzles. Recondensation takes a finite time; and, apparently, such time is significantly longer than the stay-time of the exhaust vapor flowing through the nozzle. Recondensation will occur only if the nozzle temperature is dropped significantly below the recondensation point for the exhaust vapor at its flow pressure.

1.10.5 Recondensation Prevention

In certain instances, it is possible to use the heat from nearby components to provide sufficient heat preventing recondensation. This is especially advantageous since the heat may be provided without an additional power demand from the satellite. However, it is not always possible to rely on such heat and the application of this nearby heat will merely depend on specific conditions aboard the spacecraft.

It is also possible to use small, low powered heaters to prevent recondensation in the SSRCS. The requirement for such heaters is discussed in detail in Paragraph 2.11. Normally, the heat requirement is 0.5 watt or less. These heaters are constructed by wrapping high resistance, insulated Nichrome wire around either the part to be heated or a small slip ring attached to the part to be heated. The wire is then potted in place with an ultra low vapor pressure epoxy and connected to a suitable electrical connector or terminal strip. These heaters can be designed and built with great versatility and flexibility and maybe sized to fit practically any requirement.

In certain instances, a conventional resistor of sufficient power rating may be used as a heating element. If this is the case, the resistor is attached to the surface to be heated by imbedding it in an aluminum block or by attaching it with heat transfer epoxy. Beryllium oxide-filled epoxy is particularly advantageous in this case since it is a good heat transfer agent and provides an excellent electrical insulation.

Other types of heating elements may also be considered. Where large surfaces must be uniformly heated, a commercially available resistive element may be sprayed on the surface. Care must be exercised to obtain the correct resistive properties. The procedure for utilizing such a heating element consists of: coating the surface to be heated with an insulating layer; applying the conducting heater paint; and applying a second layer of insulation. Electrical connections are made at each end of the surface to complete the heater.

The use of small radioisotope capsules, using alpha emitters to minimize the radiation hazard, may be packaged in small aluminum or stainless steel capsules and attached where heating is required. The advantage of these capsules is their extremely high reliability and simplicity. At the present time, the basic disadvantage is their relatively high cost and the problems associated with placing of radioisotopes aboard satellites for flight. This type of heater appears to be extremely promising for future use when costs are lower and further experience is obtained in handling and flying radioisotopes.

Recondensation in the lines can be prevented by setting the maximum anticipated line pressure at a point below the propellant equilibrium pressure at the minimum expected line temperature. SUBLEX propellant will not recondense if the vapor pressure is less than equilibrium pressure for the corresponding temperature. Line pressure can be controlled to any desired value by choking propellant flow at the propellant valve and setting line pressure with a properly sized nozzle.

Many attitude control missions, such as spinning satellites, do not require thrust control. A wide thrust variation is permitted for attitude control rockets used to maintain spin or process the spin axis.

1.11 Thrust Control

1.11.1 Requirements

The number of pulses or the pulse duration merely varies inversely proportional to the thrust level during a pulse. In such cases, there is no need to provide a mechanism for controlling the change in pressure which occurs when the subliming solid propellant changes. The thrust is allowed to vary as the vapor pressure varies.

A second class of attitude control missions does require a certain degree of thrust tolerance. Such missions are limit cycle missions and orbit positioning. The thrust tolerance may vary from the mean by plus or minus five percent to plus or minus 25 percent. Since the environment and propellant temperature will generally vary by at least a few degrees during a typical mission, the corresponding variation in vapor pressure may exceed the acceptable thrust tolerance limits. In such cases, a means of thrust control must be provided on the SSRCS.

The thrust control mechanism may control either the pressure, mass flow rate, or propellant temperature to maintain a constant thrust level.

1.11.2 Pressure Regulator

Upstream pressure of the exhaust nozzle may be controlled by means of a conventional pressure regulator. The pressure regulator operates on the basis of a differential force between the upstream, or free, pressure and a reference pressure. The unbalanced pressure force acts against a spring to vary the size of a flow orifice. Pressure regulators can be built very compactly and are of well proven design. However, there are two difficulties which often occur in the application of a pressure regulator to the SSRCS. The first problem is that pressure regulators often foul when exposed to gas flow containing even small amounts of particulate matter. Any pressure regulator used in a SSRCS should have an effective filter either built into it or directly upstream. The degree of filtration required depends on the regulator design and its tolerance levels.

The second, and more critical problem, is hardware size and weight. As the operating pressure of a pressure regulator decreases, the required differential diaphram area increases. Pressure regulators designed for a SSRCS may enlarge due to relatively low operating pressures. In certain very low thrust applications, where vapor pressure may be extremely low, the pressure regulator concept becomes impractical. If the use of a pressure regulator is anticipated, the requirements should be transmitted to a manufacturer for evaluation to determine whether the use of a pressure regulator is practical.

There are several manufacturers of regulators and many different types from which to choose. Three pressure regulator manufacturers are:

a. The Whittaker Corporation

b. Sterer Engineering

c. Carleton Controls

1.11.3 Thermally Controlled Variable Orifice

The thermally controlled variable orifice is a device used to maintain a constant mass flow from a SSRCS propellant tank. This device consists of two main parts: A temperature control expansion element and a variable orifice. The temperature controlled actuator is a bellows or piston device filled with a solid or liquid of very high thermal expansion properties. Therefore, small changes in temperature will cause large movement of the actuating element. The expansion element is carefully designed so that an accurate and repeatable graph of actuator movement versus temperature can be carefully plotted. A temperature controlled actuator may be designed so that the effective flow area through the orifice is inversely proportional to vapor pressure. This is accomplished through the use of plotted linear motion versus temperature and propellant temperature versus vapor pressure curves. If the variable orifice is then choked, permitting sufficiently low downstream pressure, a constant mass flow will result.

A thermally controlled variable orifice may be constructed in a number of different ways. The thermal actuator in the variable orifice may be close coupled, separated by some distance, or separated by a flexible shaft. The device is completely nonelectric. The major advantage of the thermally controlled variable orifice over the pressure regulator is that it is not dependent upon pressure forces for actuation and, therefore, may be packaged in a more compact manner. The unit is also lightweight and may be easily fitted into a variety of different propellant tank configurations.

The major disadvantage of the thermally controlled variable orifice is that of accurately sensing propellant temperature. If propellant temperature is not sensed within 1° to 2°F of the actual propellant temperature, significant errors can result in the propellant flow and, therefore, thrust level. It is necessary to place the thermal actuator within the propellant bed or on a metal surface which accurately senses the propellant bed temperature. The device also responds slowly to changes in temperature; thus, it can be used only in those instances where propellant flow is sufficiently low during the on thrust where propellant temperature changes slowly. The thermally controlled variable orifice may not be used with any propellant which exhibits subliming rate limitations, because the operation pressure does not correspond to the equilibrium vapor pressure at any given temperature.

Two experimental thermally controlled variable orifices were purchased from Pyrodyne, Inc., for evaluation on a SSRCS. Pyrodyne's evaluation was not conclusive, since either materials incompatibility or the design of the slide shaft connecting the thermal actuator with the variable orifice caused hang-up of the device. See Paragraph 2.13 for further details on these tests. Further investigation of the thermally controlled variable orifice is recommended since it is a lightweight small volume, thrust control device requiring no electric power.

1.11.4 Thermostatic Thrust Control

The third method of thrust control is the use of a temperature thermostat to maintain propellant temperature at some fixed value. This value is usually a few degrees above the maximum environmental temperature expected during the operating mission. The system is insulated from its surroundings to minimize the heat lost when the environmental temperature falls far below the thermostat controlled temperature. A temperature sensor is placed somewhere within the propellant tank to monitor propellant temperature. This thermostat, in turn, controls the amount of auxiliary heat required to maintain propellant temperature. Since the propellant temperature is maintained, the vapor pressure will likewise remain constant; and, a constant thrust can be expected.

A distinct advantage of thermostat thrust control is that it requires no mechanical or moving parts. There are many different types of temperature sensing units which may be used to control the thermostat system. These are well developed and many have been qualified and used for space vehicles. They are lightweight and generally quite efficient.

While the thermostat thrust control system is a very straightforward and highly reliable system, it has some limitations. First, the propellant used must be completely free of subliming rate limitation. If a rate limitation exists when propellant flow is started, the pressure in the propellant tank will drop below the equilibrium vapor pressure. As a result, the thrust level is not predictable since the drop in pressure depends upon the amount of propellant in the propellant tank. The thrust level will not remain constant with propellant temperature. Secondly, since the propellant tank is insulated from its surroundings and the temperature of the propellant within the propellant tank is higher than its surroundings, no heat transfer can be obtained from the spacecraft. Therefore, it is necessary to obtain all the heat of sublimation from electric power. The instantaneous thrust level which may be maintained on an extensive pulse is limited by the electric power which may be instantaneously applied to the heater within the propellant tank. Some small amount of electric power will also be required continuously during the off periods to maintain the propellant tank temperature. This amount of power can generally be kept below 0.5 watt if good insulation is employed.

Further experimental work on subliming solid thermostat thrust control systems is required to determine the best location for temperature sensing and propellant heating insuring that a constant vapor pressure is maintained. The location at which propellant temperatures are measured is extremely critical, since the highest temperature location will determine the vapor pressure within the propellant tank. The temperature sensing unit must be effectively shielded from the radiant energy source or from the heater as direct exposure to the heater element may result in erroneous temperature indications due to radiant pickup.

Propellant vapor pressures vary widely with small changes in temperatures. As a result, a temperature thermostat thrust control system will require close regulation of propellant temperature to maintain control of pressure and thrust. A regulation of plus or minus 10 percent in thrust will generally require a temperature regulation of better than plus or minus 3°F. While temperature controllers are available which will maintain temperature plus or minus a fraction of a degree Fahrenheit, the main limitation is still the accuracy and response of the temperature measuring system. The response of the heater is also questionable, requiring further experimental work before the effectiveness of a thermostat thrust control system can be evaluated.

1.11.5 Plenum Chamber Thrust Control

A fourth means of thrust control is the plenum chamber. The plenum concept involves the use of a pressure switch, a valve, and a separate plenum chamber. The size of the plenum tank volume is dependent upon a number of factors, the importance of which will vary with specific conditions and missions to be accomplished. These factors are: The thrust pulse level and duration; the mission life; the response of the interconnecting valve and pressure switch system; the thrust tolerance to be maintained; and the number of cycles allowed on the pressure switch and interconnecting valve.

The plenum chamber thrust control system operates in the following manner: The propellant tank, which is allowed to vary in temperature, acts as a reservoir for a constant pressure plenum tank. The plenum tank is set at a pressure slightly below the lowest anticipated pressure, which is, in turn, set by the lowest anticipated temperature in the main propellant tank. The valve which connects the propellant tank with the plenum tank is controlled by means of a pressure switch. The pressure switch has a deadband through which the plenum tank pressure will be allowed to vary. For example, this may be set at 10 percent of the nominal pressure to be established in the plenum tank. When the pressure in the plenum tank falls below the deadband level, the valve to the propellant tank is opened emitting more vapor to the plenum tank. The valve is closed when the pressure in the plenum tank reaches the top point of the deadband. The thruster valves are fed from the plenum tank.

The plenum concept has been demonstrated on a hydrazine control rocket and a thrust level was maintained to better than plus or minus 10 percent. The concept should be equally valid for the SSRCS. However, the size of the pressure switch can sometimes become unwieldy due to low vapor pressures and, therefore, low plenum pressure settings.

The heat of sublimation in the plenum system can be supplied from the spacecraft. The tank may be connected thermally to the spacecraft since sublimation occurs in the main propellant tank, and the propellant tank temperature is allowed to vary.

The basic disadvantages of the plenum system are that it requires two additional active components (a pressure switch and a valve) and the size, weight, and volume of the system may be undesirable. The size of the plenum tank is a function of the desired thrust level and the anticipated maximum pulse duration.

Also, the number of pressure switch and interconnecting valve pulses should be kept to a minimum. The larger the plenum volume, the fewer refill pulses required. This requires a compromise between system size and the desirability of minimum pressure switch and interconnecting pulses. However, plenum volume must be sized on the basis of pressure switch and interconnecting valve life and reliability. The equation which governs the volume required as a function of the number of pulses of an interconnecting refill valve is as follows:

$$V = \frac{I_T RT}{N I_{SP} \Delta_p M}$$

Where:

V	=	plenum volume, ft^2
N	=	number of refill valve pulses
I_T	=	total impulse, lbf-sec
I_{SP}	=	specific impulse, lbf-sec/lbm
R	=	universal gas constant, 1545 ft-lbf/lbm - mol - °F
T	=	vapor temperature °R
Δp	=	plenum volume deadband, lbf/ft^2
M	=	vapor molecular weight

The plenum should be sufficiently large so that the opening of an exhaust valve, producing thrust, does not deplete the plenum below its deadband value faster than the propellant tank can maintain the desired pressure limits within the plenum tank. A plenum system designed for fairly long pulses becomes undesirably large due to the fact that the low density of the subliming solid vapor must be set lower than the propellant tank.

The plenum concept can be used as a means of reducing the possibility of recondensation in SSRCS and care must be exercised to insure that recondensation does not occur in the plenum tank.

Recondensation can be completely prevented without using auxiliary heaters if the plenum pressure is set below the pressure where recondensation occurs at the minimum temperature to be expected in any part of the SSRCS, including exhaust lines, valves, and nozzles.

Since the plenum system concept has been demonstrated, it can be considered for design of flight subliming solid systems. Consideration must be given to the availability of the pressure switch, the life and reliability of the active components, and the additional volume required.

1.11.6 Differential Bellows Thrust Control

A fifth method requiring development involves the control of impulse rather than thrust. Impulse control involves the trapping of a fixed mass of exhaust vapor in a special mechanism and is then exhausted through the exhaust valve and nozzle. The mechanism uses a differential bellows or piston that adjusts itself to provide a volume which is inversely proportional to the pressure. In this manner, a constant mass is trapped that, when exhausted, will produce a constant impulse bit.

This device does appear promising, however, the size may be large, due to low propellant vapor density encountered in the SSRCS.

2.0 SPECIFIC TEST RESULTS

2.1 General

This section contains specific test results obtained during Phase I development of the SSRCS.

2.2 Propellant Production

The propellant used for development of the SSRCS was produced in a gaseous synthesis type plant. The production plant consists of a large dry box modified to accept the propellant inlet, exhaust, and water cooling lines so that the entire propellant reaction chamber can be placed inside. The dry box is continuously purged with dry nitrogen from a liquid nitrogen source to prevent air or moisture from contacting and contaminating the propellant during manufacture or during the transfer to storage jars. See Figure 2-1 for photographs of the assembled propellant production plant.

The dry box was used, in addition to propellant production, for all propellant transfer and storage and for loading of test apparatus. Propellant production was at the rate of 0.25 pound per half-hour run or one to two pounds per day.

Removal of propellant from the chamber was made easier and less time consuming by the use of a polyethelene bag inside the chamber. Propellant forms within the bag and the bag is then removed from the chamber, crushed, and the crystalline propellant poured out into a storage jar. Without the bag, the propellant condenses on the reaction chamber walls forming a hard cake that sticks to the wall.

2.3 Propellant Purity

Tests were performed to determine if propellant vapor pressure remained constant with a constant temperature as the propellant sublimes. In addition, the amount of residue remaining was weighed. The test was accomplished as follows: Preweighed propellant samples were placed in the subliming rate apparatus and slowly sublimed away. Sublimation was periodically halted and the propellant brought to the predetermined equilibrium temperature in order to check vapor pressure. When the vapor pressure dropped, weight of the remaining propellant was checked to determine the percentage of remaining lower vapor pressure propellant.

The results showed that no pressure drop occurred throughout the subliming process. Since the mole ratio was correct and no vapor pressure drop occurred, it was concluded that pure SUBLEX A was being produced. See Table 2-1 for a chart of vapor

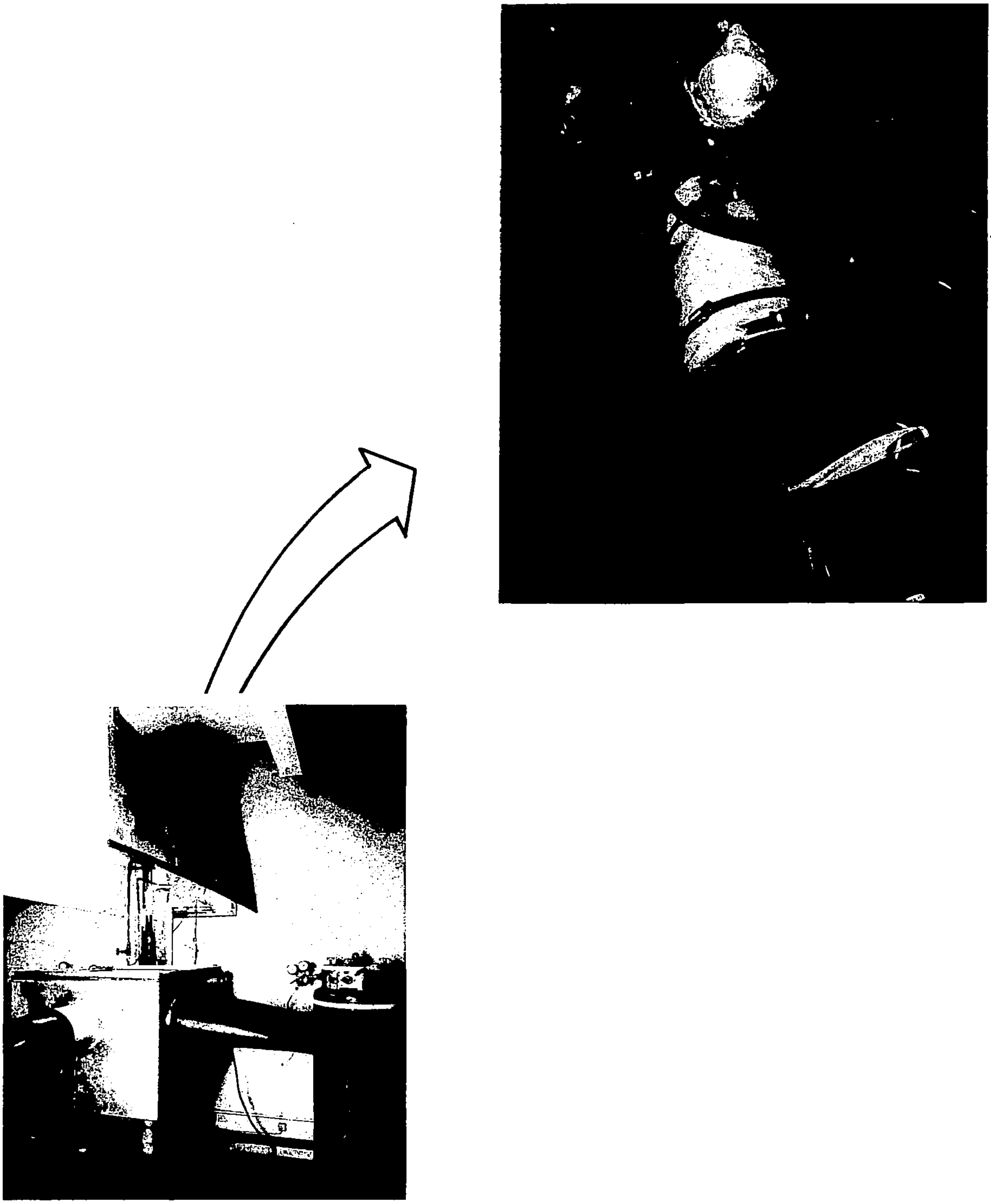


FIGURE 2-1 PROPELLANT PRODUCTION PLANT PHOTOGRAPH

TEST NO. 1

Time Min.	Tank Temperature °F	Tank Pressure Start		Remarks	
		In. Hg.	psia		
1:30	55°	13.85	13.95		Propellant white
1:50	75°	6.1	6.1		
2:00	34°	14.55	14.65	Vacuum pump operating	Small amount of yellow substance visible on remaining propellant.
2:10	70°	8.1	8.1		
2:20	52°	14.55	14.65	Vacuum pump operating	Sublimation of sample half completed. Small amount of yellow substance still visible.
2:30	72°	7.5	7.5		
2:40	48°	14.55	14.65	Vacuum pump operating	
2:50	74°	7.8	7.75		Very small amount of yellow residue remaining at end of sublimation process. No reduction in vapor pressure.
Test Conditions: Room temperature 71 °F. Barometer 29.91					

TABLE 2-1

VAPOR PRESSURE AND RESIDUE OF SUBLEX A EXPOSED TO NITROGEN

Sheet 1 of 2

TEST NO. 2

Time Min.	Tank Temperature °F		Tank Pressure Start		Tank Pressure End		Flow Pulse Duration Min.	Remarks
	Start	End	In. Hg.	psia	In. Hg.	psia		
0	55	75	2.0	0.98	17.8	8.8		Small amount of yellow substance visible on propellant.
							10	
30	34	70	1.0	0.49	13.8	6.8		
							10	
50	52	72	1.0	0.49	17.0	8.4		
							10	
70	48	74	1.0	0.49	14.4	7.15		Very small amount of yellow residue remaining.
Test Conditions: Room Temperature 70°F. Barometer 29.91								

TABLE 2-I

VAPOR PRESSURE AND RESIDUE OF SUBLEX A EXPOSED TO NITROGEN

Sheet 2 of 2

pressure vs. temperature during sublimation of test samples of SUBLEX A exposed only to nitrogen and the residue remaining. See Table 2-2 for test results of samples exposed to air.

2.4 Propellant Storage

Tests to determine the effects of long term storage on SUBLEX A propellant placed in aluminum and glass containers were performed. A propellant sample, in both types of containers, was carefully covered and stored in dry nitrogen. Another sample was momentarily exposed to air before sealing and then stored in air.

Following a three month storage period, the samples were tested for correct vapor pressure and amount of non-volatile residue. A 26 gram sample, which had been stored in air, was tested and found to have the correct equilibrium vapor pressure at room temperature through nearly the entire sublimation process. At test conclusion, a small amount of residue remained producing substantially lower vapor pressure. The residue weighed 0.5 gram (approximately two percent of the original sample). A similar quantity of SUBLEX A, stored in dry nitrogen, had the correct vapor pressure during the complete sublimation process and left a trace of residue which was so small that it could not be weighed.

For further details on material compatibility, which also must be taken into consideration, see Paragraph 2.5.

As a result of these experiments, storage under dry nitrogen is considered the most reliable method.

2.5 Materials Compatibility

Compatibility between the propellants and various materials was checked during a 15 week test storage period. The materials were placed in 1 inch by 2.75 inch test vials which contained 1 to 2 grams of propellant in a nitrogen atmosphere and stored at room temperature for 15 weeks.

Copper and copper based materials, such as brass and bronze, were attacked by the SUBLEX A propellant a short time after the vials were put into storage. The apparent reaction was the formation of a black coating on the metal surface. The copper based materials immediately formed a blue salt in the SUBLEX A. This reaction spread throughout the SUBLEX A as storage time increased.

Time	Tank Temperature °F		Tank Pressure Start		Tank Pressure End		
Min.	Start	End	In. Hg	psia	In. Hg	psia	Remarks
0	53	56	2.9	1.43	12.4	6.15	
30	30	68	1.0	0.49	13.4	6.65	Sublimation of sample half completed.
75	36	76	1.0	0.49	13.6	6.75	Shaking of flask caused extremely rapid rise in vapor pressure.
175	76		1.0	0.49			
240	72		0.3				
Yellow residue remaining weighed 0.5 gram. (10% unweighed still in flask). See comment.							
Test Conditions: Room temperature 70 °F. Barameter 29.41 Comment: Yellow residue turned white and reduced its volume by one half when heated. One layer of whitish residue over coil of hot plate melted and started to boil.							

TABLE 2-2

VAPOR PRESSURE AND RESIDUE OF SUBLEX A EXPOSED TO AIR

At the end of the 15 week storage period, the temperature of the samples was raised to 100°F and then reexamined. The materials tested and the extent of reaction are shown on Table 2-3.

At the end of 20 weeks, the samples were again examined. The following changes had occurred:

a. Buna-N "O"-ring

A slight yellowing of SUBLEX A near the "O"-ring.

b. Graphite

The propellant turned dull gray with no change to the graphite.

c. Tungsten Carbide

A very slight darkening of the tungsten carbide was evident.

d. Electrical Epoxy

A slight darkening of the electrical epoxy developed.

2.6 Effects of Air and Moisture

Tests to determine the effects of air and moisture on SUBLEX A and SUBLEX B propellant and the tolerance level of the propellant to air and moisture were performed. Propellant degradation is indicated by yellowing of the white crystals.

Samples of SUBLEX A and SUBLEX B propellant were placed within small containers and injected with dry air, moist air, and distilled water in quantities of one part per million (PPM), 10 PPM, and 100 PPM. No reaction was observed at the end of four weeks. At the end of 12 weeks, the concentration of moist and dry air was increased to 1000 parts of oxygen per million parts of SUBLEX A with no apparent visible effect on the propellant.

As a check on the air injection method employed, a second test was performed using an evacuated flask of known volume containing a known quantity of SUBLEX A which was backfilled with air.

As a result of the modified test method, it was found that after 24 hours of exposure to moist air the 100 PPM sample exhibited degradation of the top layer. Beneath the top layer, only slight degradation was visible. The 10 PPM test also showed visible signs of degradation of the top layer but no signs of degradation below.

MODERATE TO SEVERE ATTACKING	
1. Blue anodized aluminum	5. Si Lastic "O" Ring
2. Black anodized aluminum	6. Bronze
3. Black lacquer paint	7. Copper
4. Low carbon steel	8. Brass
	9. Cadmium plating

MINOR CORROSION (SLIGHT DISCOLORATION)
1. Chrome plating over steel
2. Blue enamel paint
3. #410 Stainless steel tubing
4. High carbon steel

TABLE 2-3

COMPATIBILITY OF SUBLEX A WITH CANDIDATE COMPONENT MATERIALS

Sheet 1 of 2

NO CORROSION	
1. Nickel plating	11. Plasticized teflon
2. Titanium	12. * Graphite
3. Tungsten	13. Polyethylene tubing
4. * Tungsten carbide	14. Teflon
5. #303, #304, #316 and #321 Stainless steels	15. Nylon
6. Gold leaf	16. Scotch cast
7. Alumina	17. *Buna "N" "O" ring
8. Aluminum	18. Surgical tubing
9. Polished Magnesium	19. Heat transfer cement
10. Black plexi-glass	20. Metal etch primer
	21. *Electrical epoxy

*Corrosive signs appeared at the end of 20 week storage (see Paragraph 2.5).

TABLE 2-3

COMPATIBILITY OF SUBLEX A WITH CANDIDATE COMPONENT MATERIALS

Sheet 2 of 2

The 100 PPM test was performed by placing 14.5 grams of SUBLEX A in a 100 ml. flask and then evacuating the nitrogen from the flask with a vacuum pump. The flask was then allowed to stand to reach room temperature (about five minutes). The flask was opened momentarily to air and closed again. The propellant degradation check was made at the end of 24 hours. Since reaction between air and propellant occurred at 100 PPM, the 1000 PPM test was not performed.

2.7 Vapor Pressure

Laboratory tests were performed on SUBLEX A and SUBLEX B propellants to develop a vapor pressure versus temperature curve for the temperature range of a typical satellite system (-60°F to +140°F).

The tests were conducted by placing the aluminum or glass test vessel, containing 0.25 pound of granular propellant, in an oven or ice pan as required to vary propellant temperature. Two thermocouples were buried in the propellant. However, with this test setup, inaccurate results were obtained due to insufficient thermal protection resulting in recondensation in lines and the pressure transducers.

After numerous modifications to the test setup, the tests were performed using the equipment as shown on Figure 2-2.

The chemical and precision pressure gauges were replaced by electrical transducers. The entire test apparatus was wrapped with heater tape to maintain overall temperature required to prevent recondensation in the system. The test vessel was wrapped with a separate heater tape used to obtain temperature variations required for the test.

The modified equipment arrangement yielded accurate and repeatable test data which is presented in graph form on Figure 1-2.

2.8 Propellant Subliming Rate Test

The purpose of this test was to determine if subliming rate limitations exist for SUBLEX A and SUBLEX B propellants.

The equipment used to perform the subliming rate tests was assembled as shown on Figure 2-3.

Each test was performed as follows:

a. The system was evacuated to remove trapped nitrogen.

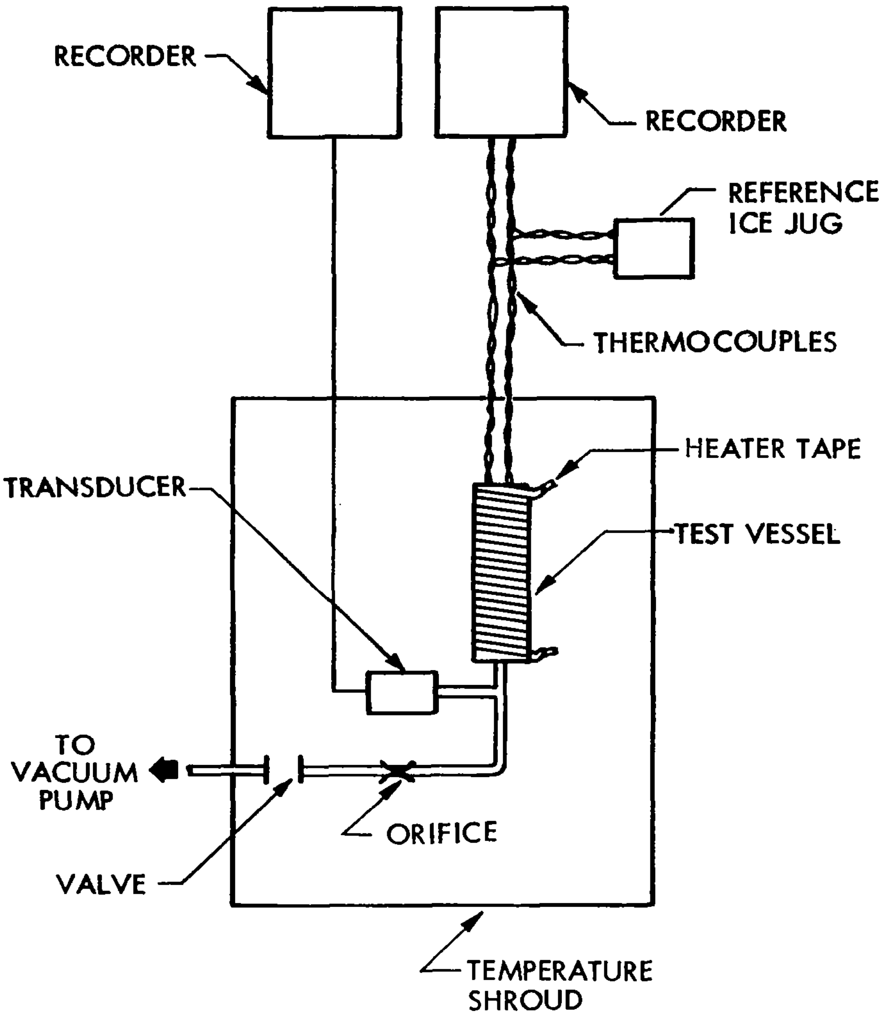


FIGURE 2-2. VAPOR PRESSURE TEST EQUIPMENT SETUP

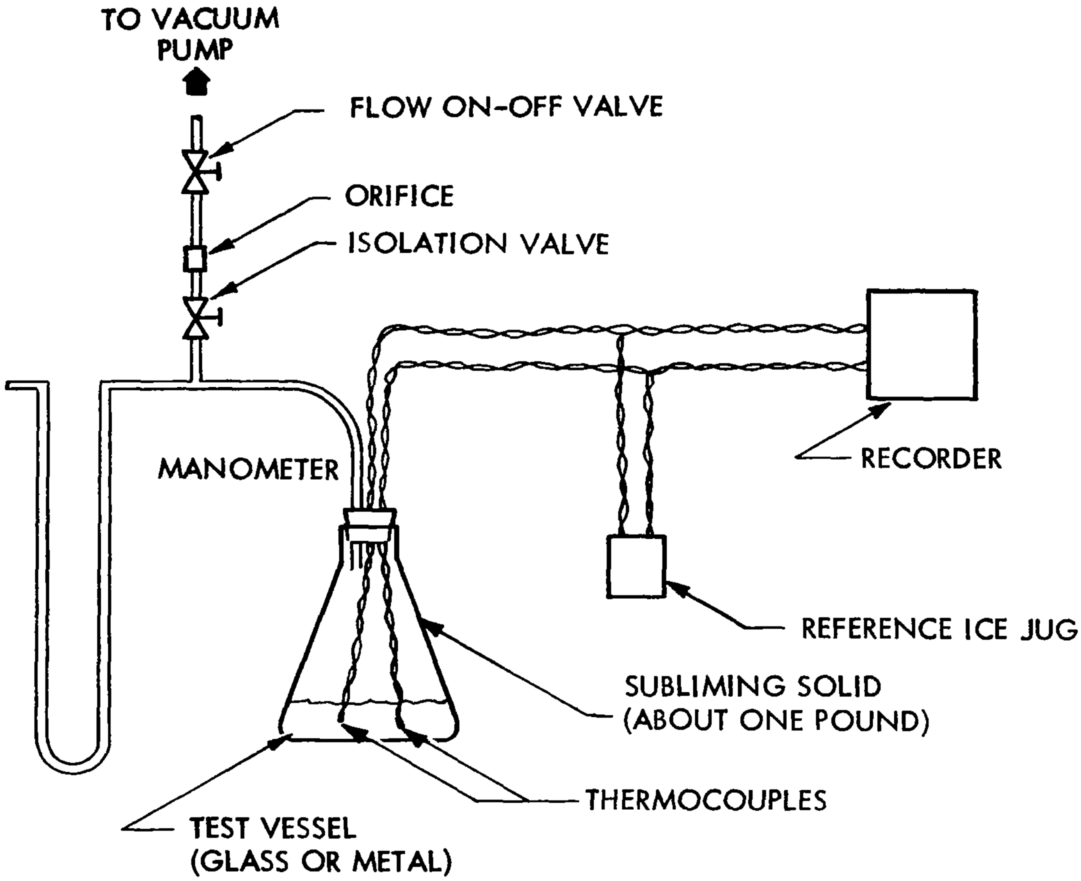


FIGURE 2-3. SUBLIMING RATE TEST EQUIPMENT SETUP

b. An equilibrium temperature was established.

c. The flow valve was opened while monitoring pressure, temperature and time.

d. Flow was then calculated from the measured pressure.

e. If a reduction in pressure below that corresponding to the measured temperature of the adiabatic vaporization curves occurred, a subliming rate limitation was indicated.

As a result of this testing, it was determined that powdered SUBLEX B does have some subliming rate limitation. However, the degree of rate limitation is meaningful only when associated with some specific system as it is dependent upon the amount of propellant in the propellant tank, thrust level, propellant temperature, and other factors. For example, in one subliming rate test 0.4 pound of powdered SUBLEX B was placed in an aluminum tank measuring two inches in diameter by eight inches long, at a simulated thrust level of 10^{-4} pounds. One thermocouple was placed in the vapor above the propellant. Under flow conditions, the propellant vapor pressure was one-half or less of the expected equilibrium pressure for the temperature measured. When the flow was stopped by shutting off the valve, pressure built up to its equilibrium value. The pressure immediately dropped again when flow recommenced, indicating a reduction in sublimation rate.

A second test result was obtained during certification testing of a laboratory test rocket delivered to the Lewis Research Center. In this test, four pounds of SUBLEX B propellant were placed in a nine inch diameter spherical aluminum tank and tested at a measured thrust level of 10^{-4} pounds. The propellant pressure under flow agreed with the equilibrium pressure for the temperature measured, indicating no sublimation rate reduction.

The specific results of the subliming rate tests for SUBLEX B are shown on Table 2-4.

Tests to determine any subliming rate limitations of SUBLEX A were performed as a part of other specific tests during the program. No subliming rate limitations for SUBLEX A were revealed during various system tests at thrust levels up to 5×10^{-2} pounds with one-half pound of propellant in the tank.

2.9 Specific Impulse Determination

Initial performance tests were run to determine the specific impulse (I_{sp}) of a SUBLEX A system. The test procedure consisted of placing a SSRCS on the Compound

TEST NO. 1

Time Min.	Tank Temperature °F		Tank Pressure		Theoretical Pressure psia	Remarks
	Top	Bottom	In. Hg*	psia		
0		68	0.1	0.098	1.20	Vacuum Pump On
5	80	64	1.1	1.08	1.05	" " "
10	86	79	1.4	1.37	1.95	" " "
12	86	75	1.1	1.08	1.55	" " "
14	84	74	1.0	0.98	1.53	Vacuum Pump Off
15	82	73	2.0	1.96	1.52	" " "
16	82	74	2.1	2.06	1.53	" " "
17	80	74	2.15	2.11	1.53	" " On
18	80	74	0.9	0.88	1.53	" " "
19	77	70	0.8	0.79	1.31	" " "
21	77	68	0.7	0.69	1.20	" " "
25	74	66	0.65	0.64	1.19	" " "
30	78	68	0.8	0.79	1.20	" " "
33	78	69	0.8	0.79	1.25	" " "
38	78	68	0.75	0.74	1.25	" " "
43	80	68	0.7	0.69	1.25	" " "
47	78	66	0.65	0.64	1.19	" " "
52	79	72	0.85	0.84	1.45	" " "
57	79	72	0.85	0.84	1.45	" " "
63	78	71	0.8	0.79	1.35	" " "

Test Conditions: 0.012 inch orifice used.
*Average, one leg of manometer.

TABLE 2-4

SUBLIMING RATE FOR SUBLEX B

Sheet 1 of 2

TEST NO. 2

Time		Tank Temperature °F		Tank Pressure End		Theoretical Pressure	Remarks
Hour	Min.	Top	Bottom	In. Hg	psia	psia	
	0	62	62	0.5	0.245	0.025	
	5	62	60	0.4	0.196	0.020	
	30	68	68	0.7	0.343	1.20	
	35	64	64	0.6	0.295	1.05	
	40	64	62	0.55	0.270	0.025	
	45	64	62	0.55	0.270	0.025	
	50	68	64	0.7	0.343	1.05	
1	5	68	64	0.55	0.270	1.05	
1	15	69	65	0.6	0.295	1.10	
1	25	70	66	0.65	0.320	1.17	
1	35	71	68	0.65	0.320	1.20	
1	45	69	65	0.65	0.320	1.10	
2	20	73	69	0.8	0.390	1.25	
2	30	73	69	0.8	0.390	1.25	
2	40	74	69	0.8	0.390	1.25	
2	50	74	69	0.8	0.390	1.25	
3	00	74	69	0.8	0.390	1.25	
3	10	74	69	0.8	0.390	1.25	
3	20	75	70	0.8	0.390	1.31	
3	30	75	70	0.8	0.390	1.31	
3	40	75	70	0.8	0.390	1.31	
3	50	75	70	0.8	0.390	1.31	
4	00	73	68	0.75	0.370	1.20	
4	10	73	68	0.75	0.370	1.20	
4	20	73	68	0.7	0.343	1.20	
4	30	73	68	0.7	0.343	1.20	
4	40	73	68	0.7	0.343	1.20	
5	20	73	68	0.7	0.343	1.20	
5	40	64	60	2.4	1.20	0.020	Heat and pump valve off
5	50	64	60	2.4	1.20	0.020	

Test Conditions: Flow pulse duration continuous and pressure read across a 0.012 inch orifice.

TABLE 2-4. SUBLIMING RATE FOR SUBLEX B

Sheet 2 of 2

Pendulum Microthrust Balance (calibrated for 10^{-2} pounds of thrust), inside the vacuum chamber. Thermocouples, required to measure the propellant and tank wall temperature, were then installed and calibrated. The tank pressure transducer was calibrated from 0 to 10 psia.

The balance was operated in the evacuated chamber to provide thrust data. Mass flow rate of the system was then measured by exhausting the nozzle into a closed plenum flask of known volume, and, at the same time, recording propellant tank and plenum tank pressures vs. time. The resulting mass flow rate is obtained by calculating the gas mass trapped in the plenum volume after a measured flow period and dividing this by the flow time. Specific impulse is calculated by divising the measured thrust by the flow rate at a common operating pressure.

The initial I_{sp} test performed resulted in an I_{sp} of between 80 and 90 seconds--about 10 seconds higher than the highest theoretical I_{sp}. This discrepancy was believed due to an error in balance calibration. Examination of the remote calibration equipment revealed that the calibrating weight mechanism was not completely releasing.

The possibility of gas eddy currents in the vacuum chamber, causing unwanted balance deflections resulting in high thrust indications, was eliminated following an eddy current test. The test was performed with the nozzle mounted just above the balance platform but not attached. The propellant tank was mounted on the platform in order to include the effect of the tank's cross-sectional area in the test. Thrusting was then simulated by flowing air through the nozzle. No balance deflection occurred, therefore eliminating eddy currents as a source of error.

The calibration system was modified by extending the weight support thread to prevent hang-up and the thrust measurements repeated. Results for a seven run test showed an average of 64.6 seconds and compared favorably with the theoretical I_{sp} of 66.6 seconds. Thrust values obtained varied from 0.65×10^{-2} pounds to 0.70×10^{-2} pounds. The results of the seven test runs are shown on Table 2-5.

2.10 Migration and Recondensation

The migration and recondensation tests revealed that some method of preventing recondensation must be employed when using SUBLEX A as a propellant. It was shown that recondensation occurs anytime the temperature of lines, valves, or orifices, connected to the propellant tank falls below the propellant temperature. The migration and recondensation tests performed are described in the following paragraphs.

Test Run Number	Measured Thrust @ 5 psia - lbf	Measured Flow Rate @ 5 psia lbm/sec.	Measured Specific Impulse I_{sp} = lbf/lbm sec.
1	$.688 \times 10^{-2}$	1.065×10^{-4}	64.6
2	$.702 \times 10^{-2}$	1.065×10^{-4}	65.9
3	$.692 \times 10^{-2}$	1.065×10^{-4}	65.0
4	$.678 \times 10^{-2}$	1.065×10^{-4}	63.6
5	$.652 \times 10^{-2}$	1.065×10^{-4}	61.1
6	$.688 \times 10^{-2}$	1.065×10^{-4}	64.6
7	$.698 \times 10^{-2}$	1.065×10^{-4}	65.5

NOTE: Theoretical maximum specific impulse (I_{sp} max) is 66.6 seconds at the area ratio and ambient pressure conditions during test.

TABLE 2-5

SPECIFIC IMPULSE TEST RESULTS
OF SUBLEX A

2.10.1 Static Migration and Recondensation in Propellant Tanks

Two tests were performed to determine the amount of SUBLEX A migrating from a glass flask to an aluminum propellant tank.

For both tests, a small sample of SUBLEX A was placed in the glass flask which was connected to the aluminum tank by means of clear plastic tubing. The empty propellant tank was placed in a container of ice and water, salt water and ice, or dry ice and alcohol, to obtain the required temperature. Throughout the tests, the propellant tank was weighed on a balance to determine the amount of recondensation and migration at the various test temperatures. The results of these tests are shown on Table 2-6.

2.10.2 Static Migration and Recondensation in Lines

The amount of static migration and recondensation of both SUBLEX A and SUBLEX B propellants in aluminum lines was determined experimentally. SUBLEX B did not exhibit rapid recondensation, however, SUBLEX A exhibited quite rapid recondensation even under moderate temperature differentials between propellant tank and line.

This series of tests was performed by connecting a section of aluminum line, closed at one end, to a flask containing subliming solid propellant at room temperature. Following system evacuation, the aluminum line was chilled for varying lengths of time. After the chilling periods, the amount of recondensation or line plugging was determined by recording the pressure drop across the plugged line section with a manometer while flowing through a fixed orifice. The results of this test, comparing static recondensation properties of SUBLEX A and SUBLEX B, are shown on Table 2-7.

SUBLEX B was also subjected to an extended recondensation check lasting eight hours. No recondensation was detected during the eight hour test. Line temperature during the test varied from -5°F to 30°F while the room temperature was maintained at 69°F.

However, a series of tests performed as part of another program did show that SUBLEX B will recondense. In this test two flasks were connected by Tygon tubing. SUBLEX B was placed in the first flask and maintained at a temperature between 60°F and 140°F, while the second flask was maintained at 0°F and 60°F. Recondensation was observed in a few minutes in the cooler flask for all cases in which a differential temperature was maintained. As the temperature difference was increased, recondensation became more rapid.

TEST NO. 1

TIME		TEMPERATURE °F.		MIGRATED PROPELLANT MILLIGRAM
MIN.	SEC.	FLASK	ALUMINUM TANK	
	10	60	32	12
	30	60	32	40
1	00	60	32	70
2	00	60	32	104
5	00	60	32	197
10	00	60	32	275
30	00	60	32	705
	10	62	-2	0
	30	62	-2	0
1	00	62	-2	0
2	00	62	-2	8
5	00	62	-2	102
10	00	62	-2	225
30	00	62	-2	685
	10	62	-10	0
	30	62	-10	0
1	00	62	-10	35
2	00	62	-10	109
5	00	62	-10	250
10	00	62	-10	330
30	00	62	-10	900

NOTE: Aluminum tank inside surface area = 27 square inches. Test run with continuous flow.

TABLE 2-6

STATIC MIGRATION AND RECONDENSATION OF SUBLEX A IN PROPELLANT TANKS

Sheet 1 of 2

TEST NO. 2

TIME		TEMPERATURE °F		MIGRATED PROPELLANT MILLIGRAM
MIN.	SEC.	FLASK	ALUMINUM TANK	
	10	62	-65	0
	20	62	-65	0
1	00	62	-65	0
2	00	62	-65	52
5	00	62	-65	390
10	00	62	-65	521
30	00	62	-65	978
	10	62	0	0
	30	62	0	0
1	00	62	0	0
2	00	62	0	30
5	00	62	0	136
10	00	62	0	300
30	00	62	0	845
	10	63	32	0
	30	63	32	0
1	00	63	32	0
2	00	63	32	20
5	00	63	32	230
10	00	63	32	609
30	00	63	32	902
	10	68	-65	0
	30	68	-65	0
1	00	68	-65	0.2
2	00	68	-65	0.475
5	00	68	-65	747
10	00	68	-65	996
30	00	68	-65	1985

NOTE: Aluminum tank inside surface area = 22 square inches
Test run with continuous flow

TABLE 2-6
STATIC MIGRATION AND RECONDENSATION OF
SUBLEX A IN PROPELLANT TANKS

Sheet 2 of 2

TEST NO. 1 SUBLEX A 0.25 in. Line

Time		Line Temperature °F	ΔP Across Aluminum Line		Remarks
Min.	Sec.		In. Hg	psia	
	10	32	0	0	
	30	32	0	0	
1	00	32	0	0	
2	00	32	0	0	
5	00	32	0	0	
6	00	32	0	0	
7	00	32	5.2	2.55	Line plugged at 7 minutes
10	00	32	5.2	2.55	
30	00	32	9.0	4.4	
	10	0	0	0	
	30	0	0	0	
1	00	0	0	0	
2	00	0	0	0	
5	00	0	0.1	0.49	
10	00	0	4.5	2.2	Line plugged between 5 and 10 minutes.
30	00	0	12.7	6.25	

Test Conditions: Flow cross-sectional area = 0.00189 square inches.
Inside wall area of 9.0 inch length of 0.25 inch aluminum tubing = 1.39 square inches. Room temperature 65°F.
Flow pulse duration 4 seconds

TABLE 2-7

STATIC MIGRATION AND RECONDENSATION OF SUBLEX A AND SUBLEX B IN PROPELLANT LINES

Sheet 1 of 3

TEST NO. 2 SUBLEX B 0.25 In. Line

Time		Line Temperature °F	ΔP Across Aluminum Line		Remarks
Min.	Sec.		In. Hg	psia	
	10	32	0	0	
	20	32	0	0	
	30	32	0	0	
1	00	32	0	0	
2	00	32	0	0	
5	00	32	0	0	
10	00	32	0	0	
15	00	32	0	0	
30	00	32	0	0	
	10	32	0	0	Line length reduced to 4.5 inches
	20	32	0	0	
	30	32	0	0	Propellant tank heated with lamp.
1	00	32	0	0	
5	00	32	0	0	
10	00	32	0	0	
15	00	32	0	0	
30	00	32	0	0	

Test Conditions: Flow cross-sectional area = 0.00189 inches.
Inside wall area of 9.0 inch length of 0.25 inch aluminum tubing = 1.39 square inches. Room temperature 65°F.
Flow pulse duration 4 seconds.

TABLE 2-7

STATIC MIGRATION AND RECONDENSATION OF SUBLEX A AND SUBLEX B IN PROPELLANT LINES

Sheet 2 of 3

TEST NO. 3 SUBLEX B 1/8 in. Line

Time		Line Temperature °F	ΔP Across Aluminum Line		Remarks
Min.	Sec.		In. Hg	psia	
	10	32	0	0	
	20	32	0	0	
	30	32	0	0	
1	00	32	0	0	
2	00	32	0	0	
5	00	32	0	0	
10	00	32	0	0	
30	00	32	0	0	
	10	0	0	0	
	20	0	0	0	
	30	0	0	0	
1	00	0	0	0	
2	00	0	0	0	
5	00	0	0	0	
10	00	0	0	0	
30	00	0	0	0	
	10	-65	0	0	
	20	-65	0	0	
	30	-65	0	0	
1	00	-65	0	0	
2	00	-65	0	0	
5	00	-65	0	0	
10	00	-65	0	0	
30	00	-65	0	0	

Test Conditions: Flow cross sectional area of line = 0.000962 square inches. Internal surface area of 9 inch length of 1/8 inch aluminum tubing = 0.99 square inches. Room temperature 70°F. Flow pulse duration 4 seconds.

TABLE 2-7

STATIC MIGRATION AND RECONDENSATION OF SUBLEX A AND SUBLEX B IN PROPELLANT LINES

Sheet 3 of 3

The results of this test seem to contradict the results previously outlined. It is possible, during testing of the aluminum lines, that leakage in the apparatus used for the first test caused a pressure differential which inhibited flow from the hot flask to the cool flask.

Since recondensation occurred in the test using two glass flasks, it must be concluded that SUBLEX B will recondense where a temperature differential exists.

2.10.3 Recondensation in Propellant Lines Under Flow Conditions

During this test, a line containing valves and orifices was connected to a propellant container and a vacuum pump as shown on Figure 2-4. The line was evacuated and cooled in the liquid for ten minutes, after which the line was removed and flow started. The tests were performed with and without the pre-choking orifice installed and with various size orifices installed. The results of the recondensation tests in lines under flow conditions for SUBLEX A and SUBLEX B without the pre-choking orifice are shown on Table 2-8. The results with a pre-choking orifice installed are shown on Table 2-9.

2.10.4 Preventative Measures

As a result of the recondensation tests performed, it was concluded that there are at least four basic methods of preventing recondensation. These four methods are:

a. Passive thermal control coatings

b. Electric heating

c. Nuclear isotope heating

d. Pre-choking

2.10.4.1 Passive Thermal Coatings

The application of selected thermal coatings to propellant lines and valves placed near warm components in the satellite vehicle is one of the simplest and cheapest means of preventing propellant recondensation. This insures that the required temperature gradient is maintained between critical components and the propellant tank.

2.10.4.2 Electrical Heating

If the use of passive thermal coating is not possible, the required temperature may be maintained by employing electric heaters. Normally, the amount of heat required is so small that these heaters may be economically used. For example, a 0.25 inch

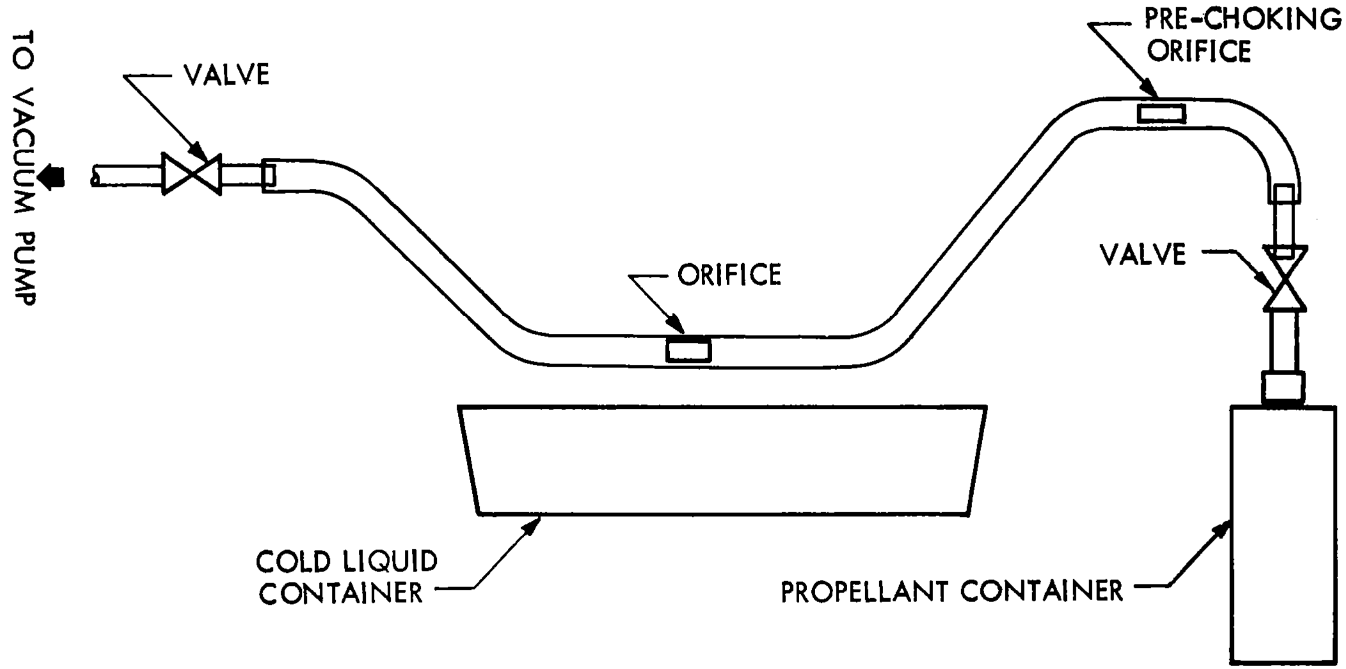


FIGURE 2-4. RECONDENSATION IN PROPELLANT LINES UNDER FLOW CONDITIONS TEST EQUIPMENT SETUP

Test No.	Propellant	Orifice Size Inches	Chill Tray Temperature °F.	Tank Temperature °F	Flow Pulse Duration Min.	Remarks
1	SUBLEX A	0.120	32	68	5	No propellant build-up.
2	SUBLEX A	0.012	32	68	5	Orifice plugged.
3	SUBLEX A	0.038	32	68	5	Line coated but orifice not plugged.
4	SUBLEX B	0.038	32	68	5	No recondensation in line.
5	SUBLEX B	0.038	32	68	5	No recondensation in line.
6	SUBLEX A	0.012	0	70	5	Line plugged solid upstream of orifice. Light coating 2 inches downstream of orifice.
7	SUBLEX A	0.038	0	70	5	Line walls heavily coated but orifice not plugged.
8	SUBLEX A	0.120	0	70	5	Line heavily coated on both sides of orifice but orifice not plugged.
Test Conditions: Each test run for 10 minutes.						

TABLE 2-8

RECONDENSATION TESTS OF PROPELLANT LINES UNDER FLOW CONDITIONS WITHOUT PRE-CHOKING ORIFICE INSTALLED

TEST NO . 1

Time Min.	Tank Temperature °F	Chill Tray Temperature °F	Flow Pulse Duration	Remarks
5	70	0	5	Pre-choke orifice 0.012 inch Test Orifice 0.12 inch No propellant deposit in line
5	70	0	5	Pre-choke orifice 0.012 inch Test orifice 0.12 inch No propellant deposit in line
5	70	-65	5	Pre-choke orifice 0.012 inch Test orifice 0.038 Light propellant coating 6 inches upstream and 3 inches downstream from test orifice. Test orifice open.
5	70	-65	5	Pre-choke orifice 0.012 inch Test orifice 0.12 inch Line nearly clear Test orifice open

Test Conditions: Equipment setup as shown on Figure 2-4 using SUBLEX A propellant.
Test duration: 30 minutes

TABLE 2-9

RESULTS OF RECONDENSATION TESTS OF PROPELLANT LINES UNDER FLOW CONDITIONS WITH PRE-CHOKING ORIFICE INSTALLED.

Sheet 1 of 2

TEST NO. 2

Time Min.	Tank Temp. °F	Chill Tray Temp. °F	Tank Pressure In. Hg	Tank Pressure psia	△ P Across Test Orifice In. Hg	△ P Across Test Orifice psia	Remarks
0	50	45	9.5	4.7	0.18	0.09	Line temperature ≈10°F below tank temperature.
10	52	40	10.4	5.1	0.18	0.09	
20	50	40	9.1	4.5	0.18	0.09	No line recondensation.
30	52	40	10.0	4.9	0.18	0.09	
0	50	34	9.3	4.6	0.18	0.09	Line temperature ≈15°F below tank temperature.
10	48	36	8.2	4.0	0.18	0.09	
20	48	34	8.7	4.2	0.18	0.09	No line recondensation.
30	50	34	9.5	4.7	0.18	0.09	
0	50	-8	9.2	4.5	0.18	0.09	Line temperature 0°F.
10	50	-2	10.0	4.9	0.18	0.09	
20	51	-6	9.5	4.7	0.18	0.09	Line plugged solid at 53 minutes.
30	50	-6	9.8	4.8	0.55	0.27	
40	51	+2	9.4	4.6	0.65	0.308	
50	52	-5	9.4	4.6	9.8	4.8	
53							

Test Conditions: Room temperature; Pre-choke orifice 0.012 inch. Test orifice 0.038 inch. Test Duration: 2.5 hours. Test equipment set up per Figure 2-4.

TABLE 2-9

RESULTS OF RECONDENSATION TESTS OF PROPELLANT LINES UNDER FLOW CONDITIONS WITH PRE-CHOKING ORIFICE INSTALLED.

Sheet 2 of 2

polished aluminum line radiating totally to space at 166°F requires 0.25 thermal watt per foot. Considerably less power is required for less severe temperature conditions and for insulated lines.

2.10.4.3 Nuclear Isotope Heating

The third method of supplying heat is by use of radioisotopes. These radioisotopes are completely passive, requiring no electric power, resulting in maximum reliability. The use of alpha emitting isotopes often reduces the radiation hazard to below typical background radiation. The radioisotope capsule is small and light weight. (A typical capsule is 1/8 inch in diameter by 1/4 to 1/2 inch long and weighs a few grams.) The cost of these capsules in small quantities is between $500.00 and $1,500.00 each for the 0.10 thermal watt size. In operation, these capsules are placed around, or in lines, valves, orifices, and filters for specific missions. See Appendix I for a detailed analysis of radioisotopes investigated as possible candidates for use on the SSRCS.

2.10.4.4 Pre-Choking

Recondensation may also be prevented in extended lines by placing a valve and choking orifice at the propellant tank outlet. The flow pressure in the line is adjusted by the size of the exhaust nozzle to a sufficiently low level preventing recondensation at the lowest line temperature.

2.11 Thermal Conditioning

Since the SSRCS requires heat to be supplied for sublimation of the propellant, an important area of investigation was thermal conditioning. A series of calculations and experiments were conducted to determine the amount of heat required for sublimation at any given continuous thrust level; the optimum methods of heat transfer into a typical subliming solid propellant tank; and the average continuous thrust which could be maintained for various environmental temperature conditions and different heat transfer modes.

2.11.1 Analytical Determination of Sublimation Heat

Theoretical calculations to determine the amount of heat required to produce a given thrust and duty cycle at a specific impulse of 80 seconds were performed for both SUBLEX A and SUBLEX B propellants. The average heat, in watts, required for SUBLEX A is equal to 1.30×10^4 x instantaneous thrust x duty cycle. The average heat, in watts, required for SUBLEX B is equal to 1.15×10^4 x instantaneous thrust x duty cycle. A graph of sublimation heat in watts vs. thrust and duty cycle is presented on Figure 2-5.

The relation between thrust and required heat transfer rate may also be expressed in terms of BTU's per hour and variable specific impulse by the following expressions:

SUBLEX A

Thrust (pounds) = specific impulse (seconds) x heat transfer (BTU's per hour) x 3.5×10^{-7}.

SUBLEX B

Thrust (pounds) = specific impulse (seconds) x heat transfer (BTU's/power) x 3.14×10^{-7}.

2.11.2 Preliminary Heat Transfer Tests

Two preliminary heat transfer tests (one hour and three hour duration), were conducted in the following manner to determine requirements for later experiments.

A six inch diameter propellant tank, painted black, was placed on a test bench and attached to a vacuum pump. The propellant tank was opened to the vacuum pump and allowed to flow. The propellant was weighed before and after each test to determine the amount actually sublimed. Propellant pressure was also monitored during these tests. The pressure reached an equilibrium of approximately 0.3 psia after approximately 15 minutes of operation and remained steady for the remainder of the test. The amount of propellant vaporized and exhausted during the one and three hour tests was equivalent to approximately 3×10^{-3} pounds of continuous thrust produced. Propellant temperature during these tests was estimated to be -10°F. Heat transfer to the tank was by radiation only, since no direct conductive paths were included.

As a result of these preliminary tests, further testing was performed under vacuum rather than atmospheric conditions. The reason for vacuum testing was moisture recondensation that occurred on the tank and convection currents of air around the tank causing unrealistic test conditions that influenced heat transfer.

2.11.3 Heat Transfer Calculations

Following preliminary tests, calculations were made on heat transfer by radiation, conduction, convection, and combinations of these, from spherically-symmetric

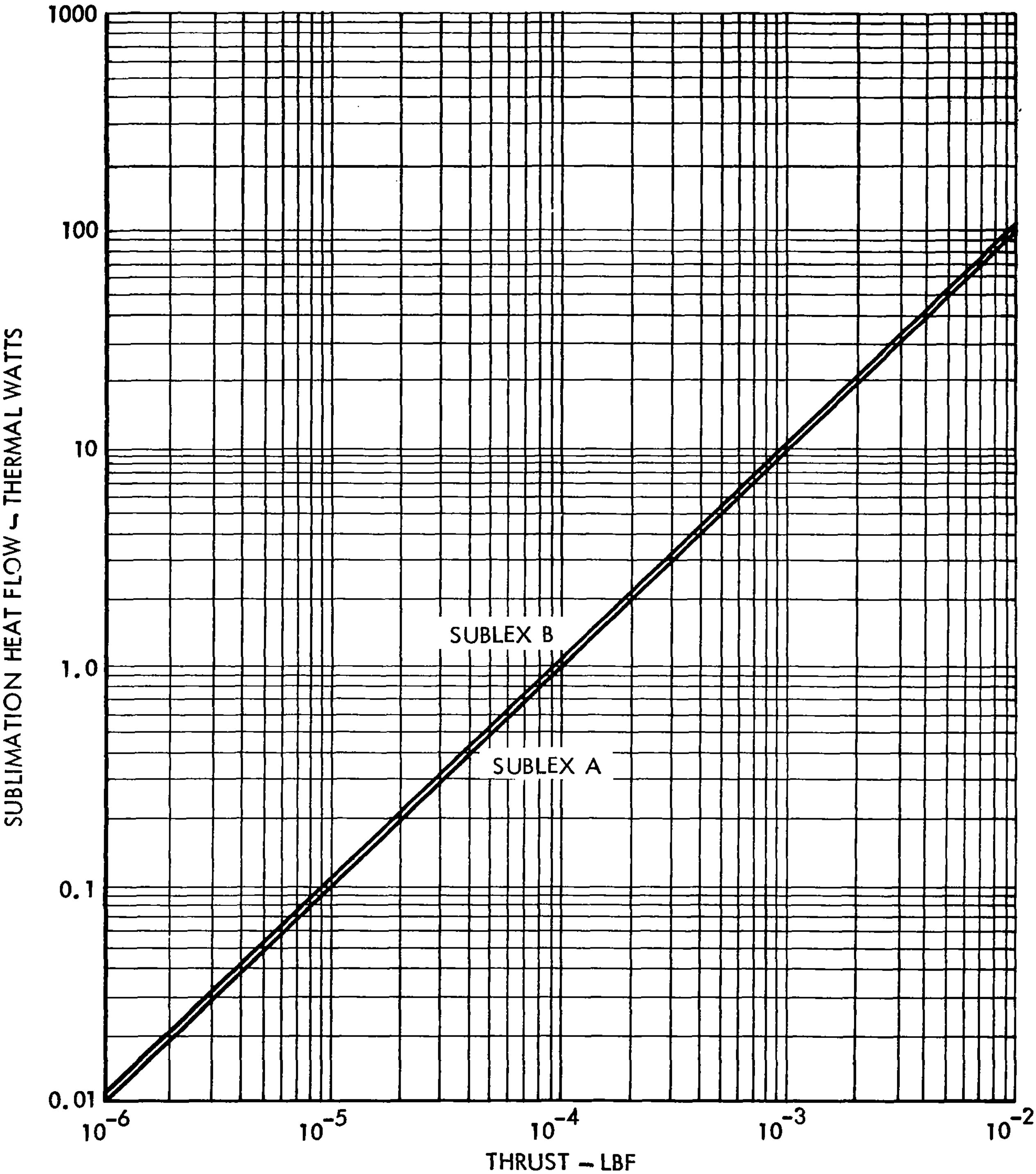


FIGURE 2-5 SUBLIMATION HEAT FOR VARIOUS THRUST AND DUTY CYCLES FOR SUBLEX A AND SUBLEX B

surroundings to a spherical propellant tank and from the propellant tank wall to the propellant. Graphs for the various heat transfer methods are shown on Figures 2-6 through 2-12. Table 2-10 lists heat transfer by conduction and radiation for SUBLEX A.

The calculation of heat transfer from the surrounding satellite structure to the subliming solid propellant was split into two parts. Heat transfer from the propellant tank surface to the subliming solid propellant surface, and from the satellite structure to the propellant tank surface were calculated. Heat transfer between the propellant tank wall and the propellant surface by means of radiation convection, and conduction heat transfer mechanisms were compared. Results of this comparison are shown on Figures 2-7 and 2-8. Propellant tank wall temperatures of +140°F and -40°F were investigated. For transfer by radiation two emissivities were considered; one with a relative emissivity of 0.099 and the other with a relative emissivity of 0.882. As expected, the higher emissivity transferred larger quantities of heat. A gas barrier space of 0.05 inch was assumed for heat transfer by conduction. A vapor velocity of 5 feet per second was assumed for heat transfer by convection.

The calculations were performed with a fixed propellant tank wall temperature and varying subliming surface temperatures. The parameter calculated was the heat energy, in BTU's per hour, transferred to the propellant surface from the propellant tank wall per unit area in square feet. Results of these calculations are shown on Figures 2-7 and 2-8.

Figures 2-9 and 2-10 are the results of calculations for combined radiation and conduction heat transfer from the propellant tank wall to the propellant surface. Calculations were performed at propellant tank wall temperatures of +140°F and -40°F with a relative emissivity of 0.882 between the propellant tank wall and the propellant surface at varying gas vapor layers. From these curves, it can be seen that it is desirable to minimize the distance between the subliming solid and the propellant tank wall. In practice, this can be done by utilizing honeycomb structures in tank construction.

The second set of heat transfer calculations was made for the transfer of heat from the satellite structure and surroundings to the propellant tank. In order to simplify these calculations it was assumed that a vehicle diameter of three feet would radiate to the propellant tank. Various propellant tank diameters and relative emissivities were

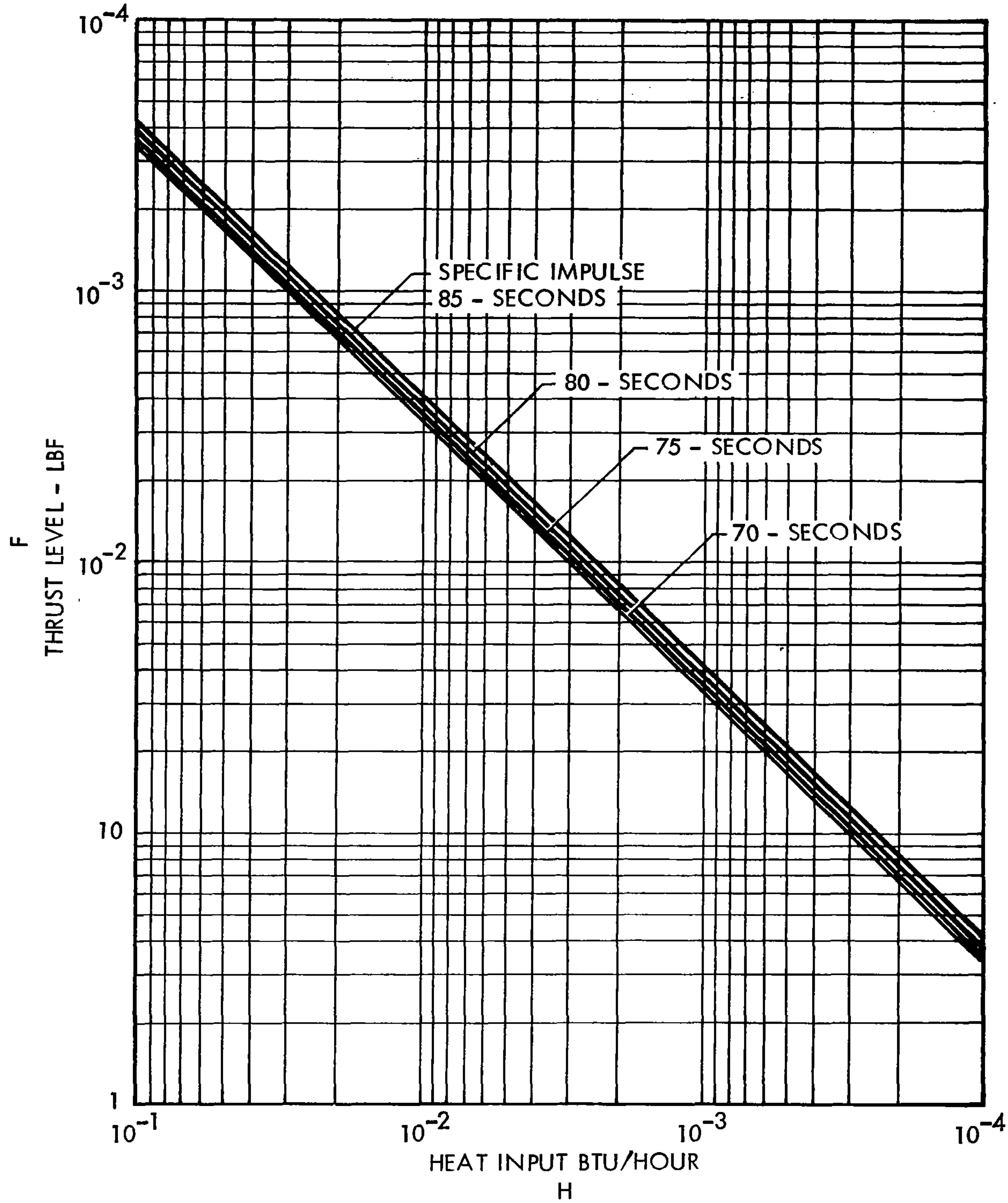


FIGURE 2-6 REQUIRED HEAT FLOW VS. THRUST LEVEL FOR VARIOUS SPECIFIC IMPULSES

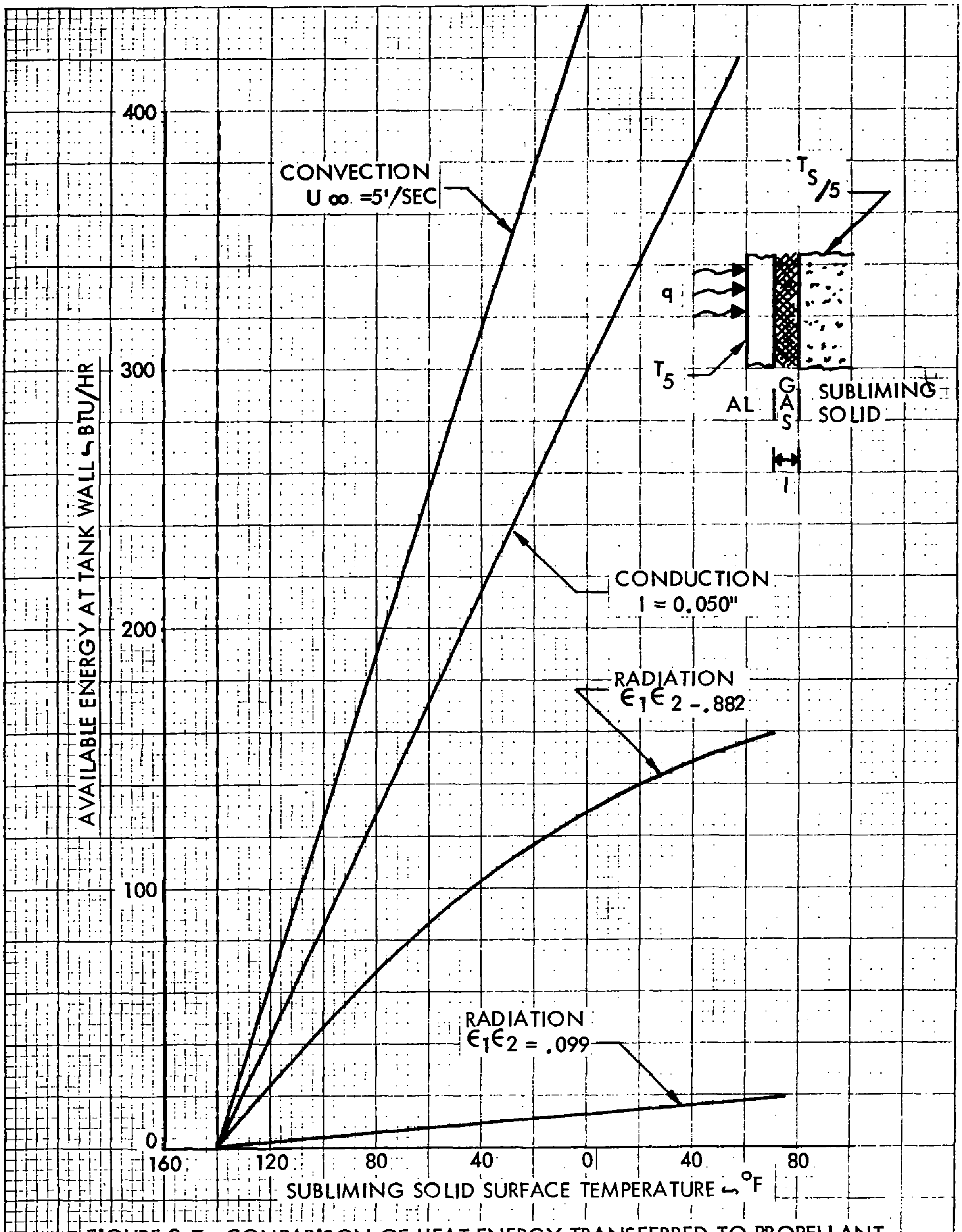


FIGURE 2-7. COMPARISON OF HEAT ENERGY TRANSFERRED TO PROPELLANT SURFACE BY CONVECTION, CONDUCTION AND RADIATION (TANK WALL TEMPERATURE +140°F)

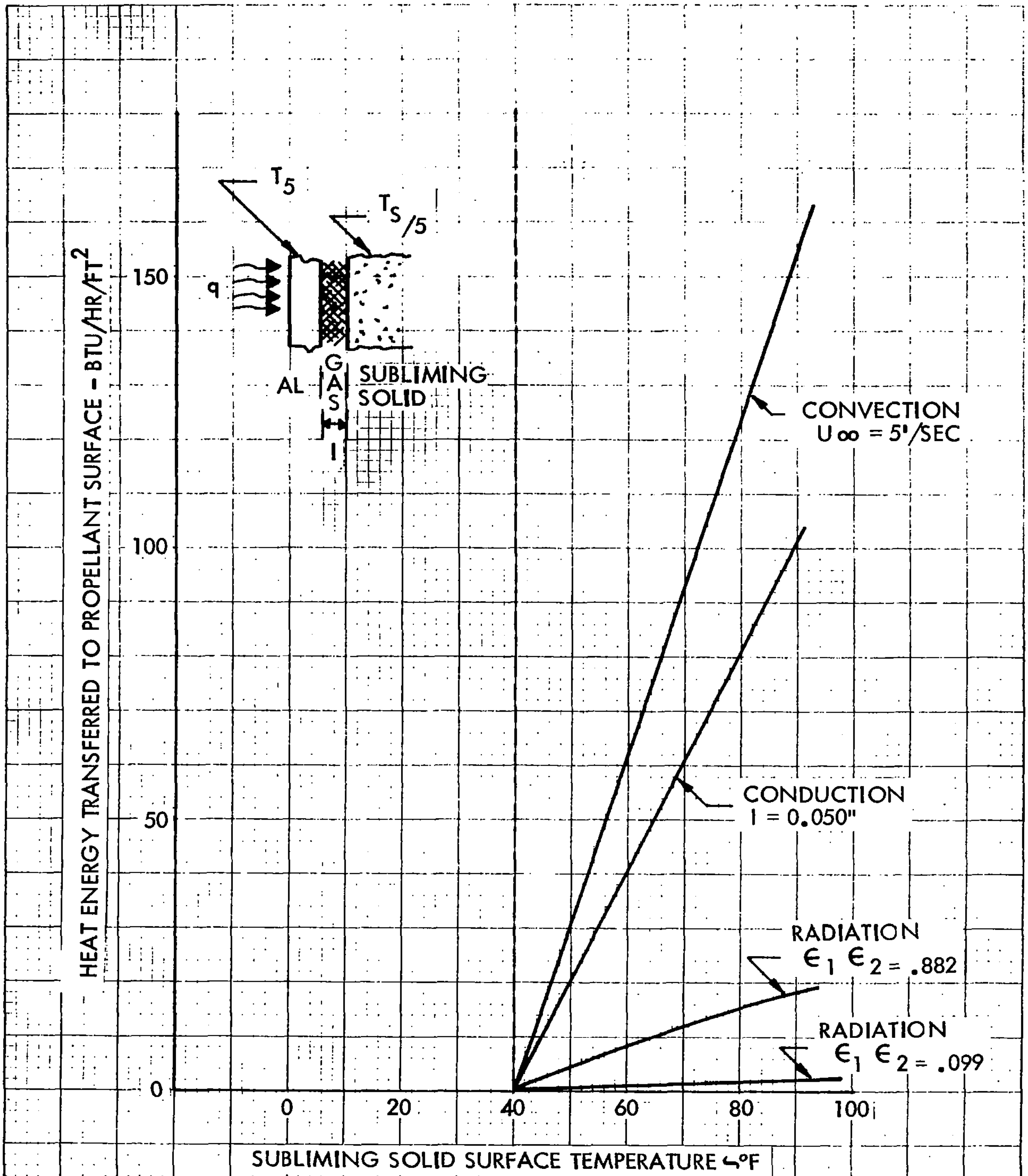


FIGURE 2-8. COMPARISON OF HEAT ENERGY TRANSFERRED TO PROPELLANT SURFACE BY CONVECTION, CONDUCTION, AND RADIATION (TANK WALL TEMPERATURE 40°F)

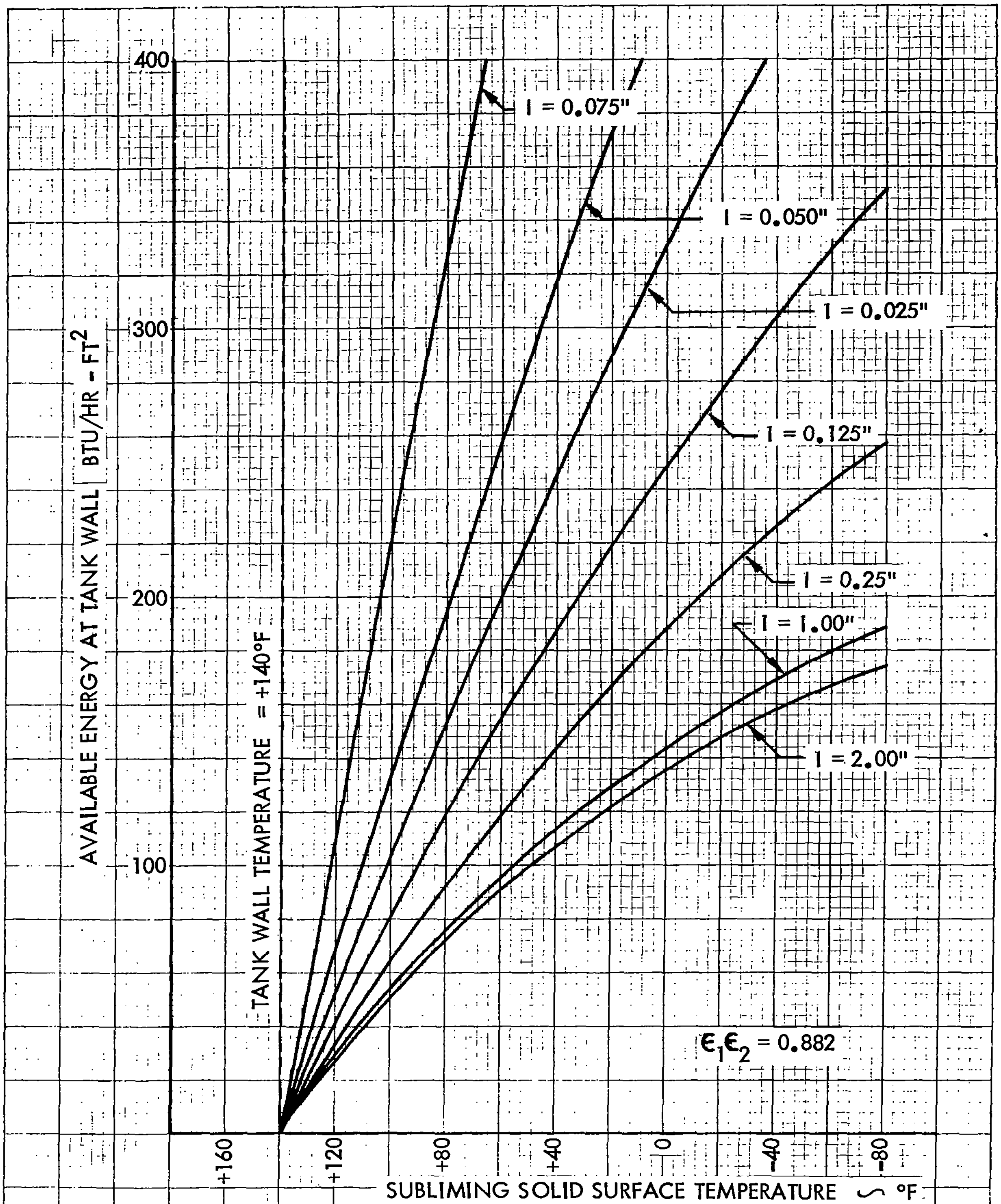


FIGURE 2-9. COMBINED CONDUCTION AND RADIATION HEAT TRANSFER BETWEEN TANK WALL AND PROPELLANT SURFACE (TANK WALL TEMPERATURE +140°F)

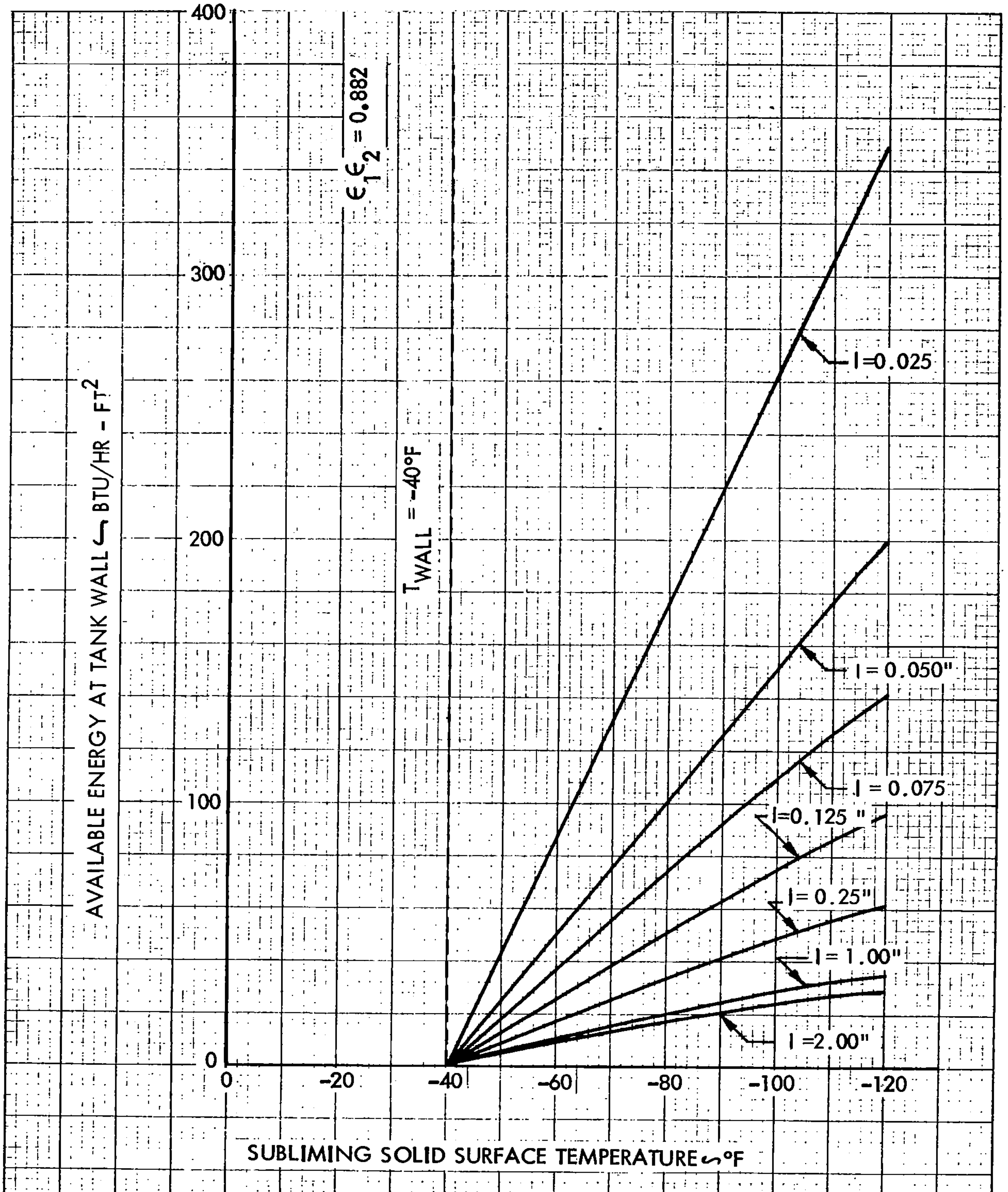


FIGURE 2-10 COMBINED CONDUCTION AND RADIATION HEAT TRANSFER BETWEEN TANK WALL AND PROPELLANT SURFACE (TANK WALL TEMPERATURE -40 °F)

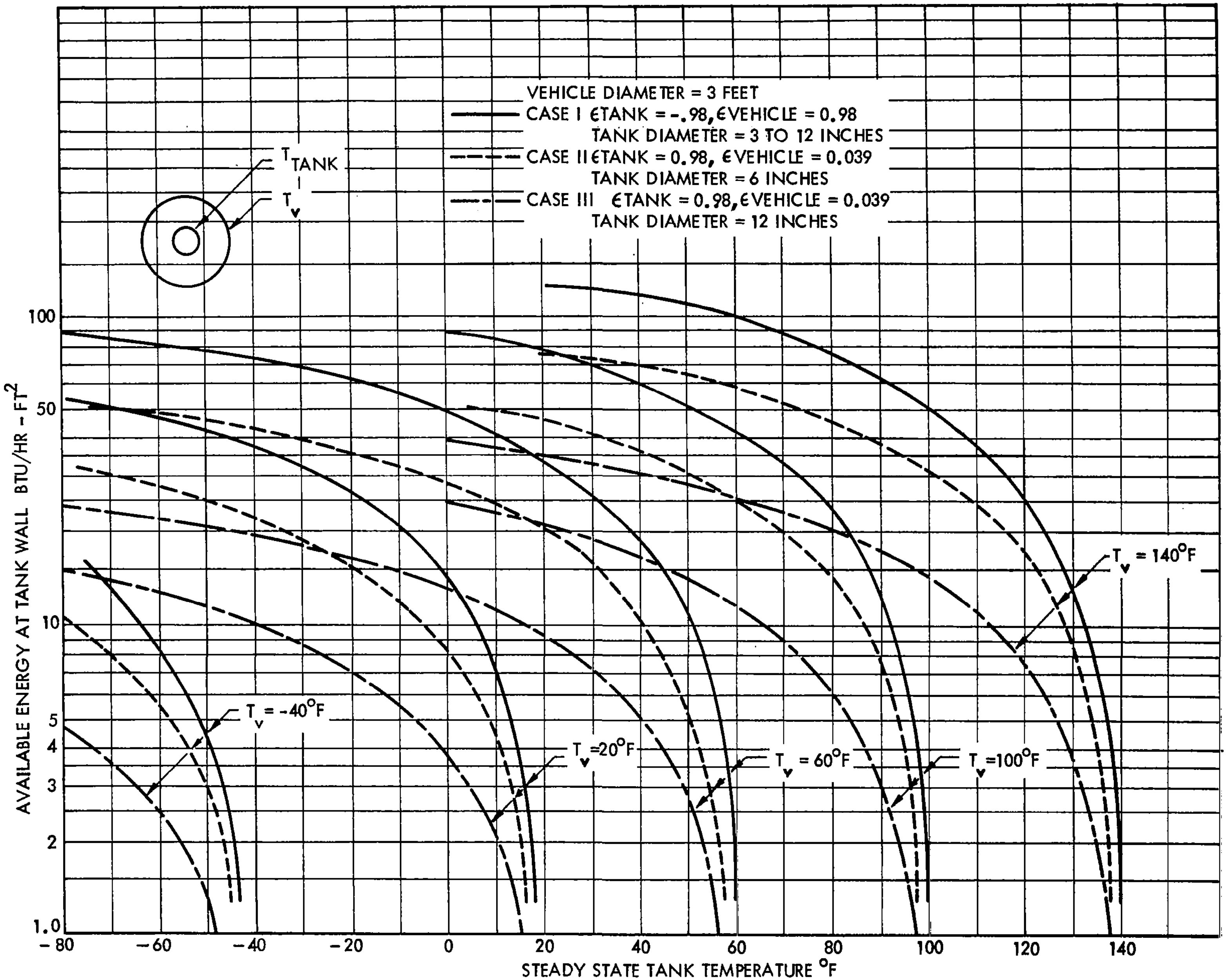


FIGURE 2-11 HEAT TRANSFER BY RADIATION FROM VEHICLE WALL TO TANK WALL

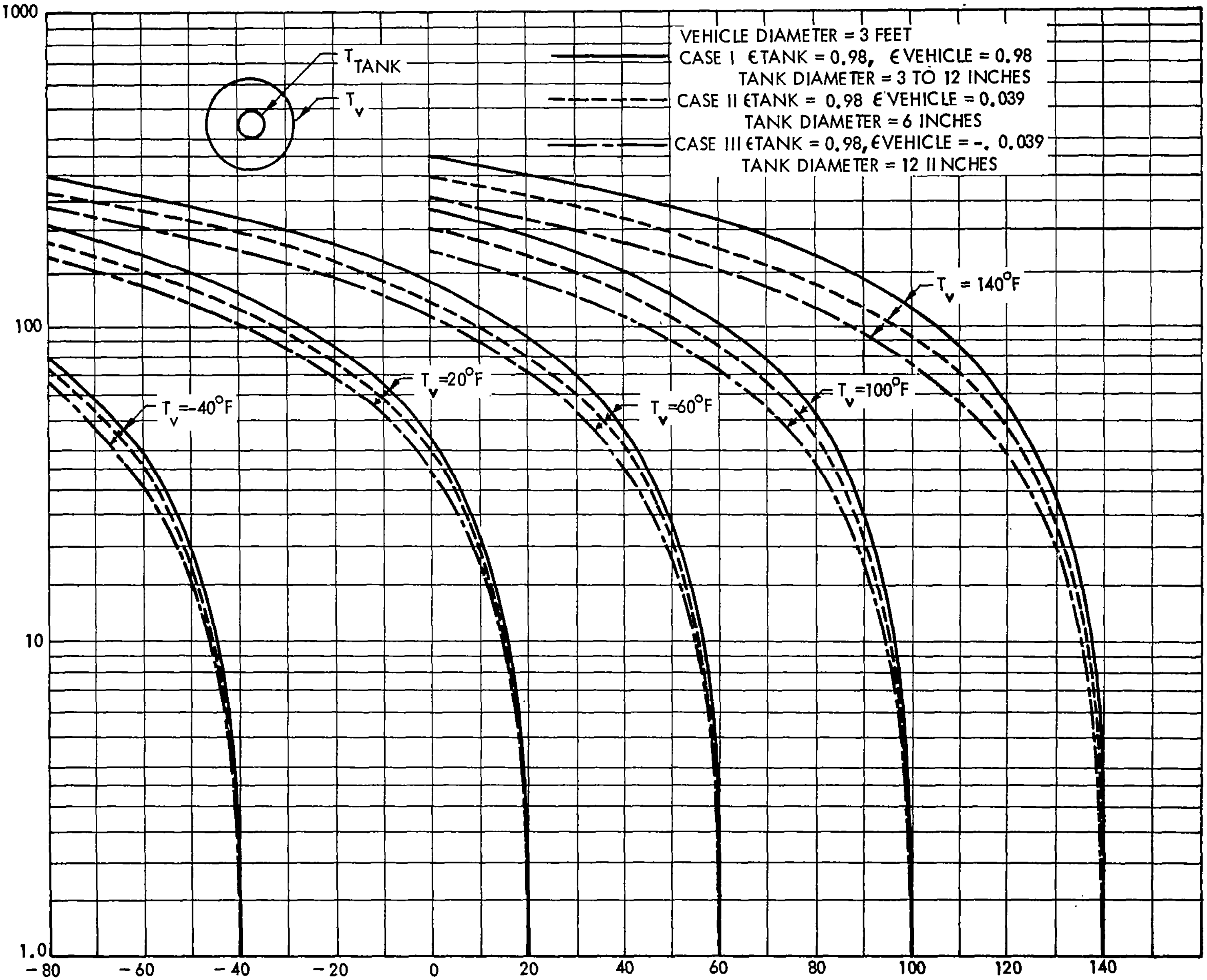


FIGURE 2-12. COMBINED HEAT TRANSFER BY RADIATION AND CONDUCTION FROM VEHICLE WALL TO TANK WALL

Heat Transfer Mode	Ambient Temp.	Equilibrium Propellant Temp.	Equilibrium Propellant Pressure	Equilibrium Thrust Level	Duty Cycle
Radiation	100° F	20°F	1.1 psia	2×10^{-3} lb.	Continuous
Conduction	100°F	51°F	3.7 psia	7.5×10^{-3} lb.	Continuous
Conduction	56°F	34°F	1.7 psia	3.5×10^{-3} lb.	Continuous

TABLE 2-10

CONDUCTION AND RADIATION HEAT TRANSFER FOR SUBLEX A used. Vehicle temperatures from -40°F to +140°F were considered. Figure 2-11 illustrates heat transfer by radiation only, and Figure 2-12 illustrates heat transfer by a combination of radiation and conduction through a propellant tank mount of good thermal conductivity. The available energy for sublimation at the propellant tank wall is plotted versus the propellant tank temperature.

The information presented on the heat transfer curves (Figures 2-6 through 2-12) can be used in the following manner to size a subliming solid rocket:

First, the desired thrust level, minimum vehicle temperature, propellant tank dimensions, and the expected modes of heat transfer are established. The required heat transfer rate is then obtained for the given thrust level from Figure 2-6, and effective surface area of the propellant tank is calculated. Once the required heat transfer rate, propellant tank surface area, and vehicle temperature are determined, Figures 2-11 and 2-12 are used to determine the propellant tank temperature, and Figures 2-7 and 2-10 to determine the propellant's surface temperature. The propellant vapor pressure at that temperature is obtained from the vapor pressure versus temperature curve, Figure 1-2. This vapor pressure and the desired thrust level can be used to calculate minimum restricting orifice size. The thrust level which can be obtained from any subliming solid rocket with a fixed orifice size may likewise be determined by following the process outlined in reverse.

2.11.4 Heat Transfer Tests

A second series of heat transfer tests were conducted to correlate the heat transfer calculations performed to actual experimental fact.

The test chamber used for these heat transfer experiments consisted of a vacuum chamber five feet in diameter by eight feet long with a two andone-half foot diameter by three foot long environmental shroud, painted black, installed inside. The shroud was wrapped with copper coils and used either hot or cold liquids to enable the establishment of temperature environments ranging from -60°F to +150°F within the shroud. The vacuum chamber had a vacuum capability of approximately 10^{-2} millimeter mercury. A six inch propellant tank containing SUBLEX A was placed in the environmental shroud.

The first test was with the path of heat transfer to the propellant tank restricted to radiation alone. This test showed that at an ambient temperature of 100°F and equilibrium propellant temperature of 20°F, a vapor pressure of 1.1 psia is maintained when the system is operated on a continuous duty cycle. This corresponds to a continuous thrust level of 2×10^{-3} pounds for a system with a room temperature thrust of 1×10^{-2} pounds.

The second test included both radiation and conduction. For conduction, an aluminum bar was attached to the environmental shroud on which the propellant tank being tested was mounted with good thermal contact. The test was performed with an ambient room temperature (60°F) and a thrust of 1×10^{-2} pounds. Table 2-10 lists the results of this test.

It should be noted that some departure from these results can be expected under a g environment. However, these differences, due to momentary lack of contact between the propellant and tank wall, are not expected to be large. Since the propellant will be floating freely, any thrusting will move the satellite and bring the propellant in contact with the tank wall.

A test was also performed to determine the temperature variation of SUBLEX A at various points within the propellant tank. This was accomplished by placing six thermocouples at random throughout a tank partially filled with SUBLEX A propellant. The environmental temperature was raised to 100°F and then allowed to cool back to room temperature. A maximum variation of 7°F was observed between the highest and lowest thermocouple as the propellant tank cooled. No propellant was flowed during the test.

2.11.5 Comparison of Analytical Calculation and Actual Test Results

The actual results obtained from experimental tests proved, in most cases, that analytical heat transfer calculations are conservative. Figures 2-7 through 2-12 should be used to give only a preliminary indication of the minimum thrust level possible for a given set of heat transfer conditions since a number of complex conditions are not accounted for in the calculations. Testing should always be performed, in critical cases, to determine the exact thrust level of a flight prototype in its expected thermal environment. However, it is felt that the agreement between calculation and experiment was close enough to rely upon analytical heat transfer calculations to obtain an approximate figure for allowable continuous thrust or thrust level/duty cycle relationship.

For example, with heat transfer by radiation alone, an emissivity of 0.98, vehicle temperature at 100°F, and the propellant tank surface at 20°F, experimental heat transfer was at the rate of 84 BTU's per hour per square foot of tank area for the equilibrium thrust level. This compares with 80 BTU's analytically calculated per Figure 2-11.

Experimental heat transfer was at the rate of 315 BTU's per hour per square foot for combined radiation and conduction, with the tank and shroud in good thermal contact. Under similar conditions, the analytical curves (Figure 2-12) predicted 260 BTU's per hour when corrected for equivalent conduction path conductivity.

2.12 Filter Effectiveness and Plugging

Tests were performed to determine the degree of filter plugging when the filter was exposed to a subliming solid propellant. Equipment required for the test was set up as shown on Figure 2-13. The tests were also run under the following modified conditions:

a. The filter holder was chilled with dry ice.

b. The system was inverted so that the propellant remained in contact with the filter for the duration of the test. The filter was not chilled so it operated at the same temperature as the tank. Heat of sublimation was supplied by wrapping the tank with heater tape during these inverted flow tests while the filter was not heated.

The tests, as shown on Figure 2-13, were performed as follows:

a. Evacuate the entire system.

b. Measure the normal pressure drop across the filter with the manometer by flowing momentarily after the propellant has returned to room temperature.

c. Invert the container in order to bring the propellant into direct contact with the filter.

d. Measure pressure drop across the filter to determine degree of plugging.

e. Heat the filter by pumping hot water through the holder.

f. Measure pressure drop again, with and without direct propellant contact.

g. Cool the filter with cold water and repeat procedure.

Ten and 20 micron stainless steel screens and five and 17 micron Regimesh stainless steel screens were tested as described above at a low temperature with SUBLEX A

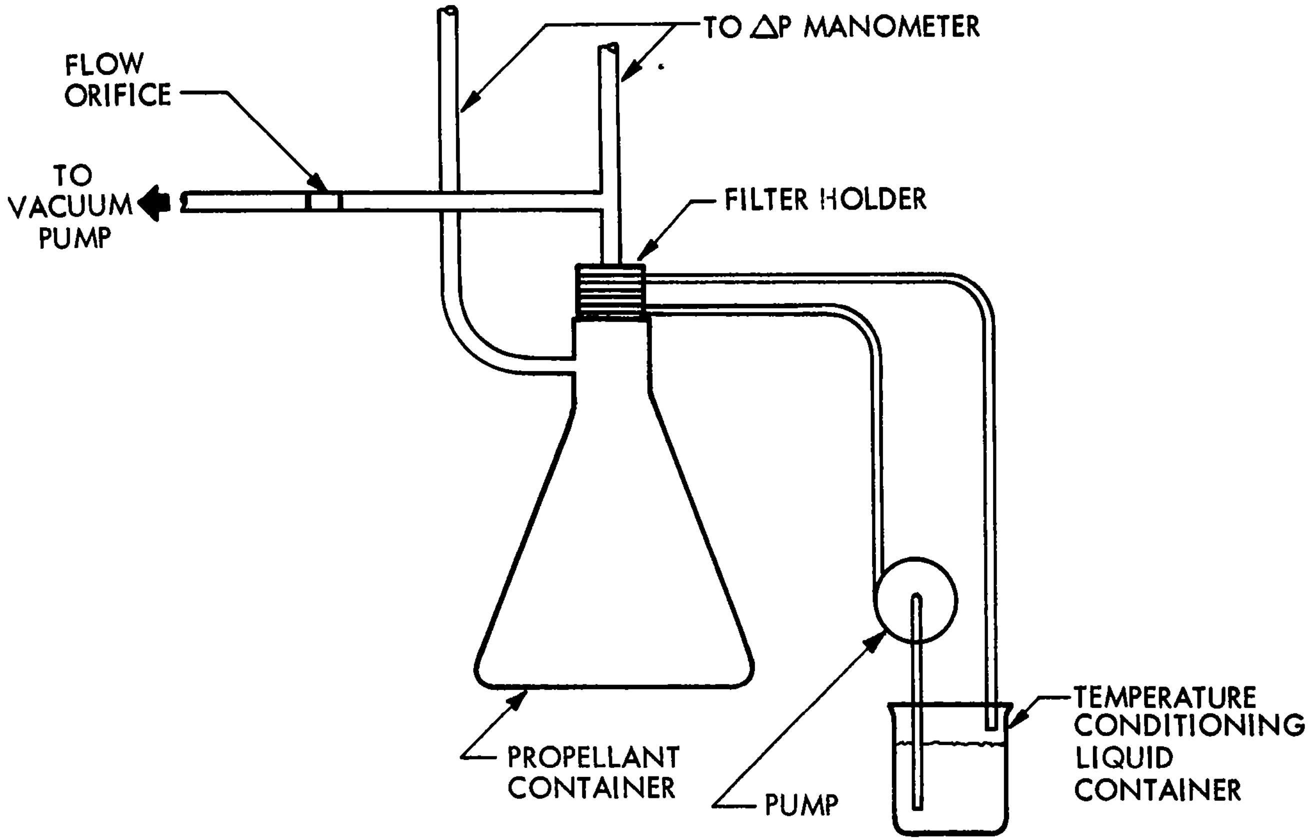


FIGURE 2-13. FILTER EFFECTIVENESS AND PLUGGING TEST EQUIPMENT SETUP

propellant. The 10 micron stainless steel screen plugged at 0°F after 20 minutes, but only in the inverted position. The 20 micron stainless steel screen and the two Regimesh screens did not plug, although they were heavily coated and did exhibit some pressure drop after 20 minutes, at 0°F. See Table 2-11 for test results.

A series of five long period tests were also performed on inverted filter assemblies with powdered SUBLEX A completely covering the filter assembly for the duration of the tests. The filter assembly consisted of 1-1/8 inch diameter, 50, 100, and 200 mesh stainless steel screens in series. The longest of these five tests lasted three hours. Constant propellant temperature was maintained under flow for the full test period through the use of heater tapes. In no case did plugging of the filter assembly occur. The immersion of the filter assembly in the propellant powder simulates flow conditions of a zero g environment.

A sixth inverted filter effectiveness test was performed using degraded propellant which was obtained by loading the propellant tank in air. The object of this test was to include the effect of nonvolatile powdered residue on filter operation. During this test a small differential pressure built up across the filter at the very end of the test when all but a small percentage of the propellant had been exhausted. When examined, a small quantity of powdered nonvolatile material was attached to the screens. This was primarily on the 200 mesh screen at the bottom of the assembly.

2.13 Variable Orifice Evaluation

Two thermally controlled variable orifices, built by Pyrodyne, Inc., Los Angeles, California, were tested to determine feasibility for SSRCS use. The tests produced inconsistent results and indicated improper orifice operation.

The first orifice, sized to maintain a flow rate of 1.25×10^{-5} lbm/sec (corresponding to a thrust level of 10^{-3} lbf), was installed in a SUBLEX A propellant tank as shown on Figure 2-14. A line was connected from the orifice to a vacuum pump and the choking orifice (with a U-tube manometer placed immediately upstream) was inserted in the line. The choking orifice was sized large enough to assure that the variable orifice remained choked under all conditions. Under flow conditions, the propellant was heated from 64°F to 100°F and then allowed to cool. Proper orifice operation would have been indicated by a constant line pressure upstream of the choking orifice, however, the results proved this was not the case. The line pressure varied almost directly with tank pressure indicating very little change in the variable orifice area, which means essentially no movement of the actuator.

TEST NO. 1

50, 100 AND 200 MESH SERIES FILTER

Time		Tank Wall Temperature °F		Tank Pressure				ΔP Across Filters		Flow Pulse Duration	Remarks
				Start		End					
Hour	Min	Start	End	In. Hg.	psia	In Hg.	psia	In.Hg.	psia		
	0	64				8	3.9	0	0	Continuous	
	10	32	75	2.4	1.15	19	9.4	0	0		
	35	75	75	1.9	0.94	18.3	8.1	0	0		Pump on and Subliming for 30 minutes.
1	40	71	71	1.0	0.49	17.4	8.65	0	0		
2	45	70	78	1.0	0.49	16.4	8.1	0	0		
4	20	72		1.0	0.49			0.1	0.05		Propellant gone

TEST CONDITIONS: Equipment set up as shown on Figure 2-13. Series filters of 50, 100 and 200 mesh. Propellant SUBLEX A (previously exposed to air.)

Barametric Pressure - 29.89

Room Temperature - 70°F

RESULTS: Very little residue remaining at end of test. Large amount of powder between 100 and 200 mesh screens.

TABLE 2-11

RESULTS OF FILTER PLUGGING TESTS

Sheet 1 of 5

TEST NO. 2

50, 100, AND 200 MESH SERIES FILTERS

Time		Tank Surface Temperature EMF	Tank Vacuum Pressure In. Hg		Δ P Across Tank In. Hg	Remarks
Hour	Min		Left Leg	Right Leg		
0	00	0.50	12.5	-12.75	--	Rubber flask seal came loose. Propellant yellow.
0	05	1.27	8.0	- 8.45	--	
0	07	1.60	8.3	- 8.75	--	Vacuum pump engaged.
0	09	1.50	8.3	- 8.80	0.05	
0	12	1.77	8.11	- 8.85	0.05	
0	22	1.87	7.95	- 8.40	0.05	
0	36	2.00	8.72	- 9.20	0.05	Very little condensate in flask.
0	46	2.02	8.80	- 9.30	0.05	
1	04	2.42	9.20	- 9.70	0.05	Constant power input to heater tape.
1	21	2.64	9.35	- 9.80	0.05	
1	31	2.83	9.65	-10.15	0.05	
1	44	2.05			0.05	Tapping flask caused temperature to drop.
1	51	2.00	9.80	-10.30	0.05	Equilibrium pressure

TEST CONDITIONS: Room Temperature - 69°F

Barametric Pressure -30.23

RESULTS: Twenty percent of propellant remaining in tank (color white).

TABLE 2-11 (Continued)
RESULTS OF FILTER PLUGGING TESTS

Sheet 2 of 5

TEST NO. 3

10 MICRON FILTER

Time		Filter Holder Temperature °F.	Δ P Across Filter		Remarks
Min	Sec		In. Hg.	psia	
	0	0	0.2	0.98	
	10	0	0.2	0.98	
	30	0	0.4	0.196	
1	00	0	0.6	0.294	
2	00	0	0.8	0.391	
5	00	0	0.9	0.441	
10	00	0	1.0	0.491	
40	00	0	1.3	0.638	

TEST CONDITIONS:

Propellant - SUBLEX A
Room Temperature -69°F.
Tank Temperature -68°F.
Flow Pulse Duration - continuous
Test Duration -40 minutes

TABLE 2-11 (Continued)
RESULTS OF FILTER PLUGGING TESTS

Sheet 3 of 5

TEST NO.4

40 MICRON FILTER

Time		Filter Holder Temperature °F.	Δ P Across Filter		Remarks
Min	Sec		In. Hg.	psia	
	10	0	0	0	Flask heated with lamp 16 inches from flask.
	30	0	0	0	
1	00	0	0	0	
2	00	0	0.4	0.196	
5	00	0	0.5	0.245	
10	00	0	1.0	0.491	
30	00	0	1.0	0.491	
FLASK INVERTED					
1	00	0	0.1	0.491	
2	00	0	0.2	0.098	
3	00	0	0.3	0.147	
5	00	0	0.4	0.196	
10	00	0	0.5	0.245	Heavy condensate on lip of holder.
20	•00	0	0.6	0.295	
30	00	0	0.6	0.295	
TEST CONDITIONS:			Flow pulse duration - continuous Tank Temperature -68°F. Test Duration -1 hour		

TABLE 2-11 (Continued)

RESULTS OF FILTER PLUGGING TESTS

Sheet 4 of 5

TEST NO. 5

40 MICRON POUROUS STAINLESS STEEL FILTER

Time		Filter Holder Temperature °F.	Δ P Across Filter		Remarks
Min	Sec		In. Hg.	psia	
0	10	68	0.4	0.196	
0	30	68	0.4	0.196	
1	00	68	0.4	0.196	
2	00	68	0.4	0.196	
5	00	68	0.4	0.196	
10	00	68	0.4	0.196	
30	00	68	0.4	0.196	No deposit on filter.
INVERTED FLASK					
0	10	70	2.4	1.18	
0	30	70	2.4	1.18	
1	00	70	3.0	1.47	
2	00	70	3.6	1.74	
5	00	70	4.4	2.16	
10	00	70	5.6	2.74	
30	00	70	3.8	1.86	No deposit on filter.
TEST CONDITIONS:			Flow pulse duration – continuous Test duration –1 hour		

TABLE 2-11 (Continued)

RESULTS OF FILTER PLUGGING TESTS Sheet 5 of 5

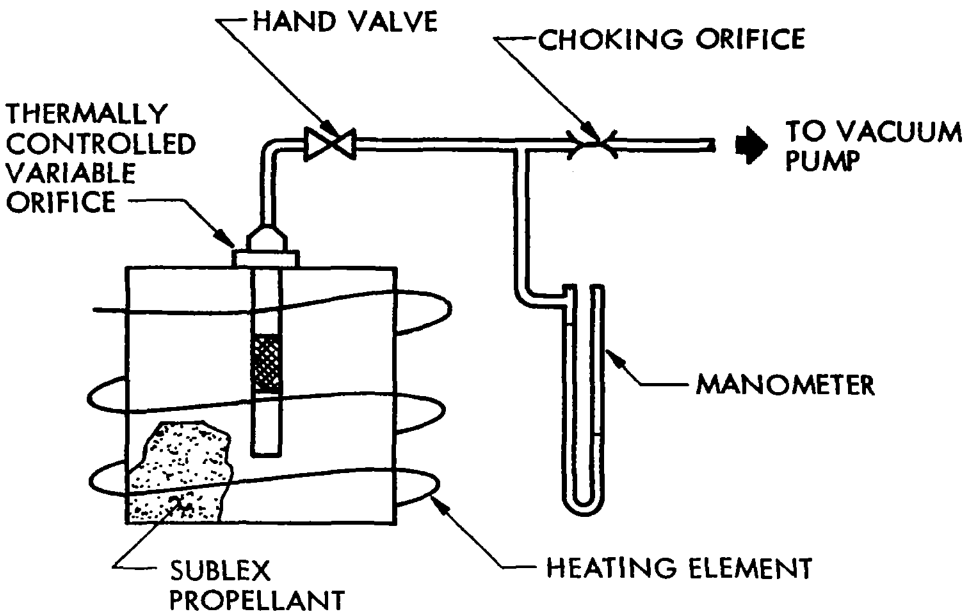


FIGURE 2-14. EQUIPMENT SETUP FOR EVALUATION OF VARIABLE ORIFICE INSTALLED ON SUBLEX A PROPELLANT TANK

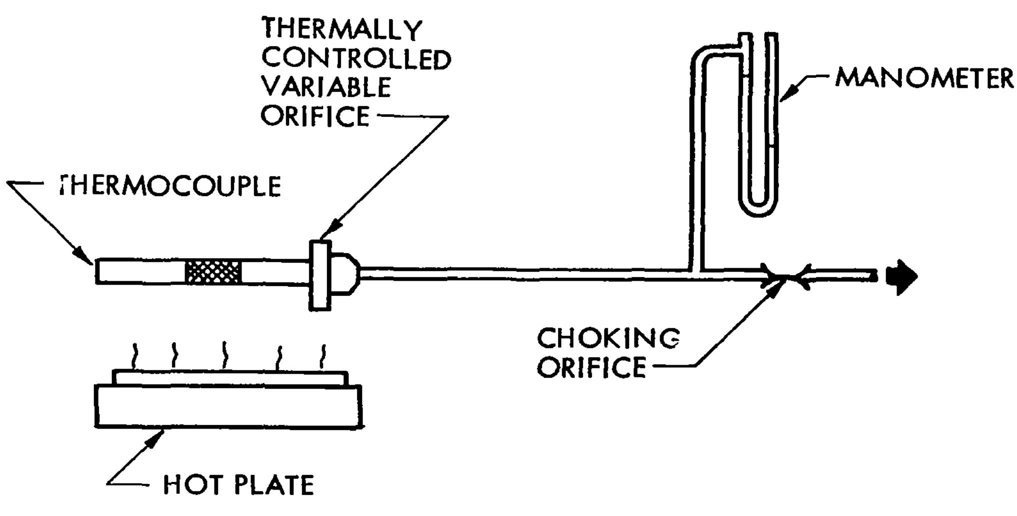


FIGURE 2-15. VARIABLE ORIFICE OPERATION IN AIR

The thermally controlled orifice was removed from the SUBLEX propellant tank, cleaned, and installed in a system operated in air as shown on Figure 2-15. The orifice expansion element was slowly heated from room temperature to 100°F and then cooled to room temperature. Proper orifice operation, in this case, would be indicated by a gradual change in pressure upstream of the choking orifice by approximately a factor of four as the temperature of the sensor changed from 70°F to 100°F. These tests indicated the valve was sticking, or hanging up, then suddenly breaking loose.

In one series of tests run in air, the pressure at the flow measurement orifice remained essentially constant at 2.4 psia, while the temperature was raised from 60°F to 100°F. At 100°F the variable orifice made a snapping noise with the pressure immediately dropping to 0.2 psia. The variable orifice sensor was allowed to cool to 95°F, where the pressure rose to 0.45 psia and remained relatively constant to 60°F.

The second thermally controlled variable orifice, designed to maintain a flow rate of 1.25×10^{-3} lbm/sec (corresponding to a thrust level of 10^{-1} lbf), was tested in air in a similar apparatus to Figure 2-15. This orifice performed much more smoothly than the first one, however, results indicated some sticking of the actuator and not as much change in orifice area as should have taken place.

Tests performed on the two thermally controlled variable orifices were not conclusive since the true cause of the problem was not determined. It appears, however, that either materials incompatibility, recondensation in the orifice area, or the design of the slide shaft connecting the thermal actuator with the variable orifice caused hang-up of the device. The latter reason is believed to be the most logical answer since the problem also occurred when the orifice was run in air. Although much development work is still required, it can be concluded the thermally controlled variable orifice still remains as a feasible solution for providing SSRCS thrust control.

2.14 Valve Ingestion

The three valve types (coaxial, poppet, and shear) were subjected to valve ingestion tests with the test equipment set up as shown on Figure 2-16. Separate tests using SUBLEX A propellant were performed on each of the three valves as described in Paragraphs 2.14.1 through 2.14.4.

Each valve was checked as follows:

a. Hand valves No. 1 and No. 2 were opened.

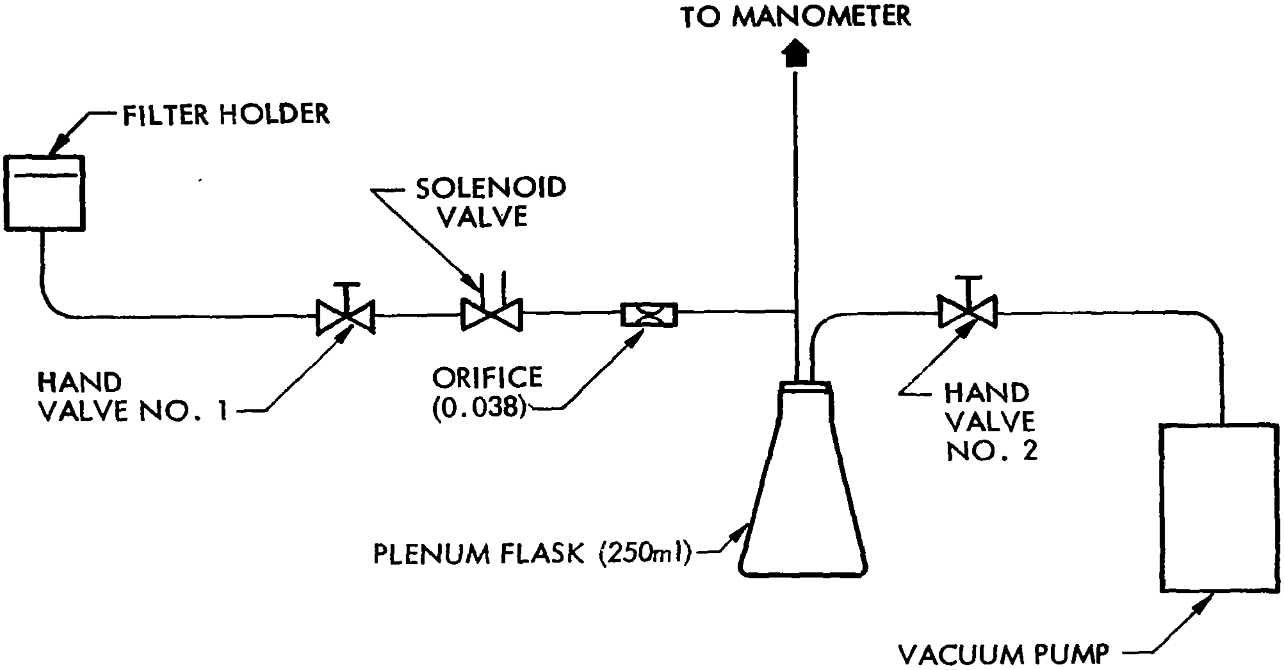


FIGURE 2-16. VALVE INGESTION TEST EQUIPMENT SETUP

b. The solenoid valve was pulsed 100 times.

c. The leak rate was checked by:

   (1) Closing hand valve No. 1

   (2) Evacuating the plenum

   (3) Closing hand valve No. 2

   (4) Opening hand valve No. 1

   (5) Monitoring pressure rise in plenum tank on manometer

d. The solenoid valve was pulsed 100 times while powdered propellant was pushed through the screen filter.

e. The leak check was repeated per step c.

f. Step d was repeated.

g. The leak check was repeated per step c.

2.14.1 Coaxial (Eckel) Valve - Integral Screen Installed

The equipment was set up as described in Paragraph 2.14 with a 50 mesh screen in the filter holder. The preliminary leak check showed no leakage after five minutes.

Next, propellant was finely crushed and distributed to a depth of 0.5 inch on top of the filter screen. The valve was then cycled while the propellant was pushed firmly against the filter screen. No propellant particles seemed to be penetrating the 50 mesh screen during cycling, so this screen was removed from the filter holder and replaced with a 25 mesh screen.

With the 25 mesh screen installed, fine propellant particles could be seen traveling toward the junction and accumulating at the junction of the valve and Tygon tube. A leak check performed at the end of 100 valve cycles showed no leakage after five minutes. This operation was repeated with no evidence of leakage.

2.14.2 Coaxial (Eckel) Valve - Integral Screen Removed

As a more severe leakage test, the filter holder and inlet tube apparatus were removed from the valve and the valve placed in a vertical position with its inlet port up. Propellant was firmly pushed directly into the inlet valve port while the valve was again cycled 100 times. No propellant particles were observed passing through the valve and no leakage occurred. Valve inspection revealed no screen plugging.

The valve was then allowed to stand without cycling for 72 hours. Again the valve was leak checked with no evidence of leakage. However, it was discovered the valve would not open when energized. Following valve disassembly it was evident the valve was sticking due to formation of a brittle yellow crust on the valve plunger and seat that was assumed to be degraded propellant. The valve was cleaned and re-assembled with the integral screen removed.

Following reassembly, the valve was cycled 100 times without propellant and subjected to a leak check, again showing no leakage after five minutes. Finely ground propellant was then poured to a depth of 0.5 inch over the 25 mesh filter screen. The valve was cycled 100 times while the propellant was pushed through the screen. An extended leak check revealed a leakage rate of only 0.1 psia per hour (0.4 cc per hour) at a pressure differential of 14.6 psia.

2.14.3 Poppet (Carleton) Valve

The equipment was set up as described in Paragraph 2.14 with a 25 mesh screen installed in the filter holder. A preliminary check showed this valve had a leakage rate of 0.4 psia per minute (800 cc per hour) at a pressure differential of 12.0 psia across the valve.

Next, the propellant was ground to a fine texture and poured on top of the filter to a depth of 0.5 inch. The valve was then cycled while the powdered propellant was pressed firmly against the filter screen. Immediately, small propellant particles could be seen faintly traveling down the three inch Tygon tube suspended in rushing inlet air. Valve leakage became so evident after 57 operating cycles that the air could be heard passing through the screen when the valve was in the closed position. When this large leakage occurred, the vacuum pump was only able to maintain a pressure differential of 5.0 psia across the valve.

This high leakage rate indicated the valve was sticking, or hanging up, in the open position between valve cycles. The cycling rate was reduced to allow five to ten seconds between open positions in order to permit the valve to close. Leakage at this slower cycling rate was 0.13 psia per second (15,000 cc per hour) at a pressure differential of approximately 10 psia across the valve.

The valve was inspected at the conclusion of the 100 cycle test which revealed a slight yellow residue on the inner surface of the inlet port. However, no residue was found on the outlet port or poppet. The 25 mesh filter was about 20 percent plugged.

2.14.4 Shear Seal (Valcor) Valve

The equipment was set up as described in Paragraph 2.14 with a 25 mesh filter installed. A preliminary leakage check showed no leakage after three minutes.

Finely ground propellant was then placed on top of the filter to a depth of about 0.5 inch. Next, the valve was cycled 100 times while propellant was firmly pressed against the filter. The leak check showed no signs of leakage after five minutes.

As a further check, the filter holder and inlet tube assembly were removed from the valve and propellant was poured directly into the valve inlet port. The valve was again cycled 100 times while propellant was poured, but not pressed, into the valve. The leak check still showed no leakage after five minutes. However, leakage did occur when the propellant was poured directly into the valve inlet port and pressed into the valve body during cycling. The leakage rate was measured at 0.67 psia per second (81,500 cc per hour) at a pressure differential of approximately 8 psia across the valve body.

At the conclusion of this leak check, the valve was momentarily inverted, allowing the propellant to fall from the inlet port. The leakage now was at a rate of 0.02 psia per second (2,300 cc per hour) at a pressure differential of 12.0 psia. The valve was again cycled an additional 100 times without inserting propellant into the inlet port. Following this cycling, leakage was at a rate of 0.1 psia per minute (200 cc per hour) at a differential of 12.0 psia.

The plenum flask was then washed, dried, and reassembled, but valve leakage remained 200 cc per hour. A visual inspection of the valve following these tests disclosed that no residue was visible, but valve ports were slightly moist.

2.15 Propellant Cake Fabrication by Recondensation

A series of tests were performed to determine the feasibility of filling a propellant tank by recondensation and to evaluate the properties of the resultant propellant cake. See Figure 2-17 for illustration of propellant cake fabrication by recondensation.

Equipment used for this test consisted of: A three inch propellant tank; a glass flask containing 360 grams of SUBLEX A propellant; a container with dry ice and water for cooling the tank; a heater tape around the line to prevent line recondensation; a vacuum pump to evacuate the system; and a vibration table. The aluminum line

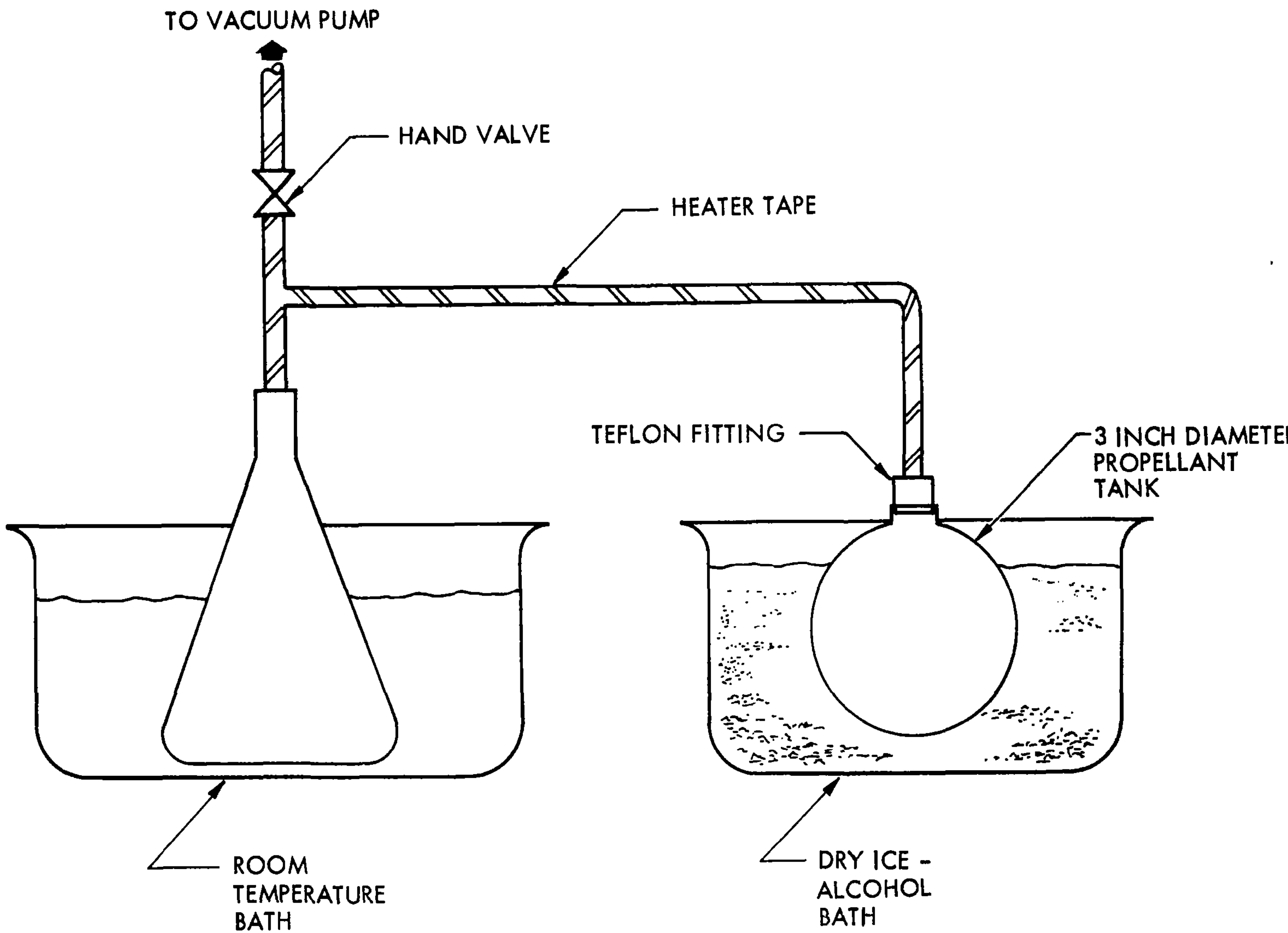


FIGURE 2-17. PROPELLANT CAKE FABRICATION BY RECONDENSATION

connecting the flask and the tank was passed through an insulated Teflon fitting near the center of the tank.

A recondensed propellant cake was obtained by immersing the aluminum tank slowly into the dry ice solution until propellant migration from the glass flask to the propellant tank ceased. Recondensation in the line connecting the flask and the tank was prevented by using a heater tape to maintain the line temperature above the point where recondensation could occur. The propellant migration from the flask to the tank took several days to successfully produce a propellant cake.

The recondensed cake, which appeared to completely fill the three inch tank, weighed 255.4 grams. This propellant cake had a density of 0.04 pounds per square inch, which compares with 0.027 pounds per square inch for a hand pressed powder.

The filled propellant tank was also subjected to vibration tests which duplicated the qualification levels for a Thor-Delta launch. The recondensed propellant cake showed no visible signs of breakup. See Paragraph 2.16.

As a result of this test, it was shown that a recondensed SUBLEX A cake has a greater density than hand pressed powder; also the recondensed cake successfully passed qualification level vibration tests for a Thor-Delta launch.

2.16 Vibration Testing

A SSRCS was subjected to a vibration test designed to simulate launch parameters of the Thor-Delta booster. These tests were performed at United Control Corporation, Bellevue, Washington. The test parameters and results are shown on Figure 2-18.

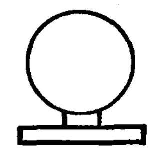
Following vibration tests, leakage increased from 25 cc per hour to 1100 cc per hour which varied with each valve cycling. However, the high leakage rate encountered was not due entirely to vibration testing. When the valve was disassembled, considerable propellant residue was found on and around the valve plunger and seat. This was probably due to prior valve testing which could account for the high leakage rate. The valve, however, did function properly following the vibration test in spite of the increased leakage. The propellant was also checked for possible property changes. There was no change in propellant color, density, or other properties, as a result of vibration testing.

VIBRATION FREQ. RANGE AND TYPICAL CYCLE				
AXIS	TIME MIN.	TIME SEC.	VIBR. FREQ. CPS	AC- G's
Z-Z	1	10	10-50	3.8
	1	40	50-500	7.5
	1	00	500-2000	21.0
X-X		25	10-18	3.0
	2	25	18-500	2.3
	1	00	500-2000	4.0
Y-Y		25	10-18	3.0
	2	25	18-500	2.3
	1	00	500-2000	4.0

AXIS OF VIBRATION

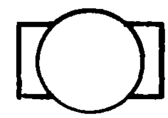
NO. 1, VERTICAL

Z-Z AXIS



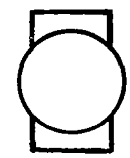
NO.2 HORIZONTAL

X-X AXIS



NO. 3 HORIZONTAL

Y-Y AXIS



TEST REMARKS: See sheets 2 through 4 for vibration graph.
Major resonance on X-X and Y-Y axis at 50 to 60 cps.
No resonance on Z-Z axis.
Propellant movement detected by sound between 10 and 600 cps.

POWER SPECTRAL DENSITY CURVE	
BANDWIDTH (CPS)	ACCELERATION DENSITY (G^2/CPS)
20-2,000	.07

TEST REMARKS: Total measured acceleration - 11.8 G's (Rms)

Test duration - 4 minutes per axis.

See sheet 5 for equalization curve.

FIGURE 2-18 VIBRATION TEST PARAMETERS AND RESULTS (SHEET 1 of 5)

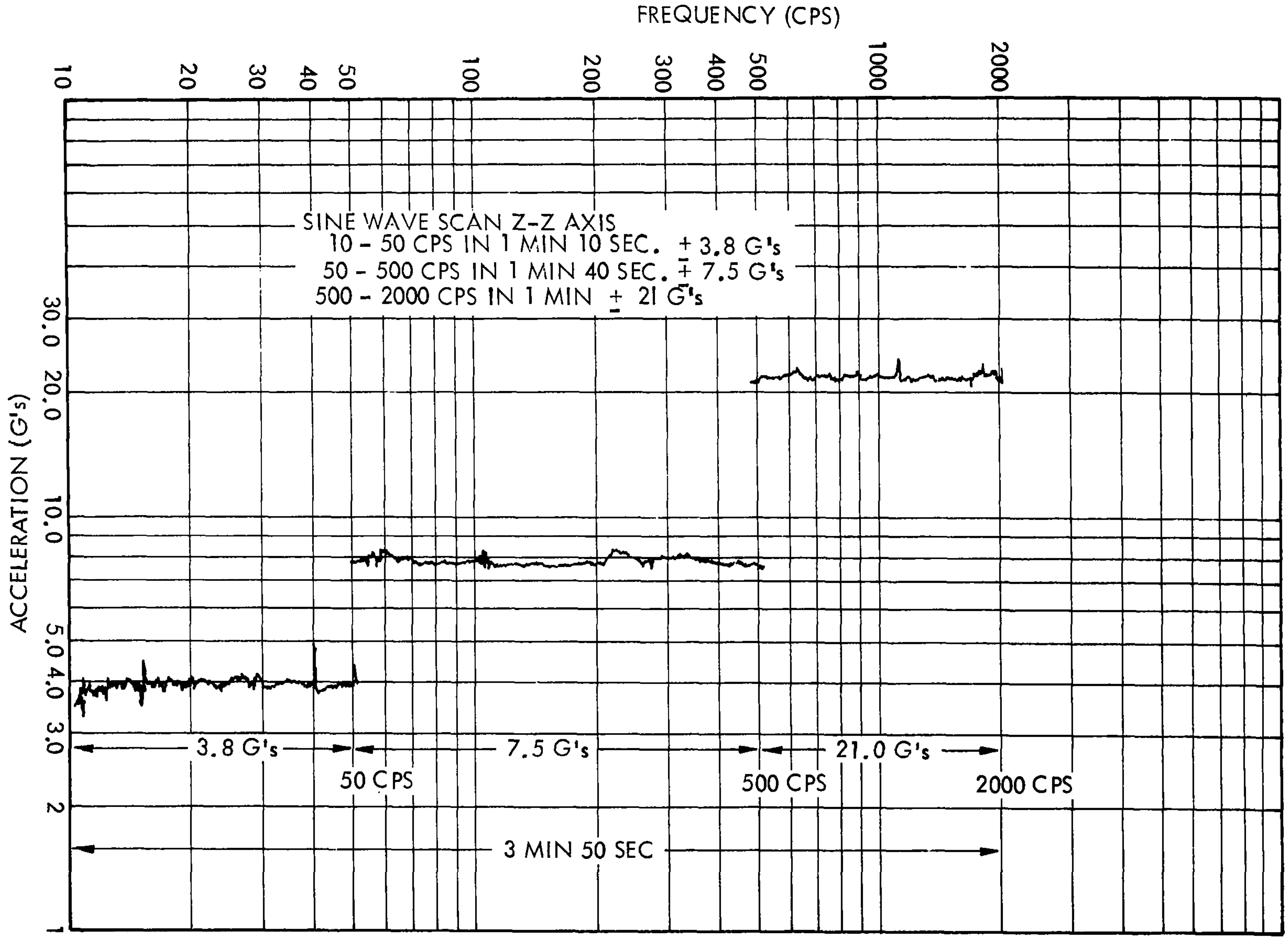


FIGURE 2-18. VIBRATION TEST PARAMETERS AND RESULTS (SHEET 2 OF 5)

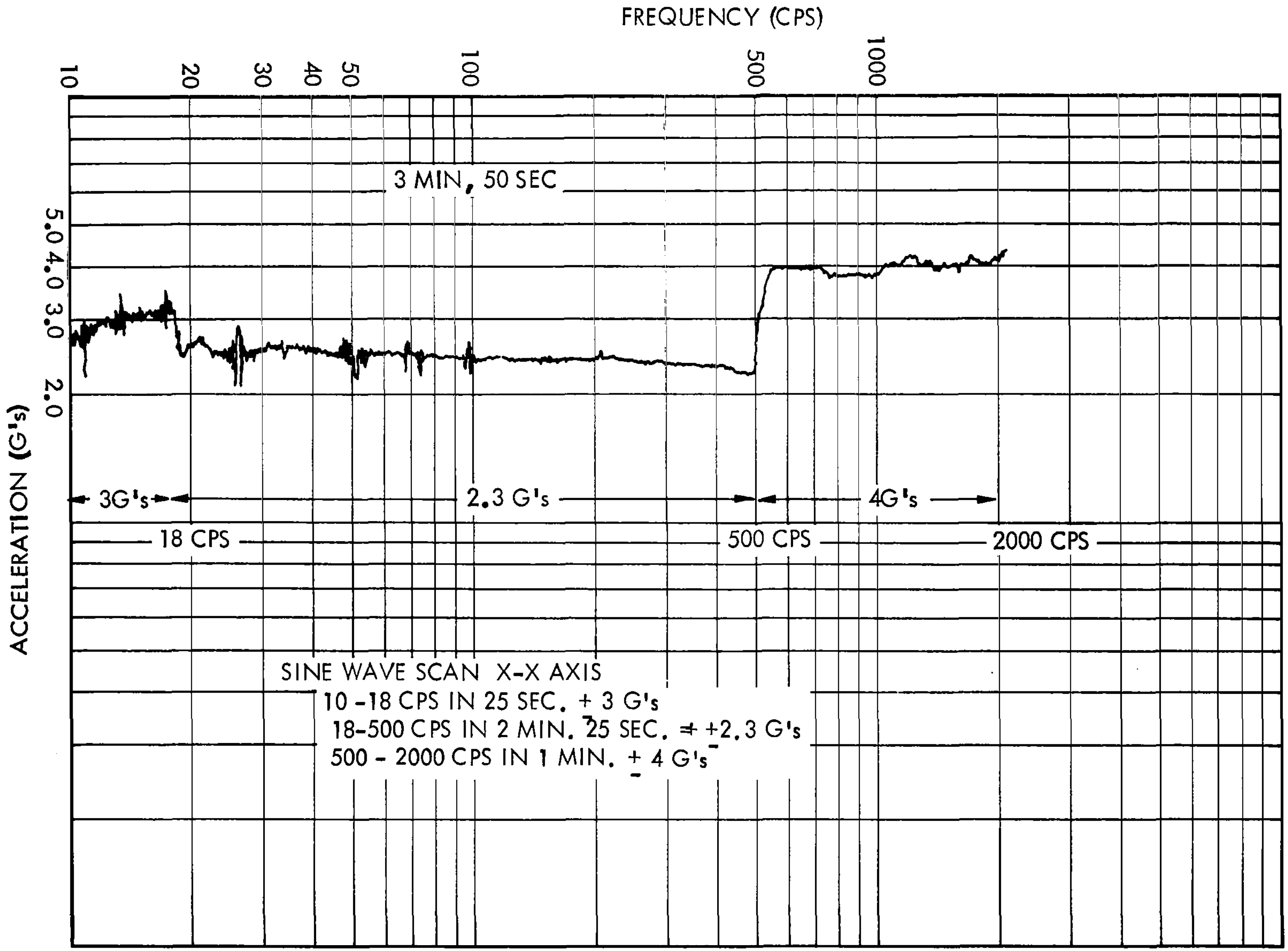


FIGURE 2-18. VIBRATION TEST PARAMETERS AND RESULTS (SHEET 3 OF 5)

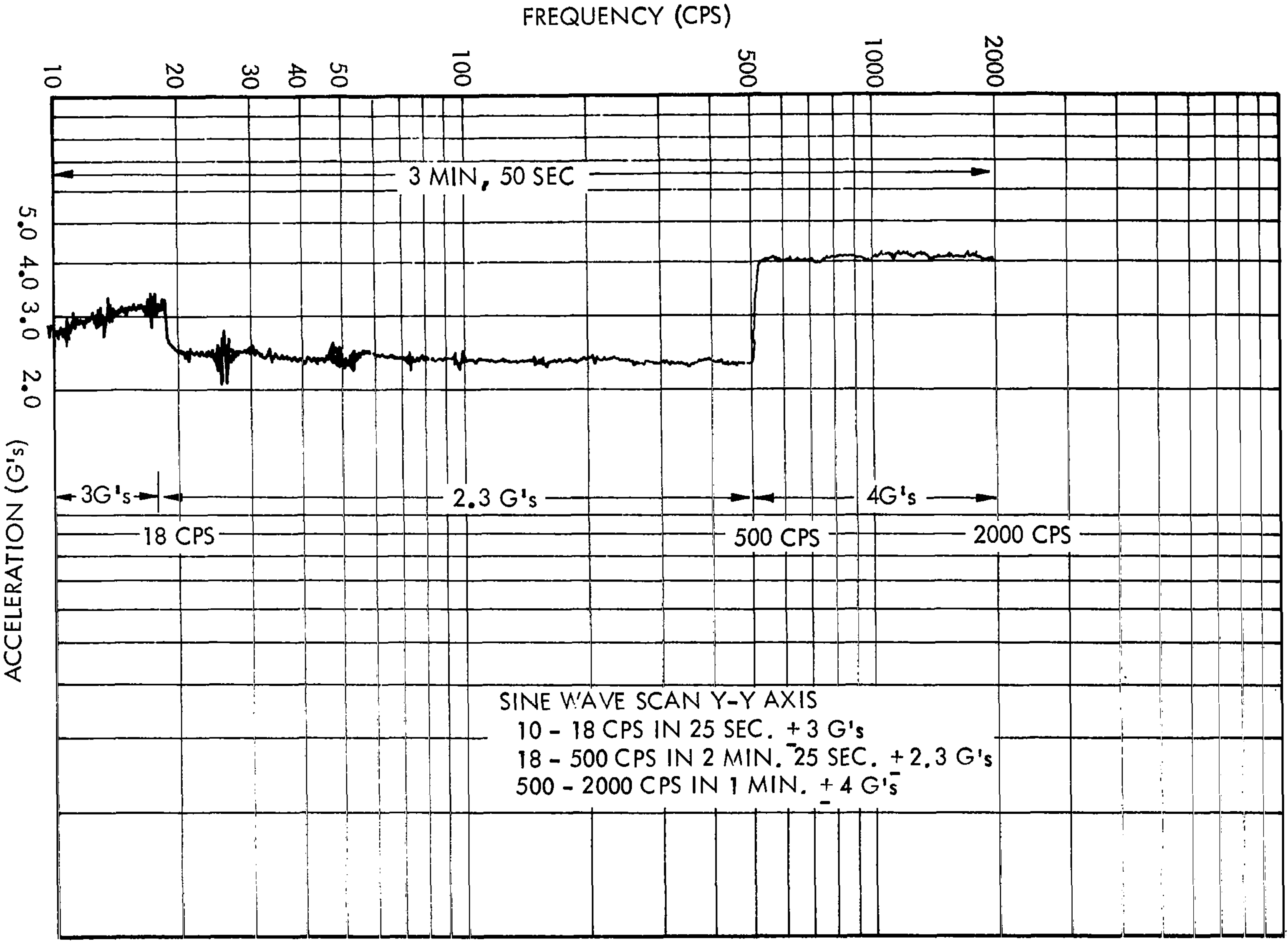


FIGURE 2-18. VIBRATION TEST PARAMETERS AND RESULTS (SHEET 4 OF 5)

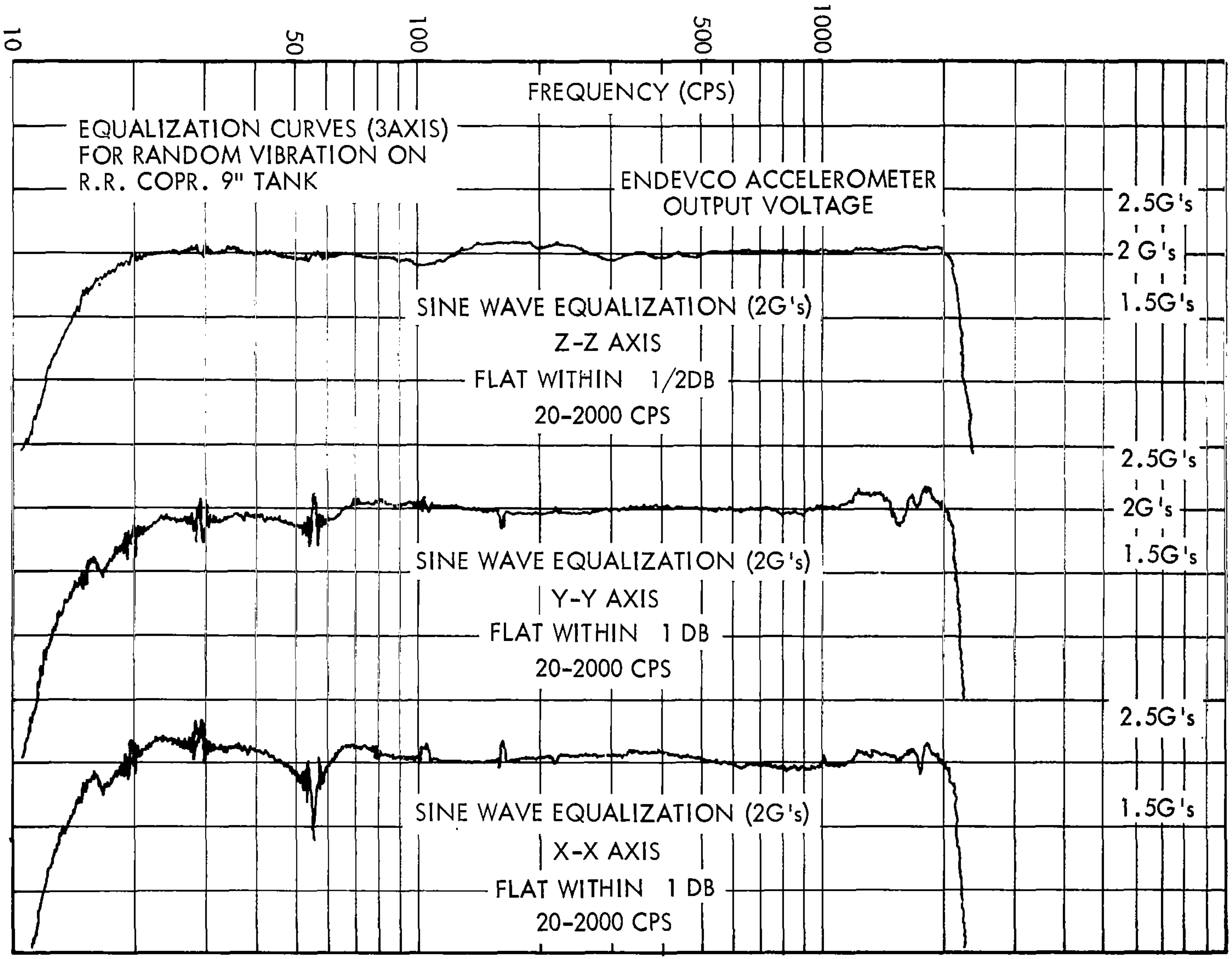


FIGURE 2-18. VIBRATION TEST PARAMETERS AND RESULTS (SHEET 5 OF 5)

2.17 System Testing

2.17.1 Sustained Pulsed Performance Test

The object of this test was to determine the effects of an extended operating period on a SSRCS. This test involved the use of the same system as previously subjected to vibration and specific impulse tests. See Figure 2-19 for equipment setup.

The SSRCS, with the propellant tank in the inverted position, was placed inside the environmental shroud within the vacuum chamber. A line was extended from the system nozzle to a vacuum pump outside the chamber. After evacuation of the vacuum chamber the vacuum pump was started and left operating for the 20 day test period with the system pulsed once a day at the rate shown on Table 2-12. Following each day's pulsed operation, the valve was checked for leaks. Leakage varied between 0.40 cc per hour to 2.0 cc per hour as shown on Table 2-12. The operating pressure and shroud temperature were also monitored during the test period.

Due to the successful test performance it was decided to let the system remain idle for an additional two weeks. Following the two week period, an attempt to pulse the system failed. After the valve had been disassembled, it was discovered the battery used to pulse the system was defective. Although some contaminant was found on the valve seat and walls when the valve was disassembled, it is believed that the valve would still have performed satisfactorily with a normal battery. This assumption was made after the valve was reassembled and pulsed with a good battery.

2.17.2 Related System Testing

Two laboratory test SSRCS's were subjected to performance testing prior to delivery to Lewis Research Center. The tests, although not a requirement of this contract, yielded data which is applicable. One control rocket, containing SUBLEX A propellant, delivered 1.0×10^{-2} pounds of thrust at 65°F and the second control rocket, containing SUBLEX B propellant, delivered 1.0×10^{-4} pounds of thrust at 67°F. Pressure and thrust were measured for pulse durations of 10, 20, and 30 seconds. At 1.0×10^{-4} pounds thrust, with a nozzle area ratio of 9:1 and 40° divergence angle, a specific impulse of 63 seconds was obtained.

System tests were conducted using the compound pendulum balance to measure thrust. A special plenum flow system was also used to measure propellant flow in the SUBLEX B system. The object of these tests was to verify proper operation and design thrust level and, in the case of the SUBLEX B system, to measure performance.

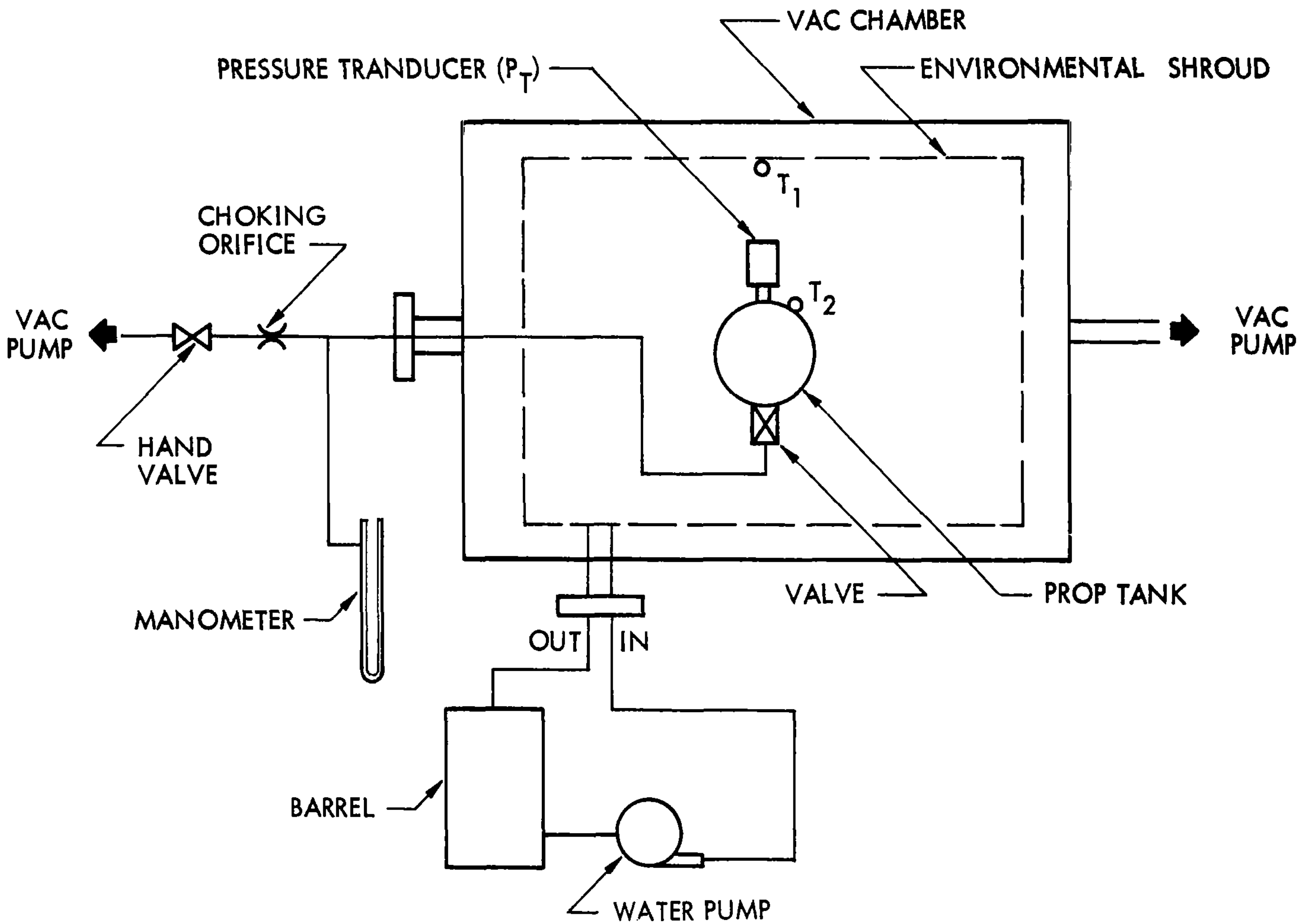


FIGURE 2-19. SUSTAINED PULSED PERFORMANCE TEST EQUIPMENT SETUP

Test Day	Total Pulse Time Min.	Number of Pulses	Pulse Duration		Shroud Temp °F		Tank Pressure psia		Valve Leakage Rate cc/Hour
			Min.	Sec.	Start	End	Start	End	
1	20	114		10	52.5	53	2.1		0.0
2	20	114		10	83	85	7.1	3.0	0.0
3		1	15	00					
	36	114		10	65	65	2.3	1.6	0.4
4	25	114		10	71	68	7.8	2.6	0.8
5	23	114		10	76	90	6.55	2.3	0.4
6	15	1	15	00	50	50	4.3	1.8	1.6
	20	114		10	51	48	5.3	2.0	0.8
7	24	114		10	62.5	61	6.1	1.75	0.8
8	20	114		10	65	63	5.9	2.1	0.4
9	15	1	15	00	92.5	90	6.5	2.9	2.0
	21	114		10	86	91	10.0+	3.4	1.2
10	20	114		10	65	48	5.8	2.1	1.6
11	21	114		10	64	61.5	5.5	2.1	1.6
12	15	1	15		60	59	5.6	2.0	1.6
	19	114		10	51	47	6.35		1.6
13	20	114		10	93	100	6.7	3.5	1.6
14	20	114		10	63.5	64	6.75	1.95	1.6
15	15	1	15	00	68	58	4.85	2.0	1.6
	18	114		10	72	70	7.4	2.3	1.6
16	19	114		10	72	67	6.4	2.0	1.2
17	19	114		10	100	100+	10.0+	2.8	1.6
18	15	1	15	00	100	100	10.0+	2.9	0.8
	20	114		10	100	95	10.0	2.7	0.8
19	19	114		10	65	62	6.2	1.85	1.6
20	19	114		10	65	63	6.2	1.75	1.6

TABLE 2-12

VALVE LEAKAGE RATE OVER A 20 DAY OPERATING PERIOD

In addition to the system test performed under this contract by Rocket Research Corporation, three other system tests are pertinent and should be mentioned. The first of these systems (Figure 2-20) is a demonstration Subliming Solid Control Rocket constructed by Rocket Research Corporation in August, 1963, to illustrate the principles of operation. The rocket consisted of a spherical propellant tank, manifold block, coaxial valve, and a nozzle. A target was located directly behind the nozzle exhaust to indicate operation of the rocket when the valve was actuated. The rocket was loaded with SUBLEX B propellant and operated periodically (about once every two weeks) for ten months without a single failure or degradation in performance.

The second system (Figure 2-21) is a laboratory test model of the Subliming Solid Control Rocket constructed by Rocket Research Corporation and delivered to Goddard Space Flight Center, NASA, in May, 1963. This rocket consisted of a single short cylindrical propellant tank and a coaxial valve, located at the outlet of the tank, with two nozzles attached. The propellant tank was loaded with one pound of SUBLEX A propellant and tested at Rocket Research Corporation prior to delivery. The system was kept in storage for ten months and then operated again. Prior to the first actuation of the valve after storage, heater tape was applied around the valve and the top neck of the tank for approximately 20 minutes. Valve actuation was then attempted, but failed. The heater tape was reapplied for approximately 40 minutes and valve actuation was successful. The unit was then operated for a long series of pulses of varying length and upon completion was placed in storage for another month. At the end of that time, an attempt was made to actuate the valve but was unsuccessful. Some corrosion was observed on the outside of the system, therefore, it was returned to Rocket Research Corporation for examination and servicing.

Rocket Research Corporation disassembled the system and found that the interior of the propellant tank was unharmed. The surface propellant had a small tinge of yellow indicating that some air had leaked into the system prior to opening of the tank, however, it is not known exactly when this occurred. The valve was removed from the tank and the exterior surfaces, as well as the interior surfaces that could be seen from the outside, showed evidence of rather severe corrosion. The valve electrical coil was intact so an attempt was made to operate the valve, but it was totally inoperative. The valve was then returned to the manufacturer, Whittaker Corporation,

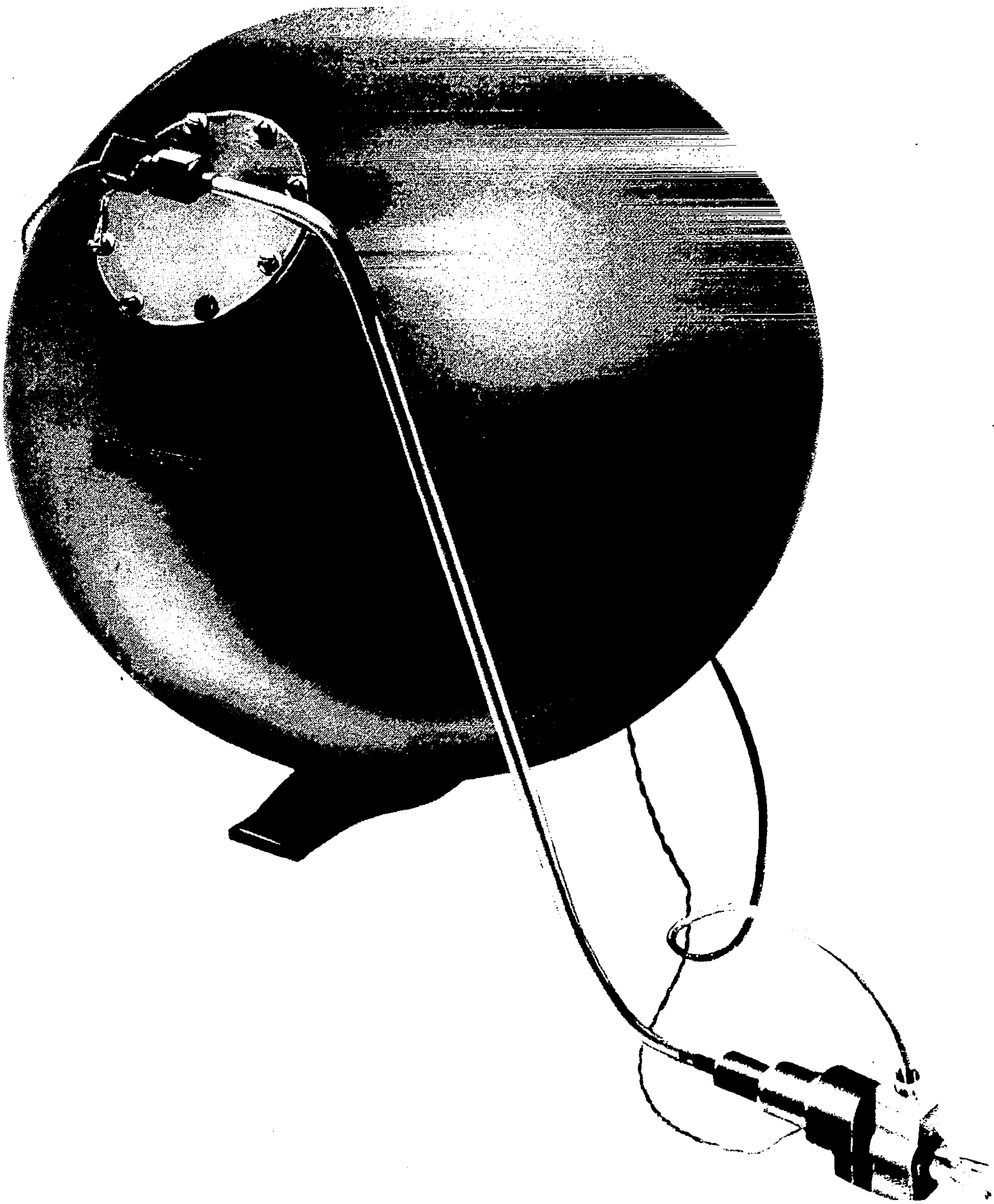


FIGURE 2-20 SUBLIMING SOLID μR-10-4SS-D LOW THRUST CONTROL ROCKET

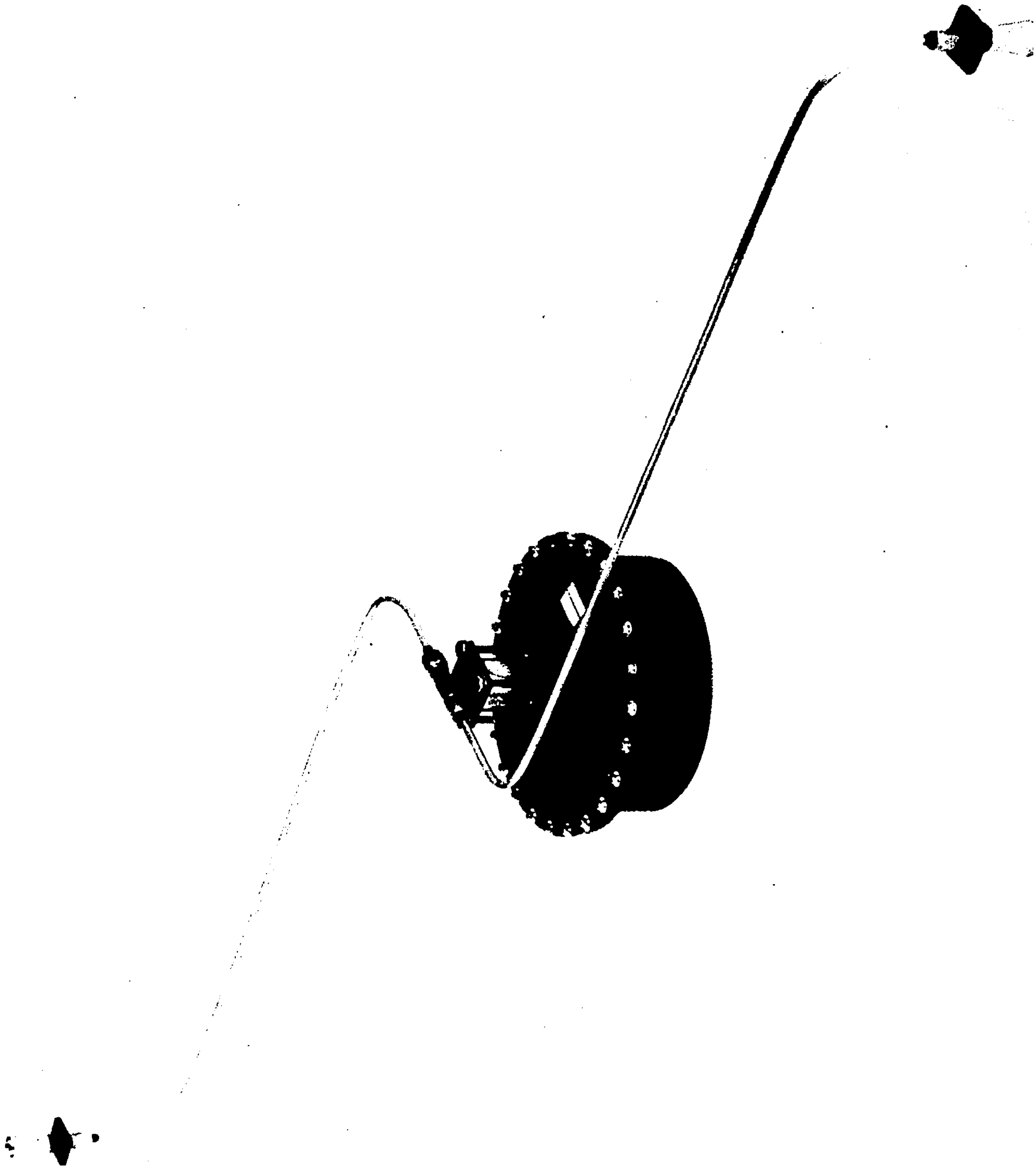


FIGURE 2-21 SUBLIMING SOLID μR-10-2SS-B FOR SPIN AXIS CONTROL

for examination. Whittaker Corporation reported that the valve was completely corroded shut and actually had to be hammered apart. When the valve construction records were checked, it was found that the valve had been constructed of mild steel that was nickel plated with a copper underplate. It was later discovered, during development of the subliming solid, copper or copper based materials should not be used in the system due to the severe corrosive action on these materials by the subliming solid propellant. This rocket was subsequently fitted with a stainless steel valve and returned to NASA.

A third Subliming Solid Control Rocket system (Figure 2-22) was also delivered to Goddard Space Flight Center, NASA, at approximately the same time as the second system discussed. This system was similar to the first, in propellant tank size and amounts and type of propellant carried, except it used a two-way shear seal valve torque motor actuated and four nozzles. This system was first operated at Rocket Research Corporation, kept in storage for approximately 15 months, then operated at NASA. Upon actuation by NASA, the system performed properly and was operated several times over a two week period. Opening of the valve was instantaneous with the first application of power and no valve failure or leakage of any type occurred during this operating cycle.

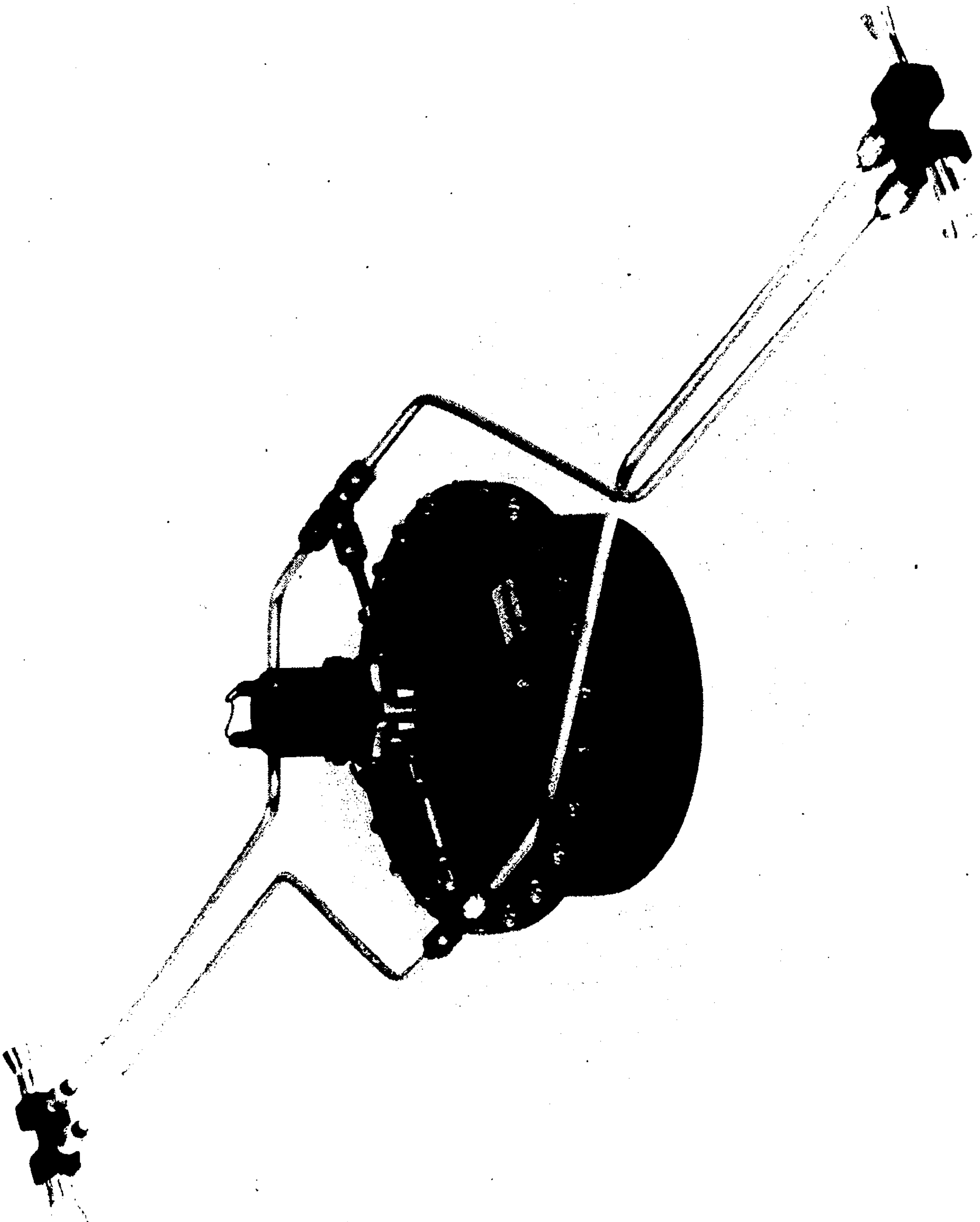


FIGURE 2-22 SUBLIMING SOLID μR-10-2SS-A FOR SPIN CONTROL

3.0 SYSTEM DESIGN

3.1 Introduction

A SSRCS is made up of as many components as required to perform a specific mission. The simplest possible SSRCS is achieved when thrust change may be tolerated during the mission. The SSRCS is composed of:

a. Propellant tank

b. Filter assembly

c. Valves as required

d. Nozzles as required

e. Connecting tubing and fittings

Additional hardware required for certain missions consists of:

a. Small heaters (required to prevent recondensation)

b. Auxiliary heat source (required if relatively high thrust level or duty cycle is anticipated)

c. Thrust control mechanism (required to attain close thrust tolerance)

Thrust is produced whenever the propellant valve is opened. The signal controlling the valve usually originates in the satellite guidance system. Most subliming solid control valves are simple and require a low amount of electrical power for operation (one to ten watts, depending on valve size).

The SSRCS is composed of exceedingly simple and light weight operating parts, with the absence of high pressure making it both a reliable and safe system. The components may be assembled for maximum convenience on any specific spacecraft. If high response is required, valves and nozzles should be close coupled. Care should be exercised to assure that recondensation will not occur in critical areas such as lines, filters, valves, or nozzles.

3.2 Important Design Parameters

There are a number of parameters which must be specified prior to design of any attitude control rocket. These parameters are established, or set, by the requirements of the mission and by the environmental conditions to be encountered. The most important of these parameters are as follows:

a. Total impulse

b. Thrust level

c. Duty cycle

d. Pulse train duration

e. Environmental temperature

f. Mission life

g. Thrust tolerance (if any)

h. Pulse duration

i. Electrical operating power

j. Envelope restrictions

k. Unusual requirements

The total impulse requirement for an attitude control rocket is dependent upon its specific function such as vehicle mass, force or torque to be applied, position in which this force or torque is to be applied, and mission life. Total impulse is usually calculated for a specific satellite and mission, then given to the attitude control designer as a firm requirement. Propellant weight is then calculated on the basis of total impulse.

The thrust level requirement for an attitude control rocket may be dependent upon many things. A low thrust level is generally desirable in order to conserve propellant, however, the thrust must be sufficiently high to permit accomplishment of the mission within the desired time period.

Thrust variation in a subliming system is the result of normal propellant temperature variation in the propellant tank. The thrust level, in certain instances, requires close thrust tolerance to an established value, therefore, requiring a thrust control mechanism. The specific type of control mechanism to be used will depend upon operating and environmental conditions.

There are many cases, however, in which close thrust control is not required. As an example, a thrust control mechanism is not required for an SSRCS used to maintain desired spin rate or to process spin axis of a spinning satellite, thus reducing overall system weight and greatly improving reliability. In cases where a thrust control mechanism is not provided, compensation for thrust variation is accomplished by changing the duration, or number, of pulses used to perform a specific function. The total impulse remains the same, however, the time required for a certain impulse bit to be delivered varies. In many instances, it is possible to tolerate thrust variation of a full magnitude or more without compromising the mission.

The pulse duration, duty cycle, pulse train length, along with thrust level and environmental temperature, establishes the operating temperature of the propellant and, therefore, the minimum propellant vapor pressure. The minimum vapor pressure of the propellant is required to size the valve and nozzle orifices. The pulse duration is defined as the time from command signal on to command signal off; duty cycle as percentage of pulse time on to the total time required to perform a specific maneuver or the entire mission; and pulse train as the series of continuous pulses required to perform a specific maneuver. In general, the duty cycle of a SSRCS, when being used to perform a specific maneuver, is of more importance than the duty cycle for the total mission. For example, a control rocket which is used only once every few weeks may have an extremely low average duty cycle for the mission. However, while the motor is being used or pulsed, its duty cycle could be significantly high. It is these periods of high duty cycle, along with the number of pulses in the pulse train, which are of most concern in determining the propellant temperature.

The temperature of the surroundings will also have a profound effect upon the temperature in the propellant tank. In addition, some knowledge of how the propellant tank is to be mounted in the satellite is also important as it will establish the heat transfer characteristics into the propellant tank. The thrust level, which establishes the propellant flow in pounds per second, is equally important in determining propellant temperature. At this point, it may seem that these parameters are exceedingly critical in the design of a SSRCS. In certain cases this is true; however, in many cases, the conditions which establish propellant temperature are not critical if the thrust level is not particularly critical. It is only necessary to obtain a general feeling for the average operating temperature of the propellant, which will follow closely the satellite operating temperature, in order to design a nominal thrust level. In these cases, a brief look at the operating mode of the engine, in terms of pulse length and duty cycle, will reveal the approximate excursions in thrust level when the engine is operating.

The response of the control rocket is a critical factor in certain limit cycle operations and, in certain instances, spin axis control of rapidly spinning satellites. The SSRCS is entirely dependent on the response of the solenoid valve. Valves are available with fairly high response rates, which allow the delivery of pulses as short as three to ten milliseconds. A high response rate is generally achieved at some small sacrifice in increased operating electrical power for the valve. In addition to specifying the valve type to be used, the high response requirement will also specify the arrangement of the valve with respect to the nozzle. In cases where exceedingly high response is required, the nozzle will have to be extremely close coupled to the propellant valve or actually machined as a part of the outlet to the solenoid valve. In cases where response is not critical and pulse length duration is 100 milliseconds or longer, it is possible to place the nozzle some distance from the valve. This is discussed further in Paragraph 1.7.

The amount of electrical power available to the SSRCS will determine the valve type to be used. See Table 1-1 for valve characteristics. Valves are presently available which require as little as one watt; however, the orifice size of such valves is limited to about 0.030 inch, which, in the case of SUBLEX A propellant, yields a thrust level of 10^{-2} pounds at 70°F. Two watt solenoid valves, with orifice area of 0.050 inch are the most common. For higher thrust levels, a 10 watt solenoid valve with a 0.250 inch orifice area is available.

If available power is limited, special power saving features may be built into the valve. One power saving device is an electric switching circuit which allows full electrical power to be used to pull the valve open, then switches to a lower "holding current" that maintains the valve in open position. This "holding current" is not sufficient, however, to open the valve from the closed position. A second power saving device is a latching valve mechanism requiring only momentary pulses to open and close it. The valve is latched first in the closed position, then in the open position by means of either a Belleville spring (for example, Whittaker P/N 127743) or a form of permanent holding magnet. The use of these devices limits the availability of valves from off-the-shelf stock items; however, they can generally be applied to any specific size valve.

Mission life is important in determining general design philosophy of the control rocket and, in particular, the valve reliability and leakage requirements. If extremely long life is anticipated, the system must be carefully designed to minimize the possibility of mechanical failure during the long unattended operation. This means that a highly reliable valve must be chosen, preferably one with a long successful performance record. In addition, a valve should be chosen with sufficiently low leakage to prevent a large percentage of propellant loss during mission life. Low operating vapor pressure of the engine is a distinct advantage of the SSRCS concept since leakage occurs at a much slower mass flow rate. Leakage, however minute, must be considered for any specific design.

Finally, the satellite environment must be carefully examined to prevent any possibility of recondensation. If propellant lines are to be routed through various parts of the satellite, a definite knowledge of thermal environment of those parts must be available. If any parts of the satellite are colder than the propellant temperature, the lines must be protected against recondensation either by active heating, by conduction from warmer parts of the system, or by pre-choking.

3.3 System Design Procedure

3.3.1 Feasibility Check

The first step in the design of a SSRCS, after system requirements have been defined, is an initial determination of the ability of the rocket to perform the mission. The most important criteria to be checked is the ability to achieve thrust level and duty cycle as specified for the mission. This involves determining the availability and amount of heat or electrical power to cause the necessary sublimation, and the required thrust level and duty cycle. This initial feasibility check can usually be performed rapidly and with experience, almost by intuition. In general, the mission can be performed if the thrust level/duty cycle relationship is 10^{-3} pounds or less and the expected operating temperature is above approximately 0°F. A more thorough analysis of the heat available will be necessary if a higher thrust level/duty cycle is required. The procedure for this analysis is found in Paragraph 3.3.11.

3.3.2 Propellant Choice

Propellant is chosen on the basis of required instantaneous thrust during pulse, expected environmental temperature for the SSRCS, and duty cycle of the rocket. These parameters will define the required propellant flow and expected vapor pressure for the propellant during operation. These two parameters will, in turn, define the minimum restricting orifice in the system. See Paragraph 3.3.6 for minimum orifice size calculation.

In addition to vapor pressure considerations, any subliming rate limitation of the propellant must also be considered if long thrust pulses are anticipated.

Assuming room temperature operation, the general design rule is that SUBLEX A is a good propellant for thrust levels down to 5×10^{-4} pounds, and SUBLEX B is a good propellant from 5×10^{-4} down to 10^{-5}.

3.3.3 Propellant Form

The next design step is to choose proper propellant form. Granular form of propellant is desirable since this allows maximum sublimation surface area and easiest loading and handling. Where extreme volume limitation is encountered, either the recondensed crystalline form or the pressed powder form may be used. However, care should be taken to avoid subliming rate limitation, particularly in the recondensed crystalline form.

3.3.4 Propellant Weight Calculation

Once a propellant choice has been made, the required propellant weight may be calculated. Propellant weight calculation requires knowledge of the total impulse, thrust level, propellant to be used, and propellant loss due to leakage. In order to account for propellant loss due to leakage, the expected operating vapor pressure, valve leakage, and mission life must be known.

The specific impulse of a low thrust rocket is dependent upon its operating thrust level. Table 2-5 lists various specific impulses and thrust levels obtained during specific impulse testing. This table was obtained as a result of limited experimental data and requires additional research to extend the data to a wider thrust range. However, until better data is obtained, this information can be used as a relatively close approximation. Since the two major candidate subliming solid propellants have relatively similar molecular weights, their specific impulse will be nearly the same at any given thrust level. The theoretical specific impulse at room temperature is 85 seconds. Based on experimental data at a thrust level of 10^{-2} pounds, 75 seconds can be assumed to be delivered. Also, based on experimental data of 10^{-4} pounds of thrust, 60 to 65 seconds can be delivered. The specific impulse will also vary with the square root of absolute temperature as shown by the equation in Paragraph 1.2.1.2. The change of specific impulse due to temperature is negligible and can usually be neglected since most SSRCS's operate between 0°F and room temperature.

The required propellant weight is obtained by dividing the specific impulse into the total impulse. It is usually good design practice to increase this amount by ten percent as a mission safety margin. In addition, if the mission is to be of relatively long duration it is necessary to include the propellant loss due to leakage. An average valve leakage rate, $\dot{V}_L$, is assumed for the mission, and the operating temperature of the rocket is estimated in order to determine propellant vapor pressure P_1. The mass leakage rate, $\dot{W}_L$, is then calculated from the following equation:

$$\dot{W}_L = \frac{144\, P_1}{\frac{R'}{M} T_1} \quad (9.8 \times 10^{-9}) \ \dot{V}_L$$

$$\dot{W}_L = \frac{1.41 \times 10^{-6}\, P_1\, \dot{V}_L}{\frac{R'}{M}\, T_1}$$

Where:

$\dot{W}_L$	=	mass leakage rate, lb/sec
P_1	=	propellant vapor pressure, psia
M	=	vapor molecular weight
T_1	=	vapor temperature, °R
$\dot{V}_L$	=	average valve leakage rate, cc/hr
R'	=	universal gas constant 1544 ft-lb/mole °R

This mass leakage rate is then muitiplied by mission life in seconds to determine total propellant mass loss. Valve leakage rates of from 0.10 cc to 10 cc per hour are common.

3.3.5 Propellant Tank Size and Configuration

The propellant tank volume can be calculated following determination of propellant weight since it is dependent upon weight and density for the propellant form chosen. As previously indicated, it is generally best to choose a loose granular form of propellant. The propellant density for this form is approximately 0.027 pounds per cubic inch for SUBLEX A and approximately 0.03 pounds per cubic inch for SUBLEX B. The pressed powder may be as high as 0.04 pounds per cubic inch and a recondensed cake as high as 0.05 pounds per cubic inch. The propellant tank volume is obtained by dividing the density into the propellant weight to be carried and adding ten percent to this figure to account for inaccessible volume. The volume of the filter holder, which may project into the tank, is also added.

The tank configuration is then chosen. A spherical configuration is generally best for minimum wall thickness tanks. However, due to its low vapor pressure, the SSRCS can be packaged in a variety of different shapes using standard structural equations as design criteria. Aluminum is the most desirable tank material due to its light weight and thermal conductivity.

There are some cases where the standard pressure vessel equations will yield wall thicknesses below a gauge of 0.020 inch for aluminum. In these cases, a minimum gauge of 0.020 inch is usually chosen to simplify handling problems. A thicker aluminum gauge also increases heat transfer around the propellant tank. The propellant tank may be designed for maximum heat transfer, maximum ease of mounting, or minimum volume. Standard design practice for welds and mounting brackets should be applied.

The propellant tank weight, after its size and configuration have been determined, is calculated as follows:

a. Calculate actual tank metal volume. Increase this amount by ten percent to account for welds.

b. Multiply this figure by the metal density. (Aluminum density is 0.1 pound per cubic inch.) Be sure to include metal volume of any flanges and outlet ports.

e. The figure obtained is the required tank weight.

3.3.6 Restricting Orifice Calculation

Flow rate and, therefore, thrust level of a SSRCS will depend upon operating vapor pressure of the rocket and size of the restricting or choking orifice in the system. The minimum orifice may be in the valve, exhaust nozzle, or some built-in orifice, depending on the specific design approach chosen. Very often the valve orifice is chosen as the restricting orifice since this allows maximum flow through the valve under any given pressure condition. When flow rate is not a particular problem, the nozzle may be used as the restricting orifice, or as will be discussed further in the recondensation section, a restricting orifice may be built into the system to drop the pressure and, therefore, minimize the possibility of recondensation in lines prior to entering the exhaust nozzle.

The size of the restricting orifice is dependent not only upon the thrust level, but also upon the degree of thrust tolerance to be assumed. There are three cases which can be considered:

a. A system designed with no thrust control. The thrust is allowed to vary from a nominal design level in any manner due to temperature changes within the propellant tank.

b. A system design in which minimum thrust level should not be exceeded.

c. A unit in which close thrust tolerance is required. This thrust tolerance may be achieved, for example, by either a thermostatic system, a pressure regulator, or a thermally controlled variable orifice.

The above cases are discussed in Paragraphs 3.3.6.1 through 3.3.6.3.

If the restricting orifice is to be the valve, valve orifice diameter must be calculated in order to determine valve type and size required. The valve orifice diameter (assuming choked flow at valve orifice) depends on thrust level F, specific impulse I_{sp}, and vapor pressure expected at the valve orifice P_1. The valve orifice pressure is, in turn, dependent upon propellant operating pressure and pressure drop occurring before vapor reaches the valve orifice. The allowable pressure drop, $(P_1 - P_2)$, through the valve orifice must be determined. Using the preceding information, the valve orifice area must be determined. Using the preceding information, the valve orifice area A_o, is determined by the following equation:

$$A_o = \left(\frac{F}{I_{sp} P_1}\right) \left(\frac{R' T_1}{2_{gM}}\right)^{1/2} \left[\frac{\frac{K - 1}{K}}{\left(\frac{P_2}{P_1}\right)^{2/K} - \left(\frac{P_2}{P_1}\right)^{K + 1/K}}\right]^{1/2}$$

Where:

A_o = orifice area, square inches

F = thrust, pounds

I_{sp} = specific impulse, lbf/lbm-sec

P_1 = feed pressure, psia

R' = universal gas constant, 1544

M = molecular weight of vapor

T_1 = absolute temperature, °R

K = specific heat ratio of vapor

P_2 = down-stream pressure, psia

(NOTE: If P_2/P_1 is less than .546, use .546 since orifice will be choked.)

In addition to using the equation above, it is also possible to make a more rapid estimate of the minimum possible valve orifice area, A_o, assuming choked flow, by using the following approximate equation:

$$A_o = \frac{F}{C_F P_o}$$

Where:

A_o = orifice area, square inches

F = desired thrust, pounds

C_F = exhaust nozzle thrust coefficient, (usually between 1.1 and 1.8)

P_o = pressure upstream of the orifice, psia

(The equation is the same as the one used to calculate the exhaust nozzle orifice to produce any given thrust level. If it is used to establish the valve orifice size, a new pressure, reduced by a factor of at least two, must be used to calculate the nozzle orifice.)

Since most valve entrance areas are sharp-edged orifices, it is necessary to multiply orifice area obtained in any of the above calculations by the inverse of 0.65, or diameter by square root of the inverse of 0.65. This is the discharge coefficient for a sharp edged orifice.

In general, orifice diameter should be between 0.010 and 0.050 inch since plugging occurs easily below 0.010 inch and valves with orifice diameters above 0.050 inch are not readily available and are usually undesirably large.

3.3.6.1 No Thrust Control

The calculation of a restricting orifice diameter is simplest if no thrust control is required. In this case, nominal operating temperature of the rocket is determined; vapor pressure which corresponds to this temperature is then obtained from the vapor pressure versus temperature graph shown on Figure 1-2; and restricting orifice diameter calculated on the basis of that pressure. A pressure drop prior to entering the restricting orifice may or may not be included in this calculation. The expected thrust excursions from nominal may be obtained by determining heat balance and expected temperature excursions for the propellant. Thrust will vary directly with any expected or calculated change in vapor pressure due to temperature changes.

3.3.6.2 Minimum Thrust Level

The determination of orifice size for minimum thrust level is more difficult since it involves a rather detailed knowledge of expected thermal conditions within the SSRCS. Not only must expected temperature excursions of the vehicle be noted, but some estimate must be made of the temperature drop of propellant itself during the on cycle in order to obtain minimum temperature and, therefore, minimum vapor pressure to be expected during operation. The general procedure for performing such a calculation is as follows: The minimum temperature to be expected in the satellite vehicle is first obtained. The desired minimum thrust level is then calculated, or established, and the heat requirement to cause required propellant flow at that thrust is calculated. The mode of heat transfer to be expected is then established and temperature drop required to transfer the necessary amount of heat into the propellant tank is determined from Figures 2-7 through 2-12. Minimum propellant temperature is obtained by subtracting the differential temperature required from the minimum expected environmental temperature. Vapor pressure at this minimum temperature is then obtained and the orifice sized accordingly, using the equation found in Paragraph 3.3.6.

3.3.6.3 Thrust Control Required

The third case, in which thrust control is required, will depend upon the type of thrust control utilized. If a thermostat system is to be used, propellant temperature will remain or be maintained at a constant level for the duration of the mission. In this case, vapor pressure will also remain constant and the calculation of the restricting orifice size is simple. The equation in Paragraph 3.3.6 is used for this calculation.

If either a thermally controlled variable orifice or a pressure regulator is to be used, then the restricting orifice is in either of these devices. In this case, the maximum orifice must be designed for minimum temperature. This calculation will follow the same rules established for minimum thrust limitation. Once again, a knowledge of thermal environment and heat transfer characteristics will be required. The same procedure is followed using the equation in Paragraph 3.3.6.

3.3.7 Valve Sizing and Configuration

In certain instances it may be desirable, or necessary due to valve size restrictions, to use the valve orifice as the restricting orifice in the system. In this case, the valve orifice size which was calculated in Paragraph 3.3.6 is used as the minimum orifice diameter and is specified to the valve manufacturer. In other cases, it may not be necessary or possible to use the valve orifice as the restricting orifice for the system. This is particularly true if a thrust control device, such as a thermally controlled variable orifice or pressure regulator, is used. In these cases, it is necessary to use a valve orifice diameter which is from 1-1/2 to 2 times the minimum restriction orifice which was calculated in Paragraphs 3.3.6 through 3.3.6.3.

Once the valve orifice has been determined, the valve type and configuration can be chosen. The rules discussed in Paragraph 3.3.6 are applicable. Usually, the valve type and configuration are chosen for maximum reliability, and the materials should be carefully chosen to insure they are compatible with the propellant to be used. An all welded construction should be used in flight valves.

3.3.8 Nozzle Size

In some SSRCS's, the exhaust nozzle will be the only restricting orifice in the system. In this case, the nozzle throat diameter is obtained using the equations found in Paragraph 3.3.6. Care should be taken to insure that pressure drops through the flow system, prior to reaching the exhaust nozzle, are taken into consideration.

In systems which employ a thermally controlled variable orifice or pressure regulator, the nozzle should be sized with a throat area at least 2-1/2 times larger than the area of the restricting orifice in either of these devices. This is necessary to insure that required choking will occur at the orifice in the pressure regulator or thermally controlled variable orifice. By establishing the nozzle throat area at least 2-1/2 times larger than the other restricting orifice area, choked flow upstream can usually be assured. If excessive line pressure drop is anticipated, it may be necessary to size the nozzle throat area greater than 2-1/2 times the area of the thermally controlled variable orifice or pressure regulator. The reason this area ratio of 2-1/2 is specified is that pressure drop across the choked orifice for the subliming solid vapor is approximately a factor of 2. Since the exhaust nozzle will also choke, it is necessary that its orifice area be large enough to accommodate the lower pressure.

The following procedure should be used to size the nozzle throat if pre-choking is used to prevent recondensation in the propellant lines of the subliming solid system:

a. Determine minimum temperature to be encountered in any section of the propellant lines.

b. Determine maximum anticipated temperature in the propellant tank.

c. Obtain propellant vapor pressure for these temperature conditions. See Figure 1-2 for a graph giving vapor pressure versus temperatures.

Normally, a restricting orifice is used at the propellant tank outlet or just downstream of the propellant valve. This restricting orifice should be sized to provide the desired system flow. The exhaust nozzle throat area is determined by taking the ratio of the propellant tank pressure to the ratio of the line pressure corresponding to the recondensation point and multiplying this figure by the restricting orifice area. As a safety factor, the exhaust nozzle throat should be increased slightly above this figure.

The area ratio of the exhaust nozzle and the divergence angle will depend on the thrust level of the rocket. As discussed in Paragraph 3.3.6, at very low thrust levels it is often desirable to use a relatively low exit to throat area ratio. Further experimental work is necessary to define the optimum area ratio for any given thrust condition. The design rules found in Paragraph 3.3.6 should be applied in the design or sizing of the nozzle for the SSRCS.

3.3.9 Line Size

In general, any propellant line which leads to a choking orifice any place in the system (valve, exhaust nozzle, or any other restricting orifice) should have a cross-sectional flow area considerably greater than twice that of the throat or orifice of the restricting orifice. A pressure drop calculation should be made if the propellant flow line is very long and/or the line cross-sectional area is close to just twice the throat area. Consult texts on viscous flow for appropriate tables, graphs, and equations for this calculation. For propellant lines of extended length, the allowable pressure drop is first determined and then the line diameter calculated.

The propellant line material and wall thickness is generally determined by availability of stock. Since stress on the line is quite low, due to the small diameter and low operating pressure, wall thickness is usually not a critical parameter.

3.3.10 Recondensation

Recondensation is a secondary effect on the size of the SSRCS. If pre-choking is to be used to prevent recondensation in a propellant line, as discussed in Paragraphs 1.10.5 and 2.10.4.4, then the propellant line and exhaust nozzle must be designed as discussed in Paragraph 3.3.8 on nozzle sizing.

The following equation should be used to calculate the amount of heat required to prevent recondensation by means of active electrical heaters, or radioisotope capsules:

$$P = S\,\sigma(e_1 T_1^4 - e_2 T_2^4)$$

Where:

P	=	required heater power in watts
S	=	outside surface area of the propellant line in square meters
σ	=	5.67×10^{-8} watts/M^2 - °K^4
e_1	=	propellant line emissivity
T_1	=	desired propellant line temperature in °K
e_2	=	emissivity of surroundings
T_2	=	ambient temperature in °K

To use this equation, first determine the maximum expected temperature in the propellant tank. Next, determine the minimum temperature of the environment. Based on these two parameters, calculate the amount of heat required to maintain line temperature slightly above, or equal to, the propellant tank temperature. The equation assumes no insulation. If insulation is used, consult a test on heat transfer for the complete equation.

3.3.11 Heat Transfer

Heat transfer requirements greatly affect SSRCS design and size. This is because the internal propellant tank configuration and the tank mounting brackets depend on the amount of heat transfer required for proper operation in any given application of the

SSRCS. See Figures 2-6 through 2-12 for heat transfer characteristics. At thrust levels below 10^{-3} pounds, thermal contact area is not normally critical. As a result, the thermal mounting area is designed from a convenience standpoint rather than as a result of heat transfer requirements.

In cases where the thrust level/duty cycle relationship is above 1×10^{-3} lbf, the tank surface contact area should be maximized. It is generally necessary to experimentally verify that a high thrust level/duty cycle relationship can be maintained for any given tank mounting area and environmental temperature condition.

3.3.12 Sample Calculation

In order to demonstrate the procedure by which the design of a SSRCS is pursued, the following sample calculation is presented. The hypothetical conditions for the SSRCS are as follows:

Total Impulse	100 lb-sec
Thrust	10^{-2} lb nominal total 10^{-3} lb minimum total
Temperature	40°F to 80°F
Impulse Bit	0.1 lb-sec
Duty Cycle	1% maximum
Number of Nozzles	4
Number of Valves	2
Line Length	2 feet
Life	1 year

3.3.12.1 Propellant Weight

Specific Impulse, I_{sp} = 75 lbf-sec/lbm at $F = 1 \times 10^{-3}$ lbf

Total Impulse, I_T = 100 lb-sec

Nominal Propellant Weight, W_I = $\dfrac{100}{75} = 1.33$ lbm

Leakage = 10 cc/hr total for two valves

Average Pressure = 6 psia at 60°F

Vapor Density = $0.03\ lb/ft^3 = 1.05 \times 10^{-6}\ lb/cc$

Leakage Rate = $1.05 \times 10^{-5}\ lb/hr$

Time = 1 year = 8.75×10^3 hours

Propellant Leakage, W_L = 0.09 lb

Total Propellant Weight = 1.33 + 0.09 = 1.42 lb

3.3.12.2 Valve Orifice Size

Using nominal specifications and assuming choked flow at the valve:

$$\text{Valve Orifice Area} = A_o = \frac{F}{P\, C_F\, C_D}$$

Where:

F = nominal thrust = 1×10^{-2} lbf

P = nominal pressure = 5 psia (from vapor pressure curve at 60°F)

C_F = thrust coefficient = 1.8

C_D = discharge coefficient = 1.0

$$\therefore A_o = \frac{1.0 \times 10^{-2}}{(5)\,(1.8)\,(1.0)} = 1.11 \times 10^{-3}\ in^2$$

and Valve Diameter

$$D_v = \left(\frac{4}{\pi} A_o\right)^{1/2}$$

D_v = .0376 in with $C_D = 1.0$

Valve diameter is small enough to use conventional coaxial or sheer seal valves.

3.3.12.3 Nozzle Size

Pre-choke to prevent recondensation: Valve located on tank.

Tank Maximum Temperature = 80°F

Minimum Line Temperature = 40°F

Maximum Pressure = 9.6 psia at 80°F

Minimum Pressure = 2.55 psia at 40°F

Nozzle Throat Area $= A_o \dfrac{P_{max}}{P_{min}} = 1.11 \times 10^{-3} \dfrac{(9.6)}{(2.55)}$

$= 4.18 \times 10^{-3}$ in^2

Throat Diameter $= \left[\dfrac{4}{\pi} A_N\right]^{1/2}$

Throat Diameter = .073 in

Area Ratio = 50:1

Nozzle Exit Area $= 50\, A_N = 2.09 \times 10^{-1}$ in^2

Nozzle Exit Diameter = 0.516 in

3.3.12.4 Tank Size

Total Propellant Weight = 1.42 lb

Use granular form of propellant

Propellant Density = 0.027 lb/in^3

Propellant Volume = 53 in^3

Filter Volume = 3 in^3

Tank Volume = (1.1) 53 + 3 = 61 in^3

Spherical Tank

Tank Diameter $= \left(\dfrac{6}{\pi} V_T\right)^{1/3}$

Tank Diameter $= (117)^{1/3} = 4.9$ in

Wall Thickness $= \dfrac{PD}{2S}$

Use allowable stress, S, of 20,000 psi for aluminum

Burst Pressure, P = 4 times maximum working pressure

= 4 times atmospheric pressure

Burst Pressure = 60 psia

Wall Thickness $= \dfrac{(60)\,(4.9)}{(2)\,(20{,}000)} = \dfrac{294}{40{,}000}$

Wall Thickness $= 7.35 \times 10^{-3}$ in

Use minimum gauge of 20×10^{-3} in

Metal Surface Area	=	πD^2_{Tank}
Metal Surface Area	=	75
Tank Wall Weight	=	$A_{Tank}\ th\ \rho_{Alum.} = (75)\ (0.02)\ (0.1)$
Tank Wall Weight	=	.15 lb

Use 20% for tank welds

Outlet Port Weight	=	0.05 lb
Tank Weight	=	(1.2) (.15) + .05 - .23 lb

3.3.12.5 <u>Component Weight</u>

Filter	=	0.15 lb typical
Valve	=	0.15 lb typical
Nozzle	=	0.05 lb typical
Lines	=	0.02 lb/ft 1/4 inch line
Fittings	=	0.01 each 1/4 inch line
Mounting Brackets	=	.02 each

3.3.12.6 <u>Weight Summary</u>

Tank (1)	=	0.23 lb
Valves (2)	=	0.30
Nozzles (4)	=	0.20
Filter (1)	=	0.15
Lines (8 ft)	=	0.16
Fittings (8)	=	0.08
Mounting Brackets (4)	=	0.08
Total Inert Weight	=	1.20 lbm
Propellant	=	1.42
Total Loaded Weight	=	2.62 lbm

3.4 Program Conclusions and Recommendations

Phase I of the subliming solid development program demonstrated the capability of various SSRCS components to perform their respective functions. It was also demonstrated that these various components can be assembled into a package which is capable of prolonged and reliable operation. The assembly of a successful flight package and resulting qualification program is being performed during Phase II of the SSRCS program. There are various component tests which can be, and should be, continued. These fall primarily in the category of:

a. Continued valve investigation.

b. Further development of thrust control mechanisms.

c. Further development of recondensation prevention, particularly the use of radioisotope capsules.

d. Further investigation on the internal propellant tank design to attain maximum heat transfer.

It is felt that continued work in these areas will help to broaden the possible application of the SSRCS and will assist in intelligent design and improve the reliability of the SSRCS.

2.0 CRITERIA FOR EVALUATING RADIOISOTOPES

The following criteria were established for evaluating various isotopes under consideration.

Power Density	Sufficient to produce 1/30th and 1/10th thermal watt per capsule while maintaining light weight and compactness.
Type and Energy of Radiation	No gamma rays over 0.1 Mev No beta rays greater than 1. Mev Not over 100 neutrons/cm^{2}sec/curie Alpha particles are not limited
Safety Considerations	Produce no hazard to 1) personnel during fabrication, testing, or launch and 2) produce no adverse effects upon other system components.
Half-life	Greater than 1 year.
Availability	Readily available in sufficient quantities for missions and produced with less than 6 months lead time.
Temperature Characteristics	Melting temperature such that upon burnup, the source is either entirely consumed or remains entirely intact.
Cost	Less than $1000 per thermal watt, although this is of less consideration, within reasonable limitations, than the safety requirements.

3.0 EVALUATION OF POTENTIAL RADIOISOTOPES

The radioisotopes, which approximate the criteria previously discussed, are listed on Table I-1 along with some of their properties. Many other radioisotopes are available, but since they did not meet any of the established criteria they were not included in the study. The most promising radioisotopes are discussed in the following paragraphs.

3.1 Cerium 144 (Ce 144)

Cerium 144 has a half-life of 285 days, a high specific power of 21.5 watts/gram, and a reasonably high fission yield of six percent from Uranium 235. The daughter product, Praseodymium 144, emits an energetic gamma ray which will require heavy shielding. The power density of the oxide is theoretically about 21 watts/cc making this source potentially the least expensive on a cost per watt basis. However, the shielding requirement of the gamma and the short half-life make this source unacceptable.

3.2 Cesium 137 (Cs 137)

Biologically, Cesium 137 is far less hazardous than other candidates and the recovery costs are not excessive. Cesium is recovered from reactor fuel processing waste materials. This includes Cs 134 with a half-life of 2.3 years; Cs 133 which is stable; and Cs 135, with a half-life of 2 million years, which, for practical purposes, may be considered stable. Cesium 137 is a beta emitter which could be shielded. However, its daughter product, Barium 137 with a half-life of 2.6 minutes, emits a rather energetic gamma ray. The specific power from Cesium 137 is 0.42 watts per gram and when used in the form of a glass, the power density is about 0.22 watts per cc.

3.3 Curium 242 (Cm 242)

Curium 242 has a half-life of 163 days, and a specific power of about 120 watts per gram. Since it is a pure alpha emitter, no shielding is required. The power density of the oxide is about 1170 watts per cc.

As with other alpha emitters, spontaneous fission at a low rate does occur and helium gas is a final product. The cost of producing Cm 242 is high. The availability and short half-life also reduce the desirability of Curium 242.

TABLE I-1

CHARACTERISTICS OF POTENTIAL ISOTOPES

Isotope	Half-Life Actual	Half-Life Effective	Power Density (w/cc)	Form	Melting Point (°C) (Compound)	Energy (Mev)	Cost ($/Thermal Watt)	Availability	Shielding, Safety Remarks
Ce 144	285 dy.	255 dy.	17.0	CeO_2	1950	3.0 β max. 2.2 γ max.	76	Abundant	Too much γ - not suitable.
Cs 137	33 yr.	> 10 yr.	0.22	Glass	646	β 0.51 (92%) 1.17 (8%)			
						γ 0.662 (92%)	63	Fair. Fission product	Require γ shielding
Co 60	5.3 yr.	~ 4 yr.	15.5	Metal	1495	β 0.306 γ 1.17 1.33	33	Available	Require heavy shielding
Cm 242	163 dy.	~130 dy.	1170.0	CmO_2	1500	α 6.11 (74%) 6.07 (26%) γ 0.044 (26%)	170	Limited	Minimal except for neutrons by spont. fission.
Cm 244	17.6 yr.	> 10 yr.	27.0	CmO_2	1500	α 5.8 (77%) 5.76 (23%) γ 0.043 (23%)	1600	Potentially available.	Minor except for neutrons by spont. fission.
Po 210	138 dy.	~110 dy.	1320.0	PoO_2	900	α 5.31 (100%)	640	Very limited in large quantities	Essentially no shielding required.
Pm 147	2.5 yr.	~ 2 yr.	2.22	Pm_2O_3	1000	β 0.223	168	Fairly Available Fission Product	Very little required if source aged prior to use.

TABLE I-1 (Cont'd)

CHARACTERISTICS OF POTENTIAL ISOTOPES

Isotope	Half-Life Actual	Half-Life Effective	Power Density (w/cc)	Form	Melting Point (°C) (Compound)	Energy (Mev)	Cost ($/Thermal Watt)	Availability	Shielding, Safety Remarks
Pu 238	84 yr.	> 10 yr.	9.3	PuO_2 PuC	2000	α 5.5 (72%) 5.45 (28%) γ 0.044 (28%)	1600	Limited	Minor Shielding
Sr 90	28 yr.	> 10 yr.	1.4	$SrTiO_3$ SrO	2430	β 0.610 2.180	45	Fission Product	Heavy shielding requirements. Biol. hazardous
Th 228	1.9 yr.	~ 1 yr.	1270.	ThO_2	3050	α 5.42 (71%) 5.34 (28%) γ 0.034 (28%)	6600	Potentially available	Heavy shielding plus cost
Tm 170	127 dy.	~120 dy.	13.	Oxide Metal	1525 23 - 2600	β 0.967 (78%) 0.883 (22%) γ 0.0842 (22%)	40	Abundant	Cold encapsulation short T1/a. Low shielding requirements
U 232	74 yr.	> 10 yr.	33.	UO_2		α 5.32 (68%) 5.26 (32%) γ 0.06 (32%)	230	Potentially available	Heavy shielding required

3.4 Curium 244 (Cm 244)

Curium 244 offers a good half-life of 17.6 years is an alpha emitter, requiring very little shielding; and has a specific power of 2.8 watts per gram. The power density of the oxide is about 27 watts per cc.

A disadvantage of Curium 244 is that it is produced through successive neutron irradiations from Plutonium, with decreasing efficiency and increasing difficulty. The method of overcoming this is to perform lengthy irradiations of the starting material (Plutonium) and allowing the concentration of Curium 244 to build up, which is the case in reactors using Plutonium fuel. However, the time requirements have not yet been met. Therefore, the availability of this source is more potential than immediate. Costs predicted are expected to be less than $4,300 per gram or approximately $1,600 per watt in large quantities. Also, the release of neutrons through spontaneous fission makes the use of this source less desirable due to the required shielding and the hazards involved.

3.5 Polonium 210 (Po 210)

Polonium 210, a rare element, is obtained in the decay chain of Uranium 238 and also through the irradiation of bismuth. It has a high specific power of 140 watts per gram. However, the half-life is only 138 days. Because it is an alpha emitter, virtually no shielding is required and in comparison with the other alpha emitters, it has a low biological hazard.

The use of Po 210 was demonstrated with the SNAP III prototype generator. The cost of this isotope is calculated to be about $640 per watt in large quantities. The short half-life is the major drawback to this isotope.

3.6 Promethium 147 (Pm 147)

Promethium 147 is produced in the fission process with about one-half the yield of either Strontium 90 or Cesium 137. Pm 147 emits a low energy beta particle in decaying to Samarium 147. Samarium 147 is considered stable due to its long half-life of 10^{11} years. The low energy beta is ideal, since no bremsstrahlung or secondary radiation of consequence is produced.

During the production of Promethium 147, some Promethium 148 is also produced. Pm 148 has a half-life of 42 days and emits an energetic gamma ray which requires rather extensive shielding. By aging the source for about one half-life of Pm 147, (about 2.5 years) radiations from the Pm 148 would be virtually eliminated, which will allow minimal shielding to be used.

The specific power of Promethium 147 is 0.41 watts per gram with the power density of the oxide being about 2.2 watts per cc. Biologically, Pm 147 is among the least hazardous of those being considered.

3.7 Plutonium 238 (Pu 238)

Plutonium 238 has a half-life of 89.6 years. While its specific power is only 0.56 watts per gram, the power density of the metal form is 3.5 watts per cc. Its gamma radiation permits use without extra shielding requirements.

Being primarily an alpha emitter, the design of the Pu 238 source must allow for the accumulation of helium gas as an emission product. Therefore, the source must provide either void space or a means of venting this gas. Spontaneous fission may also be expected with this source. However, the half-life of this reaction is quite long, resulting in very low levels of neutron flux.

Disadvantages of Pu 238 are its limited availability, low power density, high biological hazard*; and cost of about $1,600 per thermal watt. In all other aspects it is an excellent source.

3.8 Strontium 90 (Sr 90)

Strontium 90, with a half-life of 28 years, exists in reactor fuel processing wastes along with the stable isotopes Sr 86 and Sr 88 and the radioactive isotope Sr 89 with a half-life of 50 days. By aging the source for about 6 months, the contribution due to the Sr 89 is virtually eliminated. Sr 90 emits a very energetic beta particle. In addition, the daughter product, Y 90 emits on energetic gamma ray.

These two radiations require excessive shielding, not only for the primary radiations, but also for the secondary or bremsstrahlung radiation.

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*Biological hazard is directly related to half-life. In case of spill (i.e., breakage of the capsule) the radioactive material would present a hazard for a period of several half-lifes. The longer the half-life, the higher the hazard.

The specific power of Sr 90 is 0.95 watts per gram. The power density of Strontium titanate is approximately 1.2 watts per cubic centimeter. In addition to this high power density, the titanate is the ideal chemical form since it is extremely stable chemically and exhibits an almost ideal resistance to both fresh and salt water. The main disadvantage is prohibitively high shielding requirements.

3.9 Thorium 228 (Th 228)

Thorium 228 has an extremely high power density of 1,270 watts per cc in the oxide form, and an almost ideal half-life of 1.9 years. Its cost, however, is quite high ($6,600 per thermal watt), and it is not readily available as an off-the-shelf item. Thorium 228 is produced by separating Thorium from the Uranium parent. While adequate supplies of the parent are available, the development of the separation techniques is still quite far from complete. Also, this source requires shielding due to the gamma radiation. Thus, it does not appear as favorable as some of the other sources discussed.

3.10 Thulium 170 (TM 170)

Thulium 170 is economically feasible and readily available through the cold evaporation and irradiation of Thulium 169. Thulium 169 is the only stable isotope of Thulium. A disadvantage of Tm 170 is its relatively short half-life of 129 days which results in a useful power half-life of about 120 days. The oxide, Tm_2O_3, has extremely high temperature properties. Thulium 170 offers two energetic beta emissions without any strong gamma emission which minimizes shielding requirements. It is possible, by mathematical calculation to predict a heat source of about 1.75 watts per gram of Tm_2O_3.

3.11 Uranium 232 (U 232)

Uranium 232 is characterized by a high specific power of 4.4 watts per gram, and in the dioxide form, a high power density of 33.0 watts per cc. This is the result of a decay chain consisting of many daughters which emit energetic alpha practices. Some gamma radiation is also produced which requires shielding. The first daughter product is Thorium 228, which, because of its long half-life, does not reach equilibrium, and therefore its maximum heat output, until eight years after initial production of U 232. However, an aged source of U 232 may be used which is richer in Th 228 to start the process, thus enabling the source to produce maximum power immediately without the eight year delay at partial power. This maximum state may be maintained for over thirty years, offering a distance advantage of uniform and stable heat supply over long periods of time.

Biologically, U 232 is less hazardous than Pu 238, but more hazardous than some of the other candidates. Very little work has been done on the recovery of this isotope, although it is potentially available. Although U 232 has an attractive half-life, it does not appear immediately promising due to its limited availability and shielding requirements.

4.0 CONCLUSIONS

After a careful review of the sources discussed and their pertinent characteristics as shown on Table I-1, only five candidates are considered as approaching the ideal criteria previously established. These five are: Curium 242 (Cm 242), Curium 244 (Cm 244), Polonium 210 (Po 210), Promethium 147 (Pm 147) and Plutonium 238 (Pu 238). Additional data on these five sources, including radiation from each, is shown on Table I-2.

The use and development of these heat sources employing radioactive material will require compliance with the regulations of the U. S. Atomic Energy Commission (AEC). These regulations apply even though the sources being considered for this usage are quite small. The expected radiation levels from these sources will be well below the human tolerance limits unless the capsules are broken open or otherwise damaged.

Three private companies, General Electric, Battel Institute, and Isoserve Corporation, are in a position to supply radioisotopic heat sources with the desired properties.

The use of radioisotopic generators for space applications is well founded and promising. Several thermoelectric sources are currently being used for remote applications and space missions. From this study it is evident that the use of radioisotopic heat generators will greatly aid the use of the subliming solid control system.

TABLE I-2

EXPANDED CHARACTERISTICS OF ACCEPTABLE ISOTOPES

Isotope (& Half-Life)	Power Density (w/cc)	Size of Source		100 W Th., Unshielded, 1 yd. (mr/hr)	Shielding (100 Th W Source)	Remarks and Daughter Products
		0.1 Th W	100 Th W			
Cm 242 (163 dy)	1170	< 1.0 cc	1.0 cc	γ 4.3 n 32.5 Total 36.8 mr/hr	1/4 inch lucite & 1/4 inch Pb	89 yr. Pu 238 then 10^5 yr. U 234 163 days very short
Cm 244 (17.6 yr)	27	3.7×10^{-2} cc (33 Curies)	3.7 cc (33 KC)	$\gamma \sim 0.064$ n ~ 0.113 Total ~ 0.178 mr/hr	1/4" of low atomic number material, as carbon, boron, etc. encapsulation	10^3 yr. Pu 240 minimum shielding, good half life Difficult to obtain
Po 210 (138 dy)	1520	< 1.0 cc (3.12 Curies)	1.0 cc (3.12 KC)	γ 20 n 6.5 Total 26.5 mr/hr	1/4" encapsulation with low atomic number material	Stable Po 206. Half life short
Pm 147 (2.5 yr)	2.22	< 1.0 cc (12 Curies)	46 cc (12 KC)	γ 0.0115 mr/hr	1/8" encapsulation with low atomic number material and 1/4" Pb	10^{11} yr Sm 147 Best candidate
Pu 238 (89 yr)	3.5	< 1.0 cc (3 Curies)	10.8 cc (3 KC)	γ 0.01 n 0.78 Total 0.79 mr/hr	1/8" encapsulation with low atomic number material	10^5 yr U 234 Half life may be too long