NASA CONTRACTOR REPORT

NASA CR-712

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DEVELOPMENT OF THE SUBLIMING SOLID CONTROL ROCKET

PHASE II

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

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • MARCH 1967

NASA CR-712

DEVELOPMENT OF THE SUBLIMING SOLID CONTROL ROCKET

PHASE II

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Prepared under Contract No. NAS 5-9070 by
ROCKET RESEARCH CORPORATION
Seattle, Wash.

for Goddard Space Flight Center

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

ABSTRACT

Phase II of the Development of the Subliming Solid Control Rocket was concerned with two major tasks. They were:

a. The design, construction, test, and delivery of a flight subliming solid respin system

b. A performance study of the Subliming Solid Control Rocket

Additionally, work was accomplished concerning the Subliming Solid Control Rocket reliability, low flow rate measurement methods, propellant properties, and valve investigation.

The first major task resulted in the delivery of the SUBLEX Respin Rocket System (SRRS) for use on the OV2-1 Satellite. The system has a one-year life, 75 lbf-sec total impulse, and weighs only 1.67 lbs loaded, plus 1.0 lb of instrumentation. As a result of experience gained on the OV2-1 SRRS program, several recommendations can be made regarding recondensation control. Close coordination between the subcontractor and spacecraft manufacturer is essential to assure that:

a. The flight thermal environment is established to be firm

b. The storage environment is controlled and/or requirements established for shipping and handling of a SUBLEX control rocket

   1. Operate a trickle heater continuously or supply a special heater blanket operated by a battery or electrical plug-in

   2. Store and ship in a hermetically sealed container purged with dry nitrogen and packed with desiccant

   3. Control storage environment (humidity, temperature)

c. After installation of system in spacecraft:

   1. Operate valve heater continuously by use of an accessible spacecraft connector

   2. Back fill thruster lines and close off to the atmosphere

   3. Determine temperature environment in vicinity of propellant tank to assure that there are no overpowering heat sources.

The second major task, the performance study, was centered around the optimization of the nozzle. It has been found that significant losses occur in small nozzles that have not been correctly predicted by theory. The nozzle optimization program consisted of a literature survey and testing of several nozzle configurations. Several significant trends were discovered, including the fact that nozzle performance drops considerably as the throat Reynolds number decreases below 1,000. Further, throat size has a major bearing on nozzle performance in that for a given Reynolds number, nozzle performance decreases as throat size is reduced.

A third task was a reliability study of the Subliming Solid Control Rocket. Generic failure rates were determined for the system and its components. In addition, some test data were incorporated into the results. The results clearly demonstrated the fact that the solenoid valve was the least reliable component, and that system reliability could be improved by increased valve reliability.

The fourth task, the flow rate measurement study, involved experimental investigation of five methods of measuring gaseous flow below 1×10^{-4} lbm/sec. Although each of the methods yields reasonably accurate data, it is recommended that new methods be found and investigated. A literature search should be conducted to determine other flow measuring methods used in industry and to compare them with the methods described herein.

In task five, several properties of SUBLEX A were measured including vapor pressure vs temperature, crystalline and bulk density, and the heat of sublimation. It is recommended that further tests be conducted to determine:

a. Heat capacity

b. Thermal conductivity

c. Evaporation coefficient

d. Storage stability

e. Thermal stability

f. Hygroscopicity

g. Surface area as a function of particle size distribution

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1.0 INTRODUCTION

The Phase II Program (under follow-on Contract NAS 5-9070) was concerned with two main tasks. The first task was to design, construct, test, and deliver a Subliming Solid Respin Rocket System for use on the OV2-1 satellite. The second task had as its goal the optimization of the performance of the Subliming Solid Control Rocket. Performance optimization is dependent largely upon the optimization of the nozzle. In addition to these two major tasks, miscellaneous studies were conducted on subliming solid system reliability, low flow rate measurement methods, SUBLEX propellant properties, and characterization of several solenoid valve designs immediately available for test.

A description of the work that was accomplished on the Phase II program, plus the results and conclusions derived from the work, is presented in this final contract report.

2.0 OV2-1 SUBLEX RESPIN ROCKET SYSTEM

2.1 Introduction

2.1.1 General

One of the major tasks of Contract NAS 5-9070 was to design, construct, test, and deliver a Subliming Solid Respin Rocket System for flight use on the OV2-1 satellite. The OV2-1 satellite is an Air Force satellite built by Northrop Space Laboratories, Hawthorne, California. The results of this task are presented in the following paragraphs.

The OV2-1 (Orbital Vehicle 2, Model 1) consists of the basic structure of the original ARENTS vehicle with necessary structural modifications, fixed and deployable experiments, four (4) deployable solar cell paddles, spin stabilization capability, and necessary on-board and supporting equipment to meet the mission objectives. The overall mission objective is to place the OV2-1 satellite, with its complement of scientific experiments, into an elliptical orbit and to receive the optimum amount of data from the on-board space environment sensors for a period of one year.

2.1.2 Spin-Up Subsystem

Two functions are performed by the OV2-1 Spin-Up Subsystem: (1) initial spin-up, and (2) spin restoration.

The initial spin-up will be accomplished by conventional solid propellant rocket motors. Four miniature rockets will be fired simultaneously to provide the initial spin-up to approximately nine rpm. This will be accomplished within five seconds after separation from the booster. The Sublex Respin Rocket System will be used as a backup system in the event of a failure of one or more of the solid spin-up rockets.

Spin restoration will be accomplished by the SUBLEX Respin Rocket System. The angular velocity of the OV2-1 satellite will be maintained between three and ten rpm. Application of the required respin moment will be executed on command by a signal from the command subsystem. The determination of when spin restoration is required will be made at the OV2-1 operations control center and will be based on observation of the frequency of periodic variation in the telemetered data from the aspect sensor, magnetometer, or received r. f. power radiated from the satellite communications antenna. The command signal serves to actuate a solenoid valve on the SUBLEX propellant tank, which allows vapor to escape from the tank to the two nozzles operating in a couple to provide the respin required. Figure 1 shows the SUBLEX Respin Rocket System as installed in the OV2-1 satellite.

2.2 Design Requirements

The operational requirements placed on the OV2-1 SUBLEX Respin Rocket System are that it:

a. Maintain vehicle spin rate between three and ten rpm

b. Have an operational life of one year

c. Weigh less than five pounds

d. Require less than 100 cubic inches of space

e. Have a total impulse of 60 lb-sec

f. Have an average minimum thrust level of 10^{-3} lb

g. Operate in a temperature environment of 70°F $\pm$ 20°F

h. Require three watts maximum at a 1% duty cycle

i. Meet the environmental requirements set forth in Northrop Specification NSL 64-211A. (Briefly, these consist of random vibration, shock, acceleration, and thermovacuum testing.)

2.3 OV2-1 SUBLEX Respin Rocket System Design

2.3.1 System Description

The OV2-1 SUBLEX Respin Rocket System, as shown in Figure 2, consists simply of a SUBLEX-filled propellant tank, a filter assembly, a solenoid valve, a choking orifice, a propellant exhaust manifold, two exhaust lines, two exhaust nozzles, and instrumentation. The SUBLEX Respin Rocket System instrumentation system consists of a signal conditioning unit, one pressure transducer, one temperature transducer, two thermistors, and one valve heater.

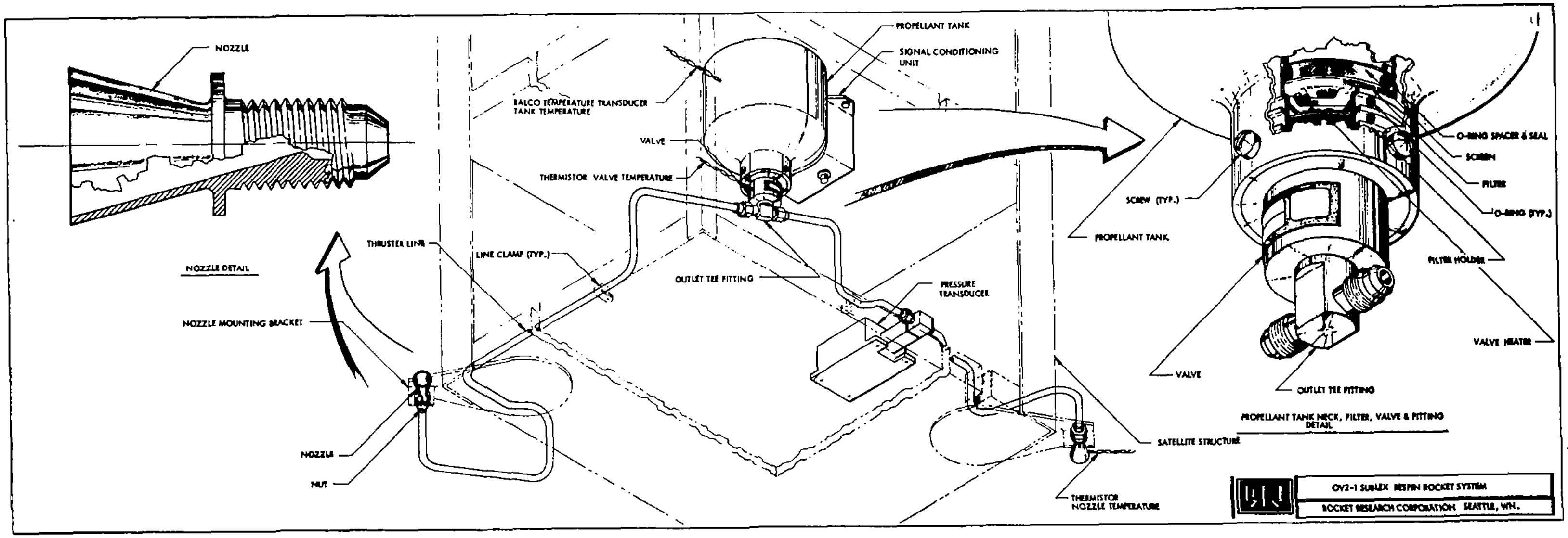
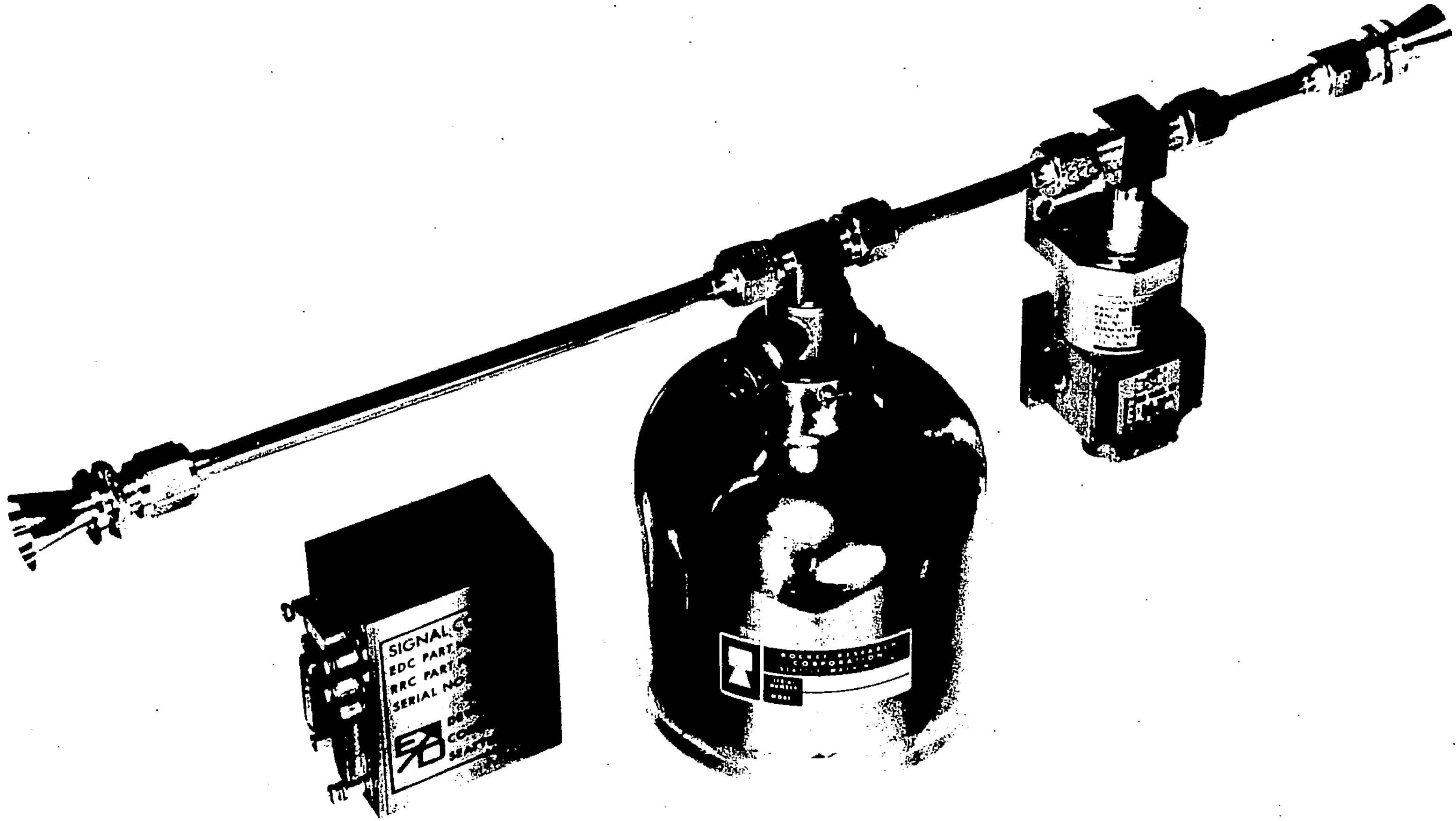

NOZZLE
NOZZLE DETAIL
NOZZLE MOUNTING BRACKET
NOZZLE
NUT
THRUSTER LINE
LINE CLAMP (TYP.)
BALCO TEMPERATURE TRANSDUCER
TANK TEMPERATURE
VALVE
THERMISTOR VALVE TEMPERATURE
OUTLET TEE FITTING
PROPELLANT TANK
SIGNAL CONDITIONING
UNIT
PRESSURE
TRANSDUCER
THERMISTOR
NOZZLE TEMPERATURE
SATELLITE STRUCTURE
PROPELLANT TANK
SCREW (TYP.)
O-RING SPACER & SEAL
SCREEN
FILTER
O-RING (TYP.)
FILTER HOLDER
VALVE HEATER
VALVE
OUTLET TEE FITTING
PROPELLANT TANK NECK, FILTER, VALVE & FITTING
DETAIL
OV2-1 SUBLEX RESPIN ROCKET SYSTEM
ROCKET RESEARCH CORPORATION SEATTLE, WN.

FIGURE 1



OV2-1 SUBLEX RESPIN ROCKET SYSTEM

FIGURE 2

2.3.1.1 Propellant

The subliming solid propellant used in the SUBLEX Respin Rocket System is proprietary to Rocket Research Corporation and shall be designated herein as SUBLEX A. (Reference Final Report on Contract NAS 5-3599.) SUBLEX A has the following physical characteristics:

Molecular weight: Vapor phase 25.5

Vapor pressure at 70°F: 7 psia

Vapor pressure change with temperature: Approximately a factor of two for every 20°F

Heat of sublimation: 782 Btu/lb @ 77°F

Density (as loaded): .025 lb/cu in.

Specific Impulse: 85 sec theoretical at 50:1 ratio

Assuming an actual realized specific impulse of 75 seconds, the propellant weight can be determined:

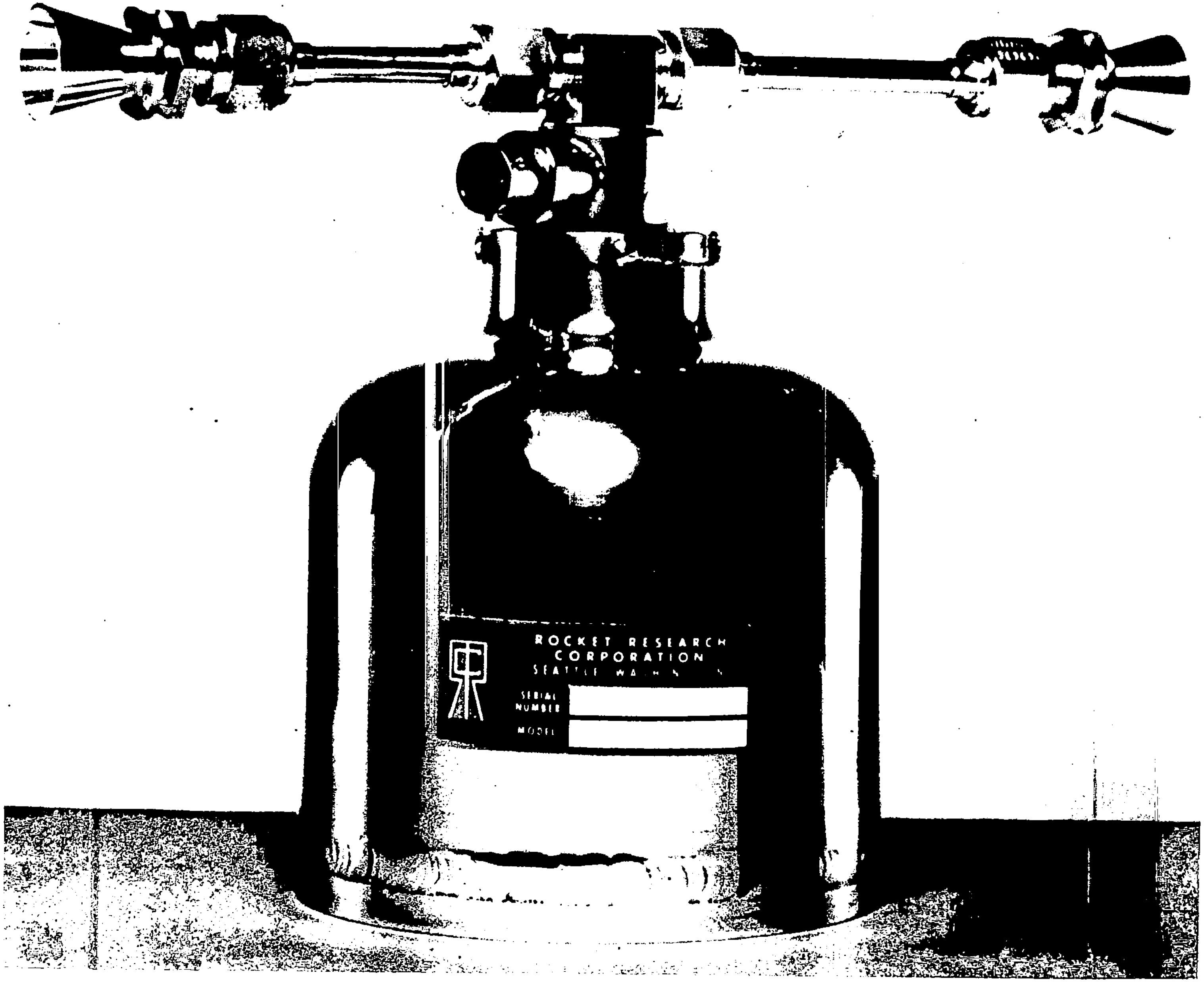
$$W = \frac{\text{Total Impulse}}{\text{Specific Impulse}}$$

$$= \frac{60}{75} = 0.8 \text{ lbm}$$

Allowing for inaccuracies and propellant losses (which includes losses due to leakage), a total propellant load of 1.0 lb was used in the OV2-1 SUBLEX Respin Rocket System.

2.3.1.2 Propellant Tank

The propellant tank is a flat-bottomed, thin-walled aluminum cylinder with nominal dimensions of 4 inches in diameter and 4.4 inches high; it is located on the bottom side of the center shelf of the OV2-1 satellite (see Figure 3). The propellant tank is designed for a burst pressure of 120 psia (four times the maximum pressure anticipated during storage and eight times the maximum operating pressure expected during flight). The propellant tank was left in the "as machined" condition inside and polished to a high degree outside.



PROPELLANT TANK

FIGURE 3

2.3.1.3 Filter Assembly

The filter assembly is located at the outlet of the propellant tank. It serves to retain the solid propellant granules in the propellant tank and to prevent escape of solid propellant particles into the valve and exhaust lines. The filter assembly contains three woven stainless steel screens in series consisting of:

a. A 50-mesh screen in contact with the propellant

b. An intermediate 200-mesh screen

c. A 40 micron filter

The 50 and 200 mesh stainless steel screen discs have an effective flow area of approximately 1.2 square inches. The 40 micron filter is a Bendix Poromesh Disc with an effective flow of 6.6 square inches (see Figure 4).

2.3.1.4 Solenoid Valve

The solenoid valve is attached directly to the filter assembly. It is a coaxial type valve, built by the Eckel Valve Company, San Fernando, California. This valve has an effective orifice diameter of .04 inch, requires 2.0 watts to operate at 32 VDC, and weighs 0.13 lb. Further specifications are given in Figure 5.

2.3.1.5 Prechoking Orifice

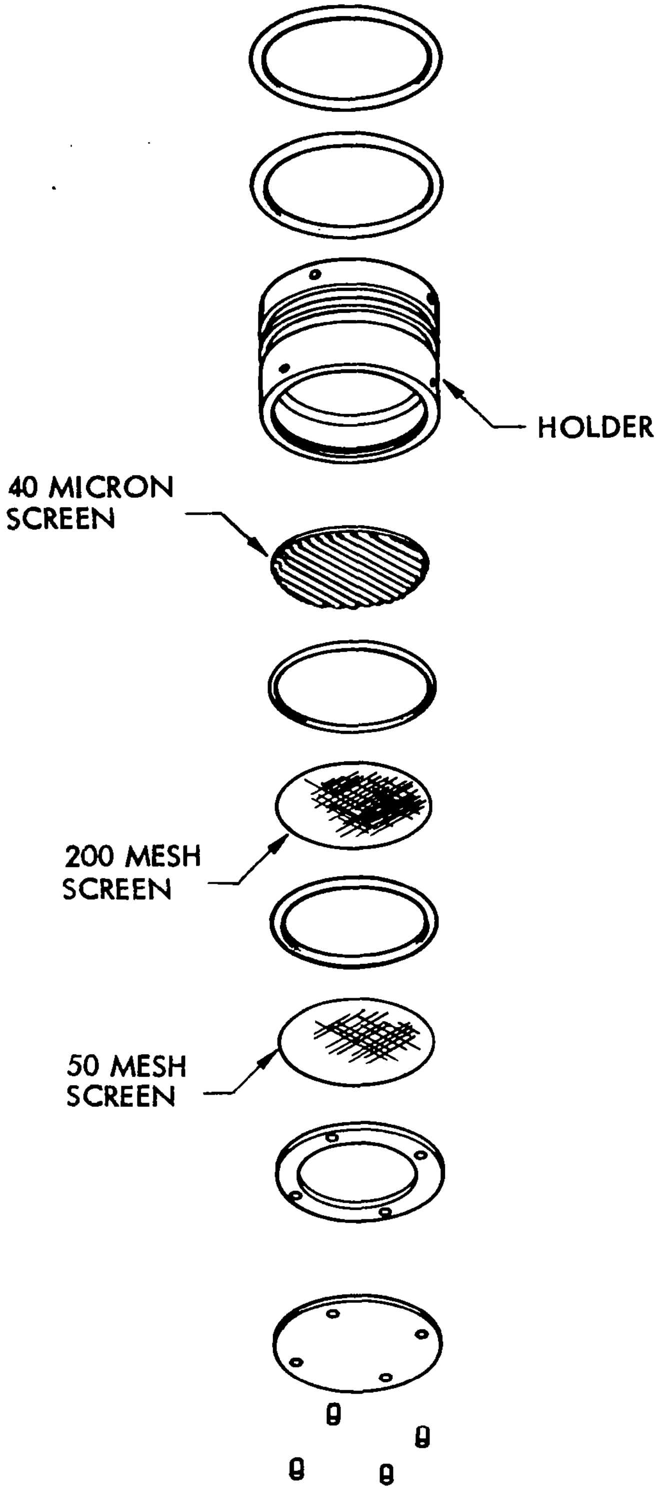
A prechoking orifice is located in the valve outlet to throttle propellant flow. It is a sharp-edged, 0.038 inch diameter orifice.

2.3.1.6 Outlet Manifold

An outlet manifold is attached to the propellant valve. The outlet manifold serves to divide the exhaust flow to the two exhaust lines and nozzles.

2.3.1.7 Exhaust Lines

Two 1/4" OD 6061-T6 aluminum propellant lines are used to connect the outlet manifold with the exhaust nozzles. (The exhaust lines are routed in the most convenient manner inside the space vehicle.)

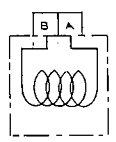


FILTER HOLDER ASSEMBLY

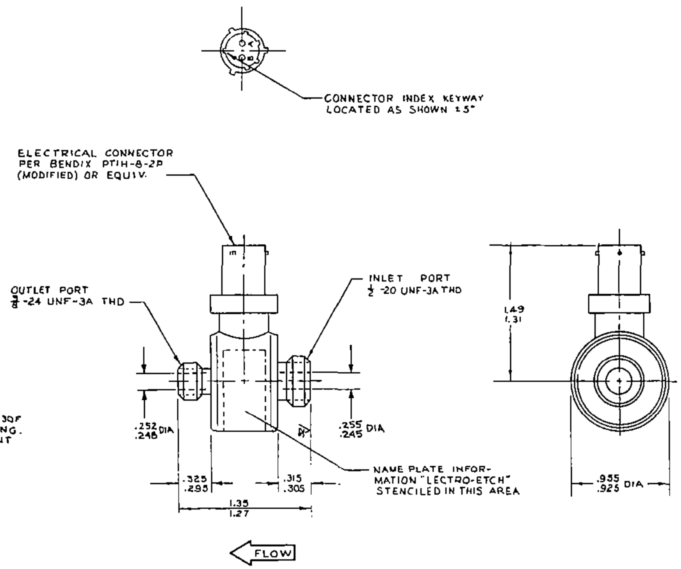
FIGURE 4

REVISIONS			
SYM	DESCRIPTION	DATE	APPD

FLOW & OPERATIONAL SCHEMATIC



WIRING DIAGRAM



12. SEAL
    SOFT SEAT
11. CONTAMINATION
    ALLOW PASSAGE OF 300 MICRON PARTICLES WITHOUT VALVE SEIZING
    ALLOW PASSAGE OF 40 MICRON PARTICLES WITHOUT EXCEEDING LEAKAGE SPECIFICATION
10. MATERIALS
    CORROSION RESISTANT STAINLESS STEEL (302, 303, 430F STAINLESS WELDED CONSTRUCTION) NO INTERNAL PLATING.
    NO COPPER, BRASS OR BRONZE IN CONTACT WITH EFFLUENT
    TEFLON ELASTOMER
9. WEIGHT
    .2 LBS MAX
8. TEST
    VALVE SHALL BE 100% TESTED IN ACCORDANCE WITH E.V.C. TEST PROCEDURE NO. AF77T-A119
7. ELECTRICAL
    VOLTAGE: 24 TO 32 DC DUTY: CONTINUOUS
    POWER: 2.0 WATTS MAX @ 32 VDC & 70°F
    SOLENOID: EXPLOSION PROOF & HERMETICALLY SEALED
6. EFFLUENT
    CLASSIFIED
5. EQUIVALENT ORIFICE
    .040 MINIMUM Cd=1.0
4. LEAKAGE
    INTERNAL: 5.0 CC/HR MAX AT 30 PSI DIFFERENTIAL
    EXTERNAL: NONE FROM 0 TO 60 PSIG
3. PRESSURES
    OPERATING: VACUUM TO 30 PSIA
    PROOF: 60 PSIG BURST: 90 PSIG
2. TEMPERATURE RANGE
    AMBIENT: +32°F TO +140°F
1. ONLY THE ITEM DESCRIBED ON THIS DRAWING WHEN PROCURED FROM THE VENDOR(S) LISTED HEREON IS APPROVED BY ROCKET RESEARCH CORP. FOR USE IN THE APPLICATION(S) SPECIFIED HEREON. A SUBSTITUTE ITEM SHALL NOT BE USED WITHOUT PRIOR TESTING AND APPROVAL BY ROCKET RESEARCH CORP. OR BY THE GOVERNMENT PROCURING ACTIVITY.

SOURCE CONTROL DRAWING

APPROVED SOURCES OF SUPPLY	
VENDOR & VENDORS PART NO.	APPLICATION
ECKEL VALVE CO. SAN FERNANDO, CALIF PART NO: AF77C-A119-1	PROPELLANT SHUTOFF VALVE IN SUBLIMING SOLID CONTROL ROCKET

	REQ'D	PART NUMBER	NOMENCLATURE	STOCK SIZE	MATERIAL	MAT'L SPEC	HT-TR
REL.	LIST OF MATERIALS						

ROCKET RESEARCH CORPORATION, Seattle, Wash.				
Scale 2/1	Drawn	D. L. AKITA	10-14-64	DRAWING TITLE VALVE, SOLENOID OPERATED, NORMALLY CLOSED, SHUTOFF
Used On 30-1073	Checked	[illegible]	11-6-64	
	Stress Ck.			
	Qual. Cont.	[illegible]	[illegible]	
Tolerances Except As Noted Linear ± .010 Angular ± ½°	Engineer	H. HAZE	11-24-64	DRAWING NUMBER 30-1029 Sht 1 of 1
	Contract Number 171-23			Rev. Sym.

FIGURE 5

2.3.1.8 Exhaust Nozzles

Two exhaust nozzles, manufactured from 6061-T6 material, are located at opposite corners on the bottom panel of the satellite. The nozzles are of conventional conical design, with an area ratio of 50:1, an included half-angle of 15 degrees, and a throat diameter of 0.10 inch. Figure 6 shows the OV2-1 SUBLEX Respin Rocket System nozzle configuration.

2.3.1.9 Signal Conditioning Unit

The Signal Conditioning Unit (SCU), shown at the lower left-handcorner in Figure 1, is mounted near the propellant tank on the lower side of the center shelf of the OV2-1 satellite. The Signal Unit was constructed by the Electro Development Corporation, Seattle, Washington. It consists of three Bendix connectors mounted on an aluminum housing, with all internal openings being potted. The Signal Conditioning Unit provides the interconnection between the SUBLEX Respin Rocket System and the OV2-1 satellite power and telemetry subsystems. A schematic diagram of the Signal Conditioning Unit showing how it connects into the OV2-1 satellite and the SUBLEX Respin Rocket System is shown in Figure 7.

2.3.1.10 Pressure Transducer

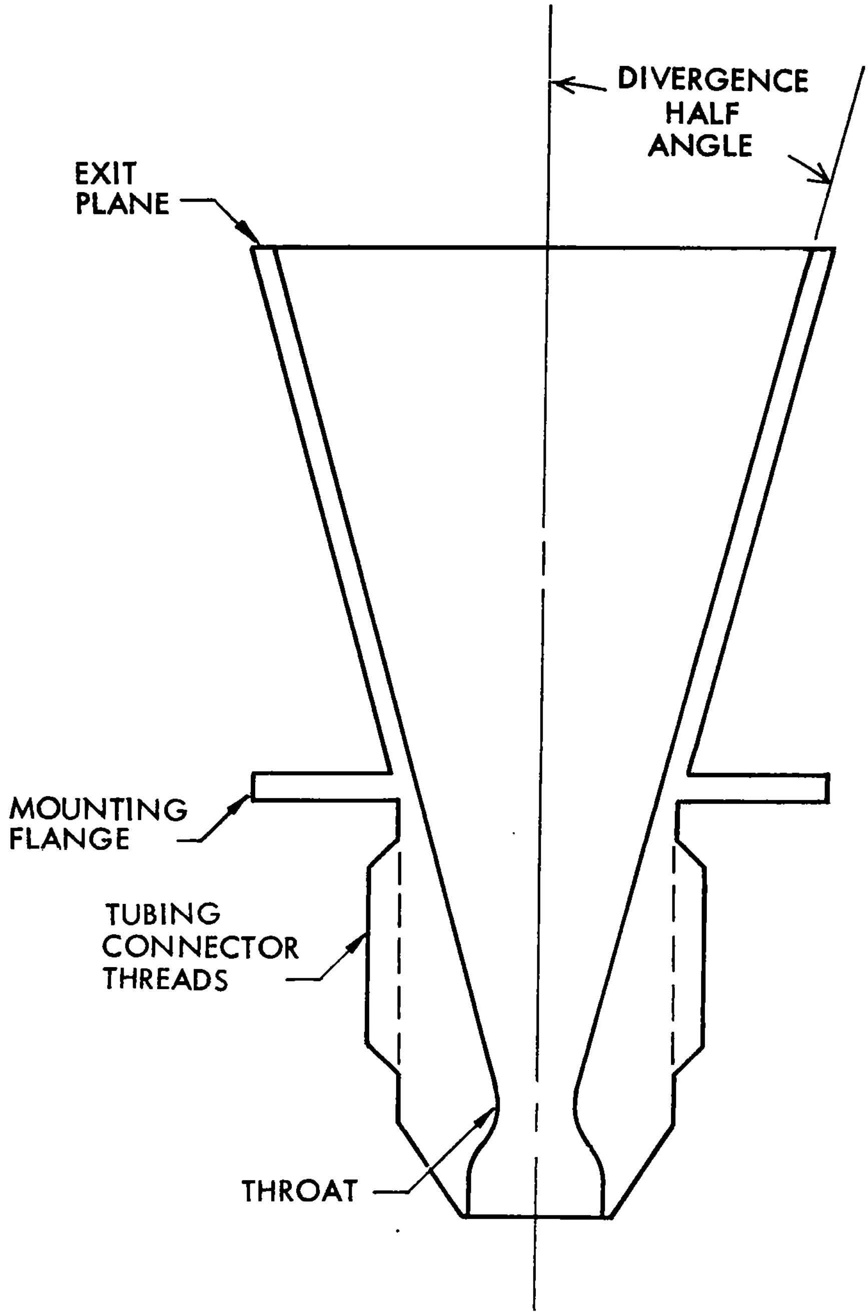
The pressure transducer (shown in the top right-hand corner of Figure 2) was manufactured by Wiancko Engineering, Pasadena, California. It has a 0 - 5 psig range, operates on 24 - 32 VDC input voltage, and generates a five VDC output voltage to the telemetry. It is mounted on the lower shelf of the satellite and is used to measure line pressure.

2.3.1.11 Temperature Transducer

One temperature transducer is mounted on the OV2-1 SUBLEX Respin Rocket tank shell to measure tank temperature. It is a Balco type operating on a 28 VDC input voltage and generating, through the Signal Conditioning Unit, a five VDC output voltage to the satellite telemetry system.

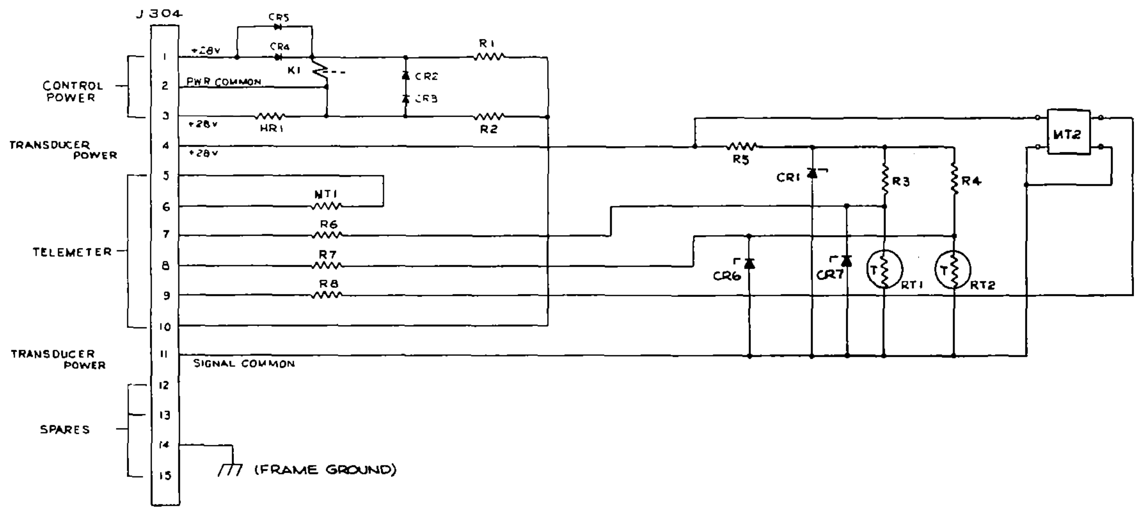
2.3.1.12 Thermistors

Two thermistors are used to measure the SUBLEX Respin Rocket System valve and nozzle temperature so that the SUBLEX Respin Rocket System status can be monitored



OV2-1 SRRS NOZZLE CONFIGURATION

FIGURE 6



COMPONENTS (ENGRG REF)				
REF DES	NOMENCLATURE	VALUE	PART NO	SUPPLIER
R1	RESISTOR	200KΩ, 1%	RN60C	IRC ELECTRA ACI
R2	RESISTOR	34 KΩ, 1%	RN 60C	IRC ELECTRA ACI
R3	RESISTOR	150KΩ, 1%	RN 60C	IRC ELECTRA ACI
R4	RESISTOR	150KΩ, 1%	RN 60C	IRC ELECTRA ACI
R5	RESISTOR	8250Ω, 1%	RN 60C	IRC ELECTRA ACI
R6	RESISTOR	2000Ω, 1%	RN 60C	IRC ELECTRA ACI
R7	RESISTOR	2000Ω, 1%	RN 60C	IRC ELECTRA ACI
R8	RESISTOR	2000Ω, 1%	RN 60C	IRC ELECTRA ACI
HR1	HEATER COIL	7850Ω		RRC
MT1	BALCO TEMP TRANSDUCER			NORTHROP
MT2	PRESSURE TRANSDUCER		P2-3002	WIANCKO
RT1	THERMISTOR	20K ±15%@25°C	GA42J2	FENWAL
RT2	THERMISTOR	20K±15%@25°C	GA42J2	FENWAL
CR1	ZENER DIODE	20 VOLT	IN3525	CONT. DEVICES
CR2,3,4,5	DIODE		IN645	
K1	SOLENOID VALVE			
J 304	CONNECTOR		DAM-15P	CANNON
CR6	ZENER DIODE	7.5 VOLT	IN3515	CONT. DEVICES
CR7	ZENER DIODE	7.5 VOLT	IN3515	CONT. DEVICES

REVISIONS			
LTR	DESCRIPTION	DATE	APPD
A	1. REVISED SCHEMATIC: ADDED CR3, CR4, & CR5 2. IN COMPONENTS LIST FROM R1 THRU R8: IN VALUE COLUMN REMOVED ⅛W, IN PART NO. COLUMN RN60C WAS MEAT-2, IN SUPPLIER COLUMN IRC ELECTRA ACI WAS IRC 3. IN COMPONENTS LIST FOR CR1: REMOVED 250 MW FROM VALUE COLUMN, ADDED IN3525 IN PART NO. COLUMN, ADDED CONT. DEVICES IN SUPPLIER COLUMN 4. IN COMPONENTS LIST FOR CR2: ADDED 3,4,5 IN REF DES COLUMN, REMOVED 45 VOLT 250 MW FROM VALUE COLUMN, ADDED IN645 IN PART NO. COLUMN 5. IN COMPONENTS LIST FOR P304: DAM-15P WAS DAM-15P-NW IN PART NO. COLUMN	11-4-64	DWN: ARITA CK: [illegible] APPD: [illegible]

REQ'D	PART NUMBER	NOMENCLATURE	STOCK SIZE	MATERIAL	MAT'L SPEC	HT-TR
REL.	LIST OF MATERIALS					

ROCKET RESEARCH CORPORATION, Seattle, Wash.

Scale
Used On
Tolerances Except As Noted
Linear ± Angular ±
Drawn
Checked
Stress Ck.
Qual. Cont.
Engineer
Contract Number 171-23

DRAWING TITLE
SCHEMATIC DIAGRAM
OV2-1 FLIGHT INSTRUMENTATION

DRAWING NUMBER 30-1036
Sht 1 of 1
Rev. Sym A

FIGURE 7

during flight. The thermistors are a Fenwal, Type GA42J2, operating on a 28 VDC voltage and generating, through the Signal Conditioning Unit, a five VDC output voltage to the satellite telemetry system.

2.3.1.13 Valve Heater

One valve heater is located around the boss on the filter holder assembly outlet, and operated continuously to supply heat to the valve inlet area for the purpose of preventing recondensation. The heater consists of .0011 inch diameter wire wound around a thin aluminum shell and requires less than 0.5 watt of continuous power at 32 VDC.

2.3.2 System Calculations

2.3.2.1 Flow Rate

The OV2-1 SUBLEX Respin Rocket System is designed to deliver 0.01 lb of thrust at a tank pressure of 7.0 psia. Under these conditions, the flow rate will be:

$$\dot{w} = \frac{\text{Thrust}}{\text{Specific Impulse}} = \frac{1 \times 10^{-2}}{75} = 1.33 \times 10^{-4} \text{ lbm/sec}$$

Due to self-cooling of the propellant during a pulse, the thrust level could drop to 1×10^{-3} lbf, which would also drop the flow rate to 1.33×10^{-5} lbm/sec.

2.3.2.2 Nozzle Pressure

The pressure immediately upstream of the nozzle, or line pressure, is established by recondensation considerations. Since the sublimation process is a reversible process, any stable subliming solid material that is vaporized will recondense when recooled. Therefore, when components or lines which are in open communication with the full pressure of the propellant vapor are cooled below the propellant temperature, the propellant may recondense to a solid at these points. However, if the component is exposed to less than full propellant vapor pressure, recondensation will not occur until component temperature drops to the SUBLEX equilibrium temperature corresponding to the pressure to which the component is exposed. In other words, SUBLEX vapor will not recondense if the component pressure is less than the corresponding equilibrium vapor pressure for the component temperature. Therefore, recondensation in the OV2-1 SUBLEX Respin Rocket System lines can be prevented by dropping the line pressure to a point below the propellant equilibrium pressure at the minimum expected line temperature. Line pressure can be controlled to any desired value by choking propellant flow at the valve and then setting line pressure by adjusting the nozzle size. Neglecting temperature changes,

$$P_n A_n = P_t A_o$$

where:

P_n = total nozzle pressure

A_n = total nozzle area

P_t = tank pressure

A_o = choking orifice area

Then

$$\frac{P_t}{P_n} = \frac{A_n}{A_o}$$

The ratio P_t/P_n will be set so that recondensation will not occur in the lines under the worst possible satellite temperature condition, which is where the propellant temperature is 100°F (propellant pressure = 18 psia) and the line temperature is 30°F (line pressure = 1.8 psia). This condition yields a pressure ratio of:

$$\frac{P_t}{P_n} = \frac{18}{1.8} = 10$$

For an additional safety margin, the OV2-1 system was designed with a pressure ratio of 21.

2.3.2.3 Prechoking Orifice Size

Assuming isentropic flow of a perfect gas:

$$\dot{w} = \sqrt{\frac{gkM}{R}\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}}\ \frac{P_t A_o}{\sqrt{T_o}}$$

where:

k = ratio of specific heats
= 1.31

R = gas constant = 1,544 ft-lb/lb-mole °R

M = average molecular weight = 25.5 lb/lb-mole

g = gravitational constant = 32.2 ft/sec^2

P_t = tank pressure, psia

A_o = choking orifice effective area, in^2

T_o = temperature, °R

$\dot{w}$ = flow rate, lbm/sec

Assume:

P_t = 7.0 psia

$\dot{w}$ = 1.33×10^{-4} lb/sec

T_o = 530°R

Then:

A_o = 7.40×10^{-4} in^2

or

D_o = .0338 in.

2.3.2.4 Nozzle Size

For system sizing, assume a pressure ratio of 15. Then:

$$\frac{P_t}{P_l} = 15$$

or

$$\frac{P_t}{P_l} = \frac{A_{n_T}}{A_o} = 15$$

where:

$$A_{n_T} = \text{total nozzle area, in}^2$$

then:

$$A_{n_T} = 1.35 \times 10^{-2} \text{ in}^2$$

$$A_n = \frac{A_{n_T}}{2} = 6.75 \times 10^{-3} \text{ in}^2$$

$$D_n = .093 \text{ in.}$$

$$D_{n_{actual}} = 0.10 \text{ in. (which yields a pressure ratio of 21)}$$

The optimum nozzle exit diameter and half-angle cannot be predicted due to uncertainty as to relative magnitudes of losses associated with small nozzles. However, based on previous experience, a 50:1 area ratio nozzle with a 15° half-angle was used. There was not sufficient time to develop experimentally the optimum nozzle configuration. However, based on data subsequently obtained, the gains in performance that would have been realized by using the optimum nozzle configuration would have been small compared to the total performance.

2.3.2.5 Line Size

A line size of area much larger than the nozzle throat area was chosen to minimize the pressure drop through the line and assure nozzle choking. One-quarter inch aluminum lines with 0.035 inch wall thickness were chosen.

2.4 Test and Results

2.4.1 Approach

The development of the OV2-1 SUBLEX Respin Rocket System included testing in four main areas:

a. Prevention and control of recondensation

b. Reliable solenoid valve operation

c. System performance characteristics

d. Qualification and acceptance testing prior to delivery

Each of these areas is discussed individually in the following paragraphs.

2.4.2 Recondensation Tests

In the case of the OV2-1 SUBLEX Respin Rocket System (SRRS), specific measures have been taken to prevent the possibility of recondensation occurring in critical areas during flight, including the lines, nozzles, propellant valve, and filter holder assembly. Prevention of recondensation has been assured by design and analysis of the SUBLEX Respin Rocket System with regard to its expected thermal environment, as well as by testing of components under the operating conditions expected in flight.

Recondensation in the valve and filter holder assembly will be prevented during flight by heating the area in two ways. First, a small electric heater generating 1/2 watt of continuous power is attached to the neck of the filter holder, raising the temperature at that point. Second, heat generated by a 2.0 watt transmitter mounted close to the valve and filter holder will be funneled to the top area of the OV2-1 SUBLEX propellant tank by the use of an aluminized Mylar shroud. These two methods, along with the assurance from the NSL thermal analysis that no adverse temperature gradients would occur, help to ensure that this area will always be as warm or warmer than the propellant, thereby preventing recondensation.

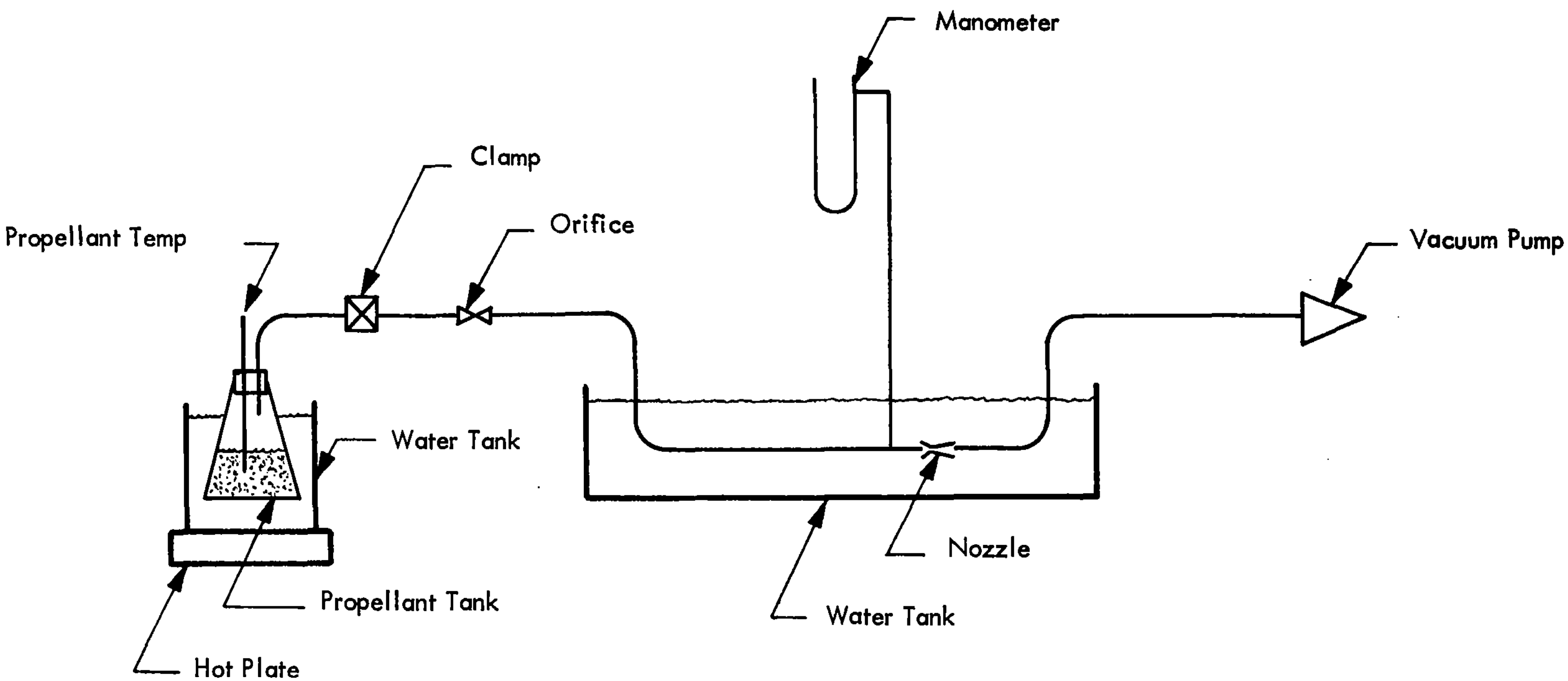
Recondensation in the lines during flight also will be prevented in two ways. First, the line pressure will be reduced to a point where recondensation cannot physically occur even when the solid propellant is at the hottest expected temperature and the lines are the the coldest expected temperature. This is accomplished by throttling propellant flow at the valve outlet and then adjusting line pressure by correct nozzle sizing. Specifically, the line pressure shall be adjusted to reach a maximum of 1.75 psia when the tank temperature is at 100°F. Pressure of 1.75 psia corresponds to a SUBLEX equilibrium vapor temperature of 30°F. Therefore, recondensation in the lines should not occur within the expected temperature limits of the OV2-1 satellite. Second, the lines shall be wrapped entirely with aluminized Mylar for insulation to help prevent temperature extremes.

It has been postulated that the expansion of SUBLEX vapor through a nozzle will cause a sufficient vapor temperature drop so as to cause recondensation in the nozzle. While in theory such recondensation may be possible, Rocket Research Corporation has proven by past tests that the nozzle temperature must be dropped significantly below the recondensation temperature corresponding to the exhaust vapor pressure for recondensation to occur; that is, due to the small nozzle size and the large gradients existing therein, extensive supersaturation may be expected due to the fact that the finite rate of homogeneous recondensation cannot keep pace with the very rapid temperature drop in a small nozzle. Further, the OV2-1 nozzles will be attached to the OV2-1 structure by a bracket which provides good thermal contact, and it is not anticipated that the nozzle temperature will drop to a point where recondensation could occur.

2.4.2.1 Propellant Lines and Nozzles

Recondensation tests on the OV2-1 SUBLEX Respin Rocket System lines and nozzles were conducted under the most severe environmental conditions expected on the OV2-1 satellite as provided by NSL. For simplicity, one line and one nozzle were used in this test, but the pressure ratio between the tank and the line was adjusted to simulate the actual expected flight pressure ratio. The propellant tank temperature was held at 100°F, 10°F over the maximum expected satellite temperature, while the line and nozzle were maintained at 30°F, 10°F lower than the minimum expected temperature. All tests were run under flow conditions.

Figure 8 shows a schematic diagram of this test apparatus. The propellant tank consisted of a glass flask filled with approximately 0.5 lb of SUBLEX, and a rubber



SCHEMATIC DIAGRAM
RECONDENSATION TEST
LINES AND NOZZLES

FIGURE 8

stopper. Propellant temperature was maintained by immersing the flask into a water tank and heating the water by means of a hot plate. A thermometer was located in the flask to monitor propellant temperature. An orifice was located at the outlet of the propellant tank. Approximately 10 inches of line and the nozzle were then routed through a 30°F water bath and directly to a vacuum pump. Both tygon tubing (1/2 in. O. D., 1/4 in. I. D.) and aluminum tubing, (1/4 in. O. D., 0.035 in. wall) were tested. A manometer was hooked in the line upstream of the nozzle to monitor line pressure. The nozzle throat diameter was 0.135 inch and the exit diameter 0.657 inch, giving a 23.7 area ratio. The nozzle half-angle was 15°. The area ratio between the nozzle and the orifice was approximately 21 to 1, thus giving a pressure ratio between the tanks and lines of approximately 21 to 1. Specifically, initial tank pressure was 18 psia, and initial line pressure was 0.86 psia. In all tests run under these conditions, no signs of recondensation occurred anywhere in the system, including the line and nozzle.

To determine the limits of the system (that is, to determine the line temperature at which recondensation would occur), the nozzle temperature was lowered in steps until plugging occurred. Recondensation first appeared in the line immediately upstream of the nozzle when the line and nozzle temperature was 0°F (100°F temperature differential). However, it gradually disappeared within three minutes. This procedure was repeated, decreasing line and nozzle temperature in steps of approximately 10°F down to -58°F. Each time recondensation would occur in the line almost immediately and then gradually disappear, but slightly slower at each lower temperature, until at -45°F, it took 50 minutes to clear the lines completely. At -58°F recondensation continued to grow, and completely plugged the line and nozzle. This is a total temperature differential between the tank and the line of 158°F. Since the total temperature differential of the OV2-1 satellite is expected to be only 40°F, the probability of plugging the lines or nozzles in flight due to recondensation during flight is considered to be negligible.

2.4.2.2 Filter Assembly

Specific tests were not conducted on the filter holder assembly on this program, since extensive tests were previously conducted on similar filter assemblies during the Phase I program. (See Final Report, Contract NAS 5-3599.) The filter assemblies were placed on propellant flasks and cooled to temperatures down to 0°F, while the propellant temperature was maintained at 60°F to 70°F. This is a particularly severe test, since in most flight applications the filter assembly is in intimate contact with the propellant tank and therefore follows propellant temperature closely. Recondensation occurred only lightly on the screen filters, and caused no significant increase in pressure drop under flow conditions.

2.4.2.3 Valve

Recondensation tests on the valve were conducted as part of the valve life tests and are described in detail in paragraph 2.4.3. Briefly, the results indicated that recondensation will occur in the valve if no heat is applied to that area. Sometimes a plug is formed at the valve inlet. However, the plug does not prevent the valve poppet from opening, and it eventually sublimes away, allowing free propellant flow. No valve failures occurred during the test program. It should be noted that the recondensation plug occurred only when the 1/2 watt valve heater was off. No plugging occurred while the heater was operating.

2.4.2.4 Recommendations

As described in the preceding paragraphs, careful design, analysis and tests were performed to control recondensation in flight. However, it is equally important to control recondensation during ground storage and preflight checkout. During the prelaunch operations on the OV2-1 SRRS at Cape Kennedy, a malfunction occurred on the SRRS due to recondensed propellant in the valve and filter that was a direct result of failure to control the ground storage environmental conditions. Briefly, particles of propellant recondensed in the valve and filter area (due to adverse temperature conditions unknown at that time) during the long storage period. Thus valve opening was prevented because of a blockage of the flow passage. As described earlier, this problem is prevented during flight by the valve heater which creates a favorable temperature gradient in the system (keeping the valve warmer than the propellant at all times), thus driving away any recondensation that might have accumulated during storage and preventing any further recondensation from occurring.

Recondensation can occur during storage either from normal storage temperature cyclings or because of adverse temperature gradients created by some nearby source. In the case of the OV2-1 SRRS , the spacecraft battery, located on the side of the tank away from the valve represented a source creating unfavorable temperature conditions. When being recharged, the battery dissipated heat increasing its temperature as much as 40°F over the tank temperature. This temperature increase was transmitted through a common bulkhead to the SUBLEX propellant tank base. Therefore, propellant was "driven" toward the vicinity of the valve and filter during its storage period. The valve heater was operated prior to the attempt to open the valve, however, its effect was negated due to the overpowering effect of the battery.

It has been experimentally verified that when air and moisture contact the propellant for an extended period of time, a reaction can occur to form nonvolatile solids. Under certain conditions, these solids can interfere with proper operation of the propellant valve, either by preventing opening or proper closing. Air and moisture were able to contact the OV2-1 SRRS valve (through the lines) during much of the SRRS storage. It is then possible that any propellant immediately upstream of the valve seat was contaminated due to a slow diffusion of air and water vapor through the valve seat. This contamination is not necessarily serious if the amount of propellant in the valve area is small. However, as described above, it appears likely that there was considerable propellant in the valve area which, when subject to contamination, could have formed a nonvolatile solid material which would then interfere with the valve operation. Under normal storage conditions, the propellant tank is topped with dry nitrogen to four or five psig, so that a positive outflow of gas occurs to prevent air and moisture from contacting the valve.

As a result of the experience gained on the OV2-1 SRRS program, several recommendations can be made. Future spacecraft applications must require close coordination between Rocket Research Corporation and the spacecraft manufacturer to assure that:

a. The flight thermal environment is established to be firm

b. The storage environment is controlled and/or requirements established for shipping and handling of a SUBLEX control rocket

1.. Operate a trickle heater continuously or supply a special heater blanket operated by a battery or electrical plug-in

2. Store and ship in a hermetically sealed container purged with dry nitrogen and packed with desiccant

3. Control storage environment (humidity, temperature)

c. After installation of system in spacecraft:

1. Operate valve heater continuously by use of an accessible spacecraft connector

2. Back fill thruster lines and close off to the atmosphere

3. Determine temperature environment in vicinity of propellant tank to assure that there are no overpowering heat sources

2.4.3 Propellant Valve Testing

The OV2-1 SUBLEX Respin Rocket System (SRRS) propellant valve is the only component on the system with moving parts, and is therefore subject to seizure due to recondensation of SUBLEX propellant. Further, it is dependent upon electrical power for operation. For these reasons, the propellant valve is the most critical component on the SUBLEX Respin Rocket System from a reliability standpoint.

The installation of propellant valves on the SUBLEX Respin Rocket System is specifically arranged to prevent valve failure in flight due to recondensation. The valves are a coaxial type and are oriented reverse to normal flow so that recondensation cannot occur on the moving surfaces around the plunger and seize the valve during storage. Therefore, the only point at which recondensation could render the valve inoperative is in the inlet orifice and seat. Under adverse temperature differentials, it is possible to build up a sufficient deposit of recondensed propellant that could inhibit the flow of propellant vapor through the seat and orifice area. Such an adverse negative temperature gradient condition will in general occur only if the valve is located in an area colder than the propellant tank. In the case of the OV2-1 Respin Rocket design, the propellant valve is located on the propellant tank in an area of positive temperature gradient during flight. Ground storage conditions must also be controlled as described in paragraph 2.4.2.4.

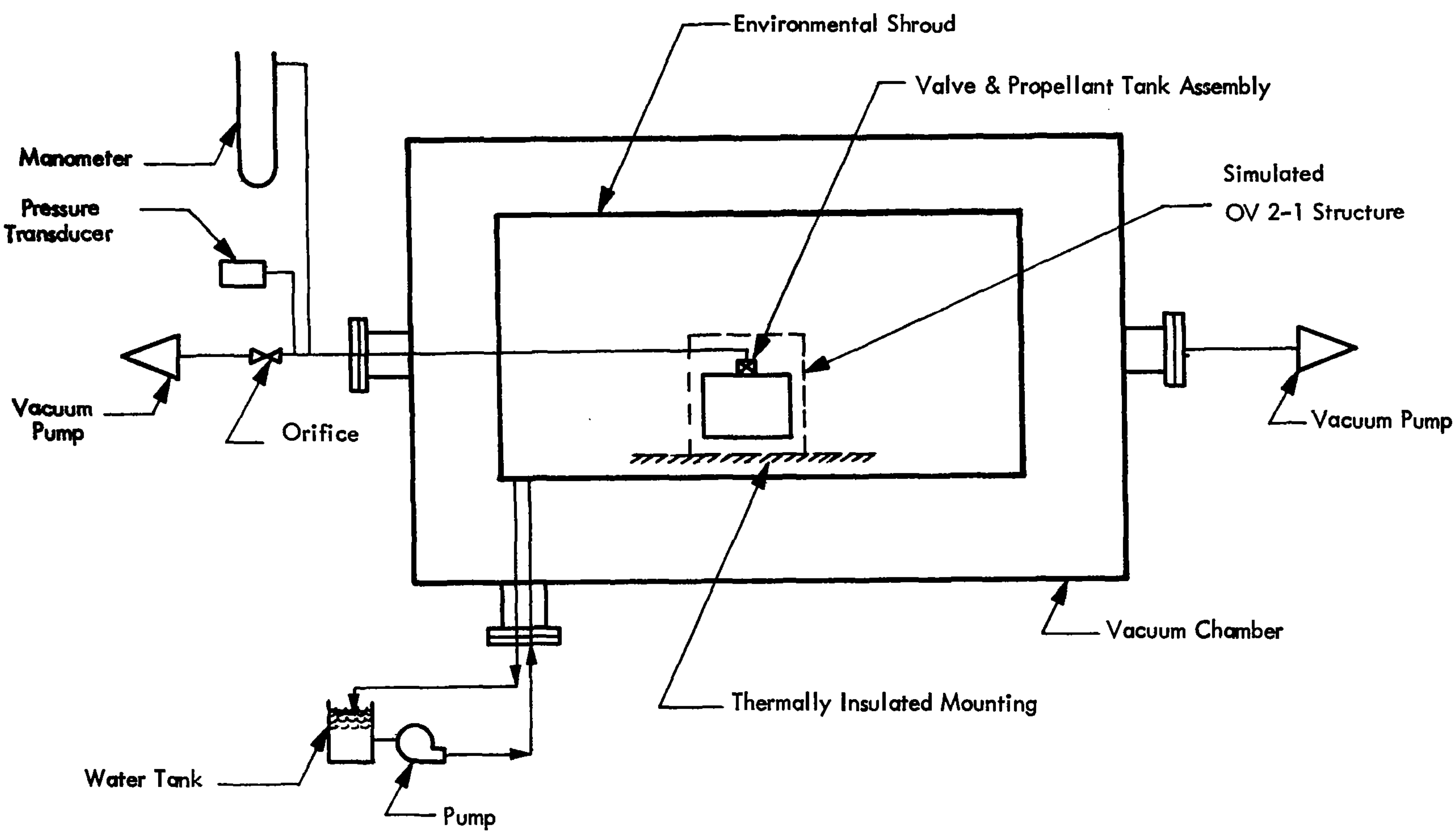
During this program, two separate valve life tests were conducted: One test on three prototype valves similar to the flight valves (Reference Rocket Research Corporation Report Number 171-23-3), and the other on a flight valve mounted on a SUBLEX Respin Rocket System (Reference Rocket Research Corporation Report Number 171-23-6).

2.4.3.1 Prototype Valve Testing

Three Eckel AF 42-562 valves were mounted on a SUBLEX filled propellant tank placed inside an environmental shroud which was located inside a vacuum chamber. A line was routed from the valve outside the chamber to a vacuum pump. Placed in the line were a choking orifice to throttle propellant flow and a manometer to monitor valve opening and leakage rate. The leakage rate was determined by measuring the change in pressure in a closed system of known volume between the valve and the manometer board over a period of time and normalizing to standard temperature and pressure. A schematic diagram of this apparatus is shown in Figure 9.

Each valve tested was operated on a different duty cycle. The first was cycled approximately every five days, the second approximately every ten days, and the third approximately every 15 days. Each cycle consisted of a leakage check and a five minute duration valve pulse, followed by another leakage check. During the test period, the environmental temperature was cycled randomly between 40°F and 100°F.

The results of the valve life test have increased the valve reliability confidence level. No valve failure, either opened or closed, occurred during the total test period of 45 days. The leakage rates were low, ranging from .01 to 1.9 cc/hr, with the average rate being 0.6 cc/hr. It was found that the greatest leakage occurred during the first 15 to 30 minutes; after that time the leakage rate essentially dropped to zero. Two reasons are believed to contribute to this effect. First, a small amount of SUBLEX vapor is trapped in the lines after a valve pulse, and will exert a small pressure when the leak check system is closed. Second, it is believed that a very small leak through the valve may be caused by micron size subliming solid particles on the valve seat, which would gradually sublime away allowing the plunger to seat properly and thereby stop leakage.



OPERATING ENVIRONMENT TEST

SCHEMATIC DIAGRAM

FIGURE 9

The indication of the valve opening was determined by monitoring line pressure upstream of the orifice. In most cases there was an immediate response and rapid rise in line pressure, indicating proper valve operation. There were four cases, however, when upon applying power to the valve the line pressure remained unchanged for approximately five seconds, then slowly rose to steady state pressure in from two to three minutes. This is believed due to a small layer of recondensed propellant covering the valve inlet, which, upon valve activation, breaks loose and sublimes away, slowly clearing the flow area.

2.4.3.2 Flight Valve Testing

A SUBLEX Respin Rocket System, consisting of a prototype propellant tank, one pound of SUBLEX propellant, a filter assembly, a flight solenoid valve, a valve heater, and an outlet manifold was mounted on a simulated OV2-1 satellite structural mock-up. The mock-up was then placed within the environmental shroud and the system arranged as described in paragraph 2.4.3.1. (See Figure 9.)

During the life test, the SUBLEX Respin Rocket System was subjected to the most severe thermal conditions expected on the OV2-1 satellite. The environmental temperature was cycled randomly between 40°F and 100°F, and because there was no additional source of heat input, the SUBLEX Respin Rocket System temperature followed suit. The propellant valve was operated for five minutes at a time at two week intervals for a period of eight weeks. Leak checks were taken before and after each valve cycle. The valve heater was not installed until the fifth week; therefore no outside heat was supplied to the valve during the first half of the test period. At the end of the eight week period, the test was extended to include a six week storage period, during which time no valve cycling took place, the valve heater was not on, and no temperature cycling other than normal environmental changes occurred.

During the initial eight weeks of the life test, the system operated satisfactorily with little or no leakage through the valve before or after each of the five actuations. The maximum leakage was .24 cc/hr.

At the end of the six week storage period, the initial attempt to cycle the valve resulted in no indication of line pressure, therefore indicating either complete plugging in the valve or a seizure of the valve poppet. After ten valve actuations the valve opened satisfactorily, as indicated by a line pressure rise. Since there was no power supplied to the valve heater during the storage period, it is conceivable that recondensation occurred around the valve seat, either temporarily seizing the valve poppet or plugging the valve inlet, but broke loose upon repeated cycling (possibly due to heat generated by valve current).

Although it is undesirable for a valve to seize or plug as described above, the occurrence is significant for two reasons. First, since there were no valve failures during the period when the valve heater was on, it indicates that the heater is effective in preventing recondensation in the valve area. Second, the valve did eventually open satisfactorily, proving that the valve may operate even though recondensation may temporarily plug or seize it. Thus temporary valve seizure is not likely to result in a system failure.

In an attempt to verify continued proper operation of the valve without mishap such as plugging or seizing, the life test unit was again subjected to a storage period, this time of three weeks duration. Also, the valve heater was turned on six hours prior to valve actuation. The results were as expected, showing immediate line pressure rise upon valve actuation, 1.38 cc/hr precycle leakage and a .32 cc/hr postcycle leakage.

2.4.4 Performance Tests

2.4.4.1 Thrust Measurement

The thrust generated by the OV2-1 SUBLEX Respin Rocket System was measured using the Rocket Research Corporation developed Compound Pendulum Balance. The results agree closely with the predicted thrust values, assuming isentropic flow of a perfect gas through the nozzles. The predicted thrust versus nozzle pressure curve (see Figure 10) was calculated using the equation:

$$F = C_f A_n (2 P_n)$$

where:

F = vacuum thrust, lbf

C_f = maximum theoretical thrust coefficient = 1.795

VACUUM THRUST VS
NOZZLE PRESSURE

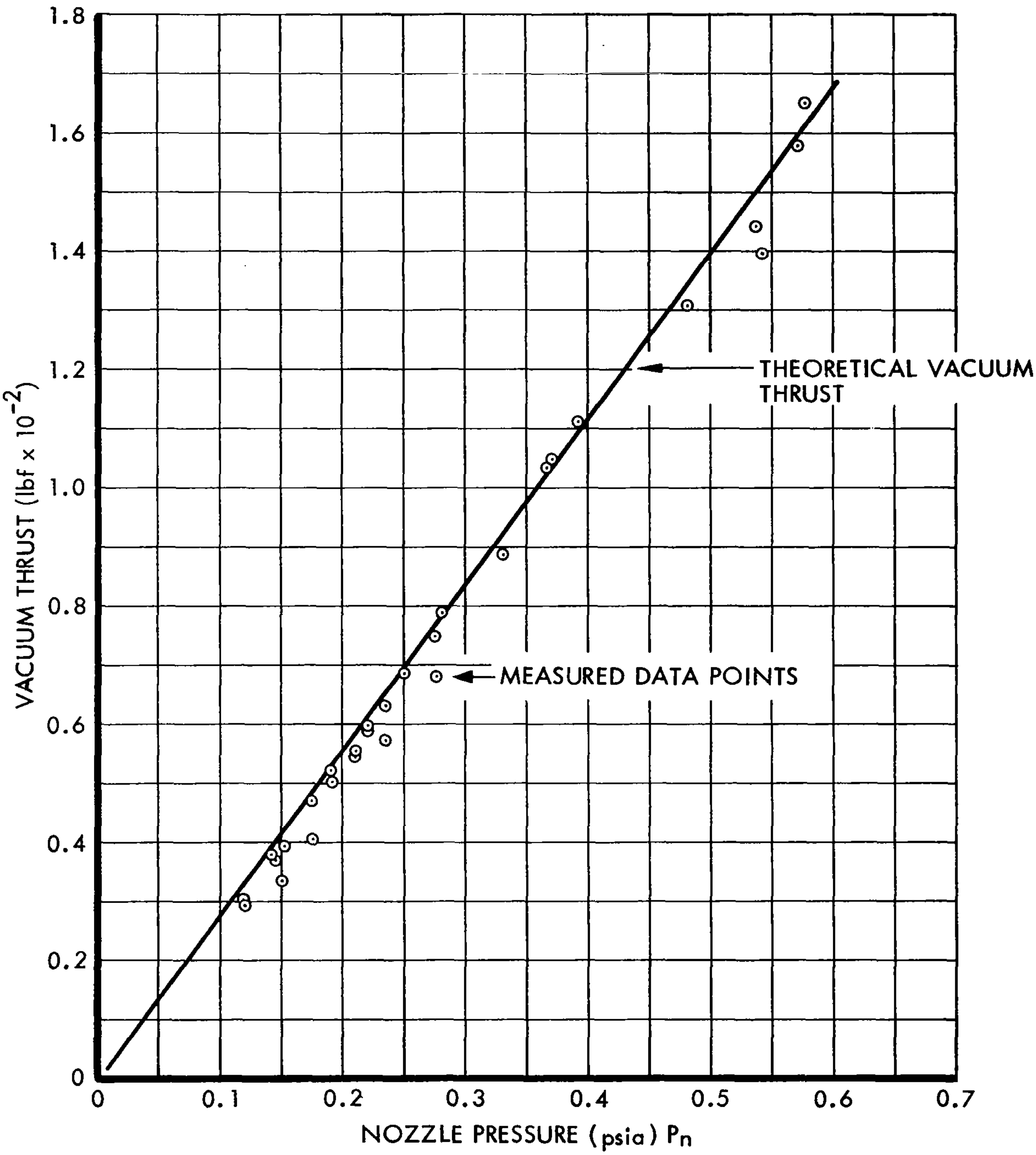


FIGURE 10

A_n = nozzle throat area of one nozzle, in^2

P_n = nozzle pressure, psia

therefore:

$$F = 1.795\,(2)\,(7.85 \times 10^{-3})\,P_n$$

$$F = 2.82 \times 10^{-2}\,P_n.$$

A schematic diagram of the test apparatus is shown in Figure 11. A SUBLEX filled propellant tank was placed on the top platform of the Compound Pendulum Balance. A line was extended from the tank to a nozzle, and a pressure transducer connected in the line to monitor line pressure. The entire apparatus is located inside a vacuum chamber. Upon valve actuation, the balance is deflected a certain distance which corresponds to a previously calibrated thrust. In this manner a continuous recording of thrust and pressure can be made with time. The results are shown in Figure 10.

2.4.4.2 Flow Rate Measurement

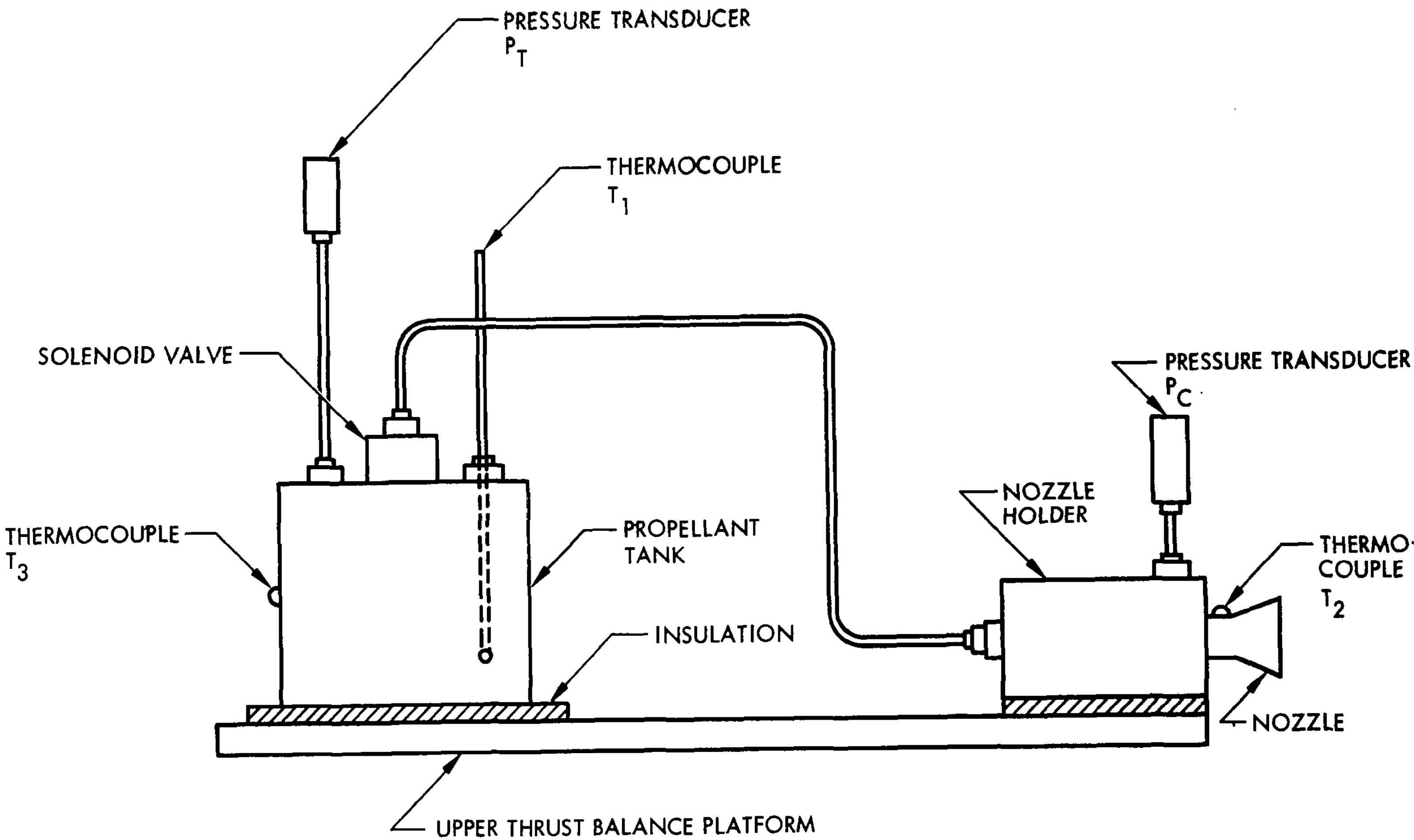
The flow rate versus nozzle chamber pressure relationship for the OV2-1 SUBLEX Respin Rocket System was measured. The results agree closely with the calculated theoretical isentropic flow rate of a perfect gas expanding through a nozzle orifice. If the theoretical flow rate is calculated using the equation shown in paragraph 2.3.2.3,

then

$$\dot{w} = 3.6 \times 10^{-4} \frac{P_t}{\sqrt{T_o}}$$

The flow rate versus nozzle pressure relationship is calculated in the same manner. The OV2-1 SUBLEX Respin Rocket System has two nozzles, therefore:

$$\frac{\dot{w}}{2} = 3.82 \times 10^{-3} \frac{P_n}{\sqrt{T_o}}$$



SCHEMATIC DIAGRAM
THRUST MEASUREMENT TEST

FIGURE 11

Flow rate was measured by determining the amount of mass lost from a plenum tank over small increments of time. A schematic diagram of the apparatus is shown in Figure 12. The system operates as described in the following paragraph.

The entire system up to V_t is first evacuated. Then, with V_2 closed and V_1 opened, vapor from the propellant tank is allowed to fill the plenum tanks to the desired pressure. V_1 is then closed and V_2 opened, allowing propellant to flow out of the plenum tank T_2 and exhaust out through the nozzles. During this operation the transducer records ΔP versus time. Flow rate can then be determined from the perfect gas law:

$$P_1 V = m_1 \frac{R}{M} T$$

$$P_2 V = m_2 \frac{R}{M} T$$

since

$$m_1 - m_2 = \bar{w} t = \frac{(P_1 - P_2)\ VM}{RT}$$

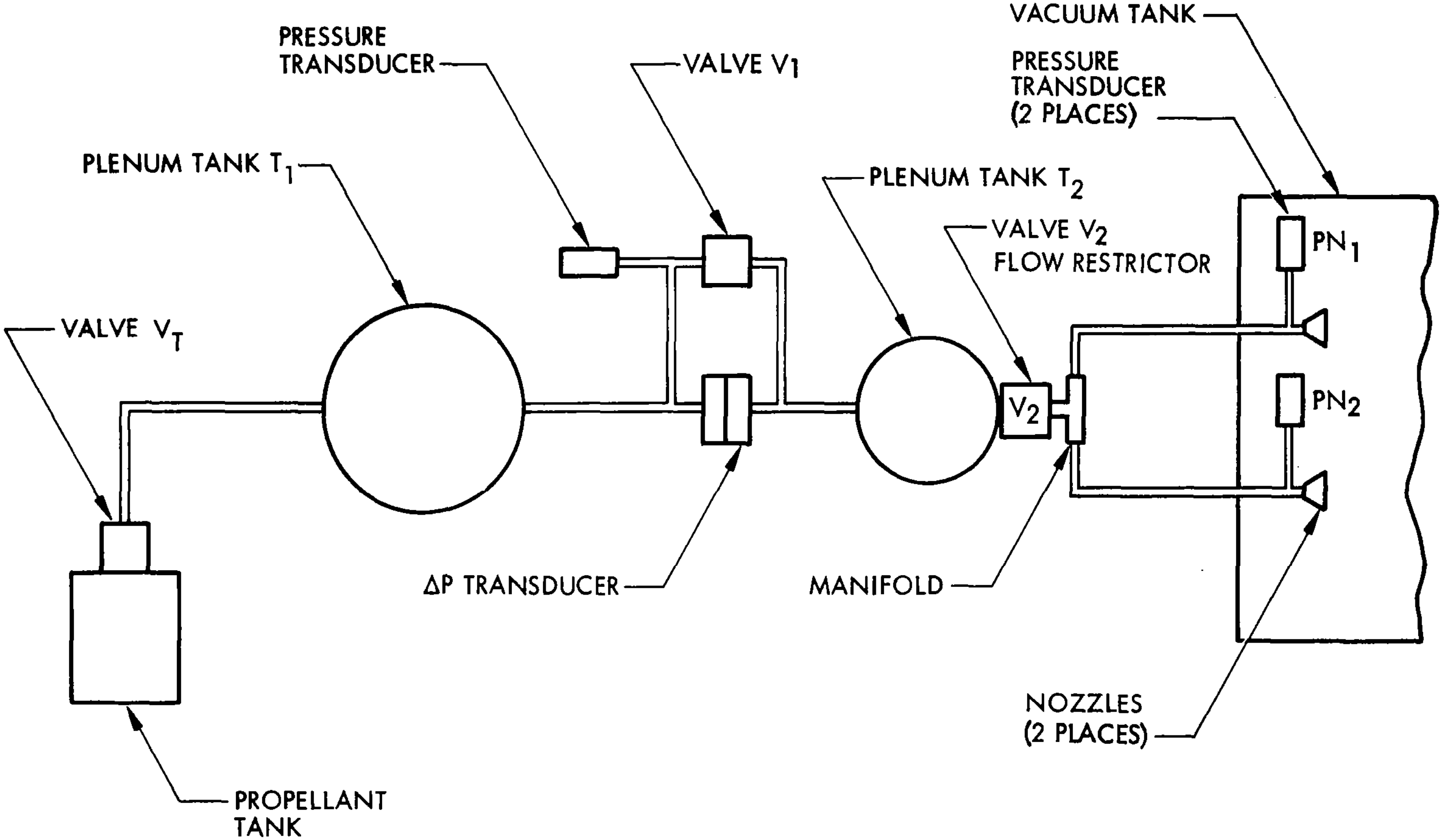
then

$$\bar{w} = \frac{(P_1 - P_2)\ V}{RTt}$$

where:

$P_1 - P_2$	=	change in plenum pressure, psia
$\bar{w}$	=	average flow rate, lbm/sec
V	=	system volume, in^3
R	=	gas constant, ft-lb/lb-mole - °R
M	=	average molecular weight, lb/lb-mole
t	=	time, seconds
T	=	absolute temperature, °R (assumed constant).

Nozzle pressure was also measured with time by means of a pressure transducer. Flow rate versus nozzle pressure could then be plotted and is shown in Figure 13.



SCHEMATIC DIAGRAM
OV2-1 SRRS FLOW RATE MEASURING SYSTEM

FIGURE 12

OV2-1 SUBLEX RESPIN ROCKET
CHAMBER PRESSURE PER NOZZLE
VS FLOW RATE

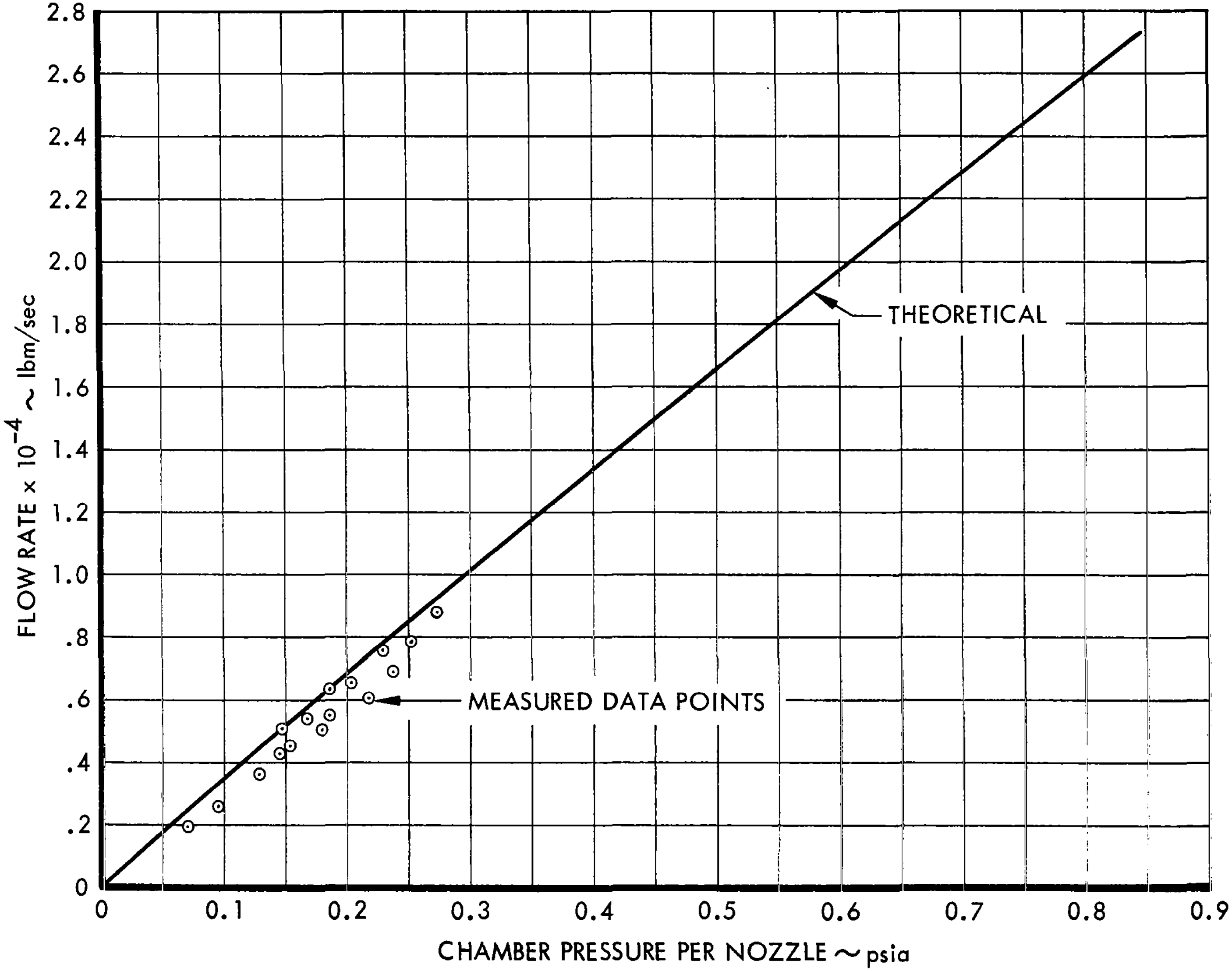


FIGURE 13

2.4.4.3 Specific Impulse Measurement

From the preceding data on thrust and flow rate, it is possible to obtain measured specific impulse (I_s) values for the OV2-1 SUBLEX Respin Rocket System and compare them with the calculated theoretical I_s. The theoretical I_s is calculated by the following equation:

$$I_s = \frac{C_f c^*}{g} = 84.9 \text{ sec}$$

where:

C_f = vacuum thrust coefficient = 1.795

c^* = characteristic velocity = 1,523 ft/sec

g = gravitational constant = 32.2 ft/sec.

Measured values of I_s can be determined in two ways: First, by dividing the measured thrust by the corresponding measured flow rate, and second, by determining actual C_f:

$$C_f \text{ meas} = \frac{F}{P_c A_t}$$

where:

F = measured thrust

P_c = measured pressure

A_t = throat area

The results determined by these two methods vary considerably. I_s values determined by the first method remain essentially constant over the entire pressure range, with the average value falling at 83 sec. Values determined by the second method vary from 85 sec to 70 sec, with an average I_s equal to about 75 sec.

Since these tests were run, several additional measurements of specific impulse have been made. The results indicate that for a system operating like the OV2-1 SUBLEX Respin Rocket System, the average specific impulse will be very close to 75 sec.

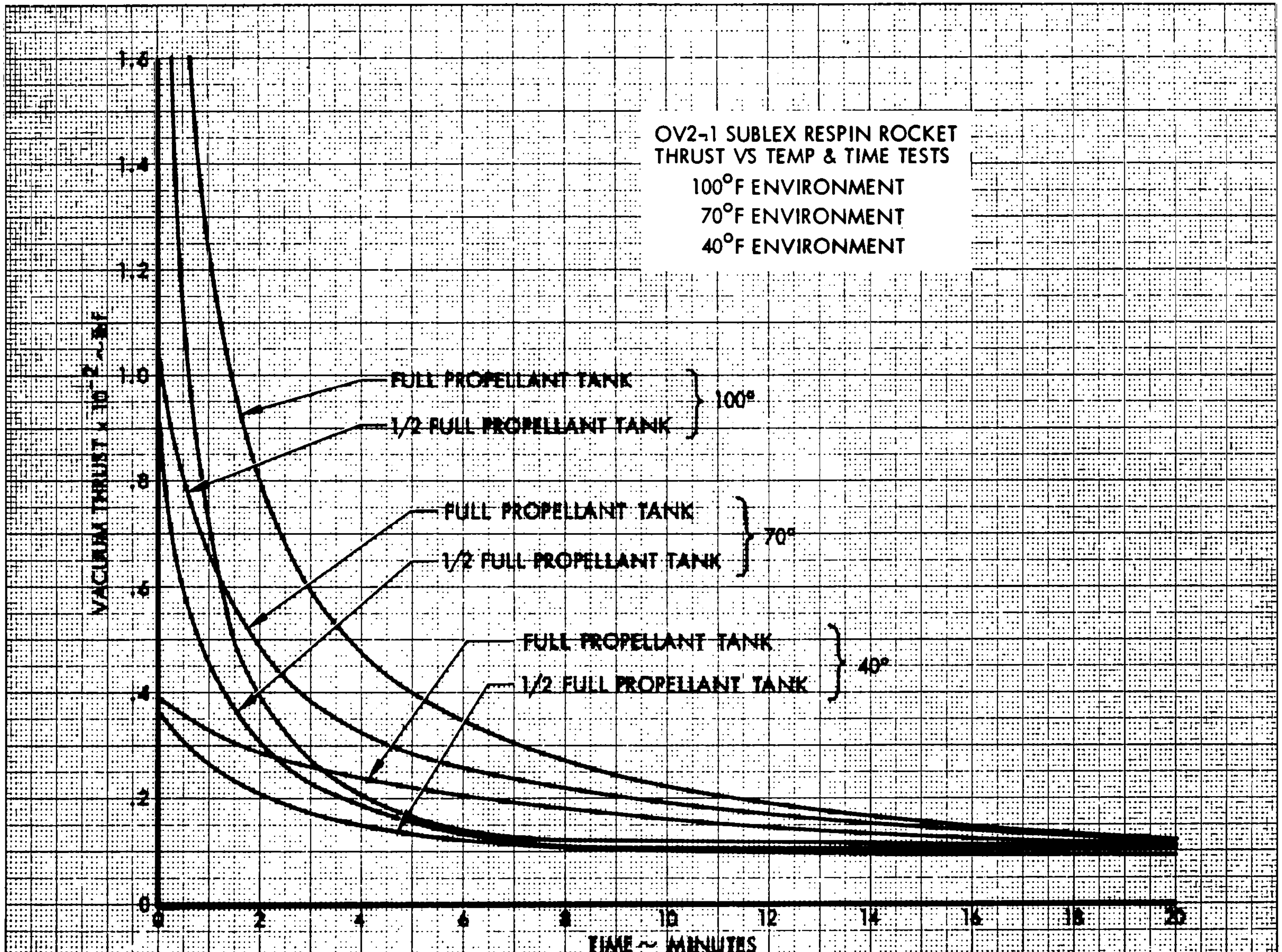
(See Section 4.0 on the nozzle optimization tests for further discussion on this point.) It is believed that the high I_s values determined by the first method are due to high corresponding thrust values. During the thrust measurement tests, it was necessary to add large correction factors to the measured thrust values to achieve vacuum thrust values, due to high nozzle back pressures. It is believed that errors in measurement of back pressures may have resulted in abnormally high correction factors and hence C_f values. Since that time, further tests have been completed, yielding slightly lower thrust performance data, which is felt to be more representative of actual delivered I_s from this system.

2.4.4.4 Thrust Versus Temperature and Time

Since there was no thrust control mechanism on the OV2-1 SUBLEX Respin Rocket System, it was necessary to determine the relationship between thrust and time for varying temperatures and varying propellant loads. This was accomplished by using apparatus similar to that shown in Figure 9. A SUBLEX Respin Rocket System was mounted inside a vacuum chamber, with two lines extended out to two nozzles also located within the vacuum chamber. Nozzle pressure was recorded immediately upstream of the nozzle by pressure transducers. The environmental shroud was maintained at temperatures of 100°F, 70°F, and 40°F. Where equilibrium temperature was reached, the valve was opened and pressure recorded with time for a period of twenty minutes. This procedure was repeated for a full and a half-full tank configuration. Pressure was then correlated to thrust from the information obtained in the thrust measurement tests. The results are shown graphically in Figure 14. It should be noted that in each case the thrust levels out at approximately 1×10^{-3} lbf.

2.4.4.5 Valve Heater

Valve heater tests were conducted to determine the valve heater power requirements. Two valve heaters similar to that described in Paragraph 2.3.1.13 were tested. One required 1/4 watt of power at 28VDC, while the second required 1/2 watt at 28 VDC. The tests were conducted by varying the propellant tank temperature while measuring the valve and propellant temperature. The 1/4 watt heater was not sufficient to maintain this valve temperature above the propellant temperature under all transient conditions. Therefore, slight recondensation could occur in the valve. However,



OV2-1 SUBLEX RESPIN ROCKET
THRUST VS TEMP & TIME TESTS
100°F ENVIRONMENT
70°F ENVIRONMENT
40°F ENVIRONMENT
FULL PROPELLANT TANK
1/2 FULL PROPELLANT TANK
100°
FULL PROPELLANT TANK
1/2 FULL PROPELLANT TANK
70°
FULL PROPELLANT TANK
1/2 FULL PROPELLANT TANK
40°
VACUUM THRUST x 10^-2 ~ LBF
1.6
1.4
1.2
1.0
.8
.6
.4
.2
0
0
2
4
6
8
10
12
14
16
18
20
TIME ~ MINUTES

FIGURE 14

the 1/2 watt heaters maintained the valve temperature a degree or two above the propellant temperature under all environmental conditions.

2.4.5 Qualification and Acceptance Testing

2.4.5.1 Prequalification Testing

Prior to the initiation of the qualification testing, a prototype propellant tank assembly was subjected to a launch environment test. This test consisted of the following:

a. A sustained acceleration test five minutes in each direction along the three coordinate missile axes as follows:

Foward Longitudinal	5.3 g
Reverse Longitudinal	None
Plus and Minus Pitch and Yaw	1.0 g

b. A random vibration test applied along each of the three mutually perpendicular axes to an 18.6 g rms level for 90 seconds in each axis.

c. A shock test performed three times in each direction along each of the three major orthogonal axes conforming to the following shock response spectrum:

Pulse Mode	Duration	Level
Saw Tooth	1.5 sec	50 g's peak

Following the launch environment tests, the SUBLEX Respin Rocket System was functionally checked for operation and visually checked for damage. No damage was incurred, and operation of the system was normal.

2.4.5.2 Qualification Testing

The SUBLEX Respin Rocket System (SRRS) qualification unit was subjected to qualification tests as specified in test specifications RRC-TS-0001 and NSL-64-211A. These tests were designed to demonstrate the ability of the SUBLEX Respin Rocket

System to withstand satisfactorily the powered flight environment of the Titan III-A (note that actual flight was on the Titan III-C Booster) Booster and to successfully operate under the space environmental conditions expected in the OV2-1 satellite after booster separation. Safety factors in amplitude and/or time were incorporated in the above qualification test criteria for proof-test of the qualification unit, which was a prototype system identical to the flight unit. The SUBLEX Respin Rocket System was satisfactorily subjected to these tests, as described below, and was therefore considered qualified for flight. (See Rocket Research Corporation Report 171-23-4 for additional information.)

The SUBLEX Respin Rocket System consists of two major subsystems: The propellant tank assembly and instrumentation subsystems. Qualification tests were conducted on the subsystem level only. No qualification testing was conducted on the component level, with the exception of a functional acceptance test on the solenoid valve.

The qualification tests consisted of the following:

a. Powered flight tests, which consisted of acceleration, vibration, and shock tests as described in paragraph 2.4.5.1. No damage was incurred.

b. Thermo-vacuum tests, in which the SUBLEX Respin Rocket System operated in a vacuum for a period of two weeks. The environmental temperature was cycled randomly. Operation of the SUBLEX Respin Rocket System was satisfactory.

c. Proof pressure test, wherein the propellant tank was pressurized to 90 psia three separate times and then inspected for damage. No damage was incurred.

d. Final System Checkout, wherein X-rays were taken and then the system disassembled and inspected. No signs of internal damage were noticed.

2.4.5.3 Acceptance Testing

The flight OV2-1 SUBLEX Respin Rocket System was subjected to the acceptance tests as described in paragraph 3.5.1 of the test specification RRC-TS-0001 and in section 4 of NSL-64-211A. The acceptance tests were designed to verify, for the flight unit, what was already extensively determined during qualification testing described in Rocket Research Corporation Report 171-23-4, i.e., the ability of the tank assembly to withstand successfully the powered flight environment of the Titan III-A booster and operate successfully under the space environmental conditions expected in the OV2-1 satellite during the duration of its mission. No visible damage or failure, either structural or functional, occurred during any phase of the acceptance testing. Therefore the SUBLEX Respin Rocket System was considered satisfactorily acceptance tested and qualified for flight. Delivery of the unit to Northrop Space Laboratories took place on schedule on January 7, 1965.

Acceptance testing of the SUBLEX Respin Rocket System consisted of the following:

a. A random vibration test applied along each of the three mutually perpendicular axes at a level of 13 g's rms overall for 30 seconds. No damage was incurred.

b. A proof pressure test wherein the propellant tank was pressurized to 60 psia. No damage was incurred.

2.5 OV2-1 SUBLEX Respin Rocket System Performance Curves

In order to be able to determine the performance of the OV2-1 SUBLEX Respin Rocket System during flight, several performance curves had to be compiled. The following curves were experimentally determined:

a. Pressure Transducer Calibration - Output Voltage versus Line Pressure (see Figure 15).

b. Line Pressure versus Nozzle Pressure (see Figure 16).

c. Nozzle Pressure versus Thrust (see Figure 11).

d. Nozzle Pressure versus Flow Rate (see Figure 13).

e. Thermistor Calibration - Output Voltage versus Temperature (see Figure 17).

f. Valve Pickoff Voltage versus Input Voltage (see Figure 18).

g. Input Voltage versus Valve Heater Power (see Figure 19).

h. Temperature Transducer Output Voltage versus Temperature (see Figure 20).

OV2-1 SUBLEX RESPIN ROCKET
PRESSURE CALIBRATION
OUTPUT VOLTAGE VS LINE PRESSURE

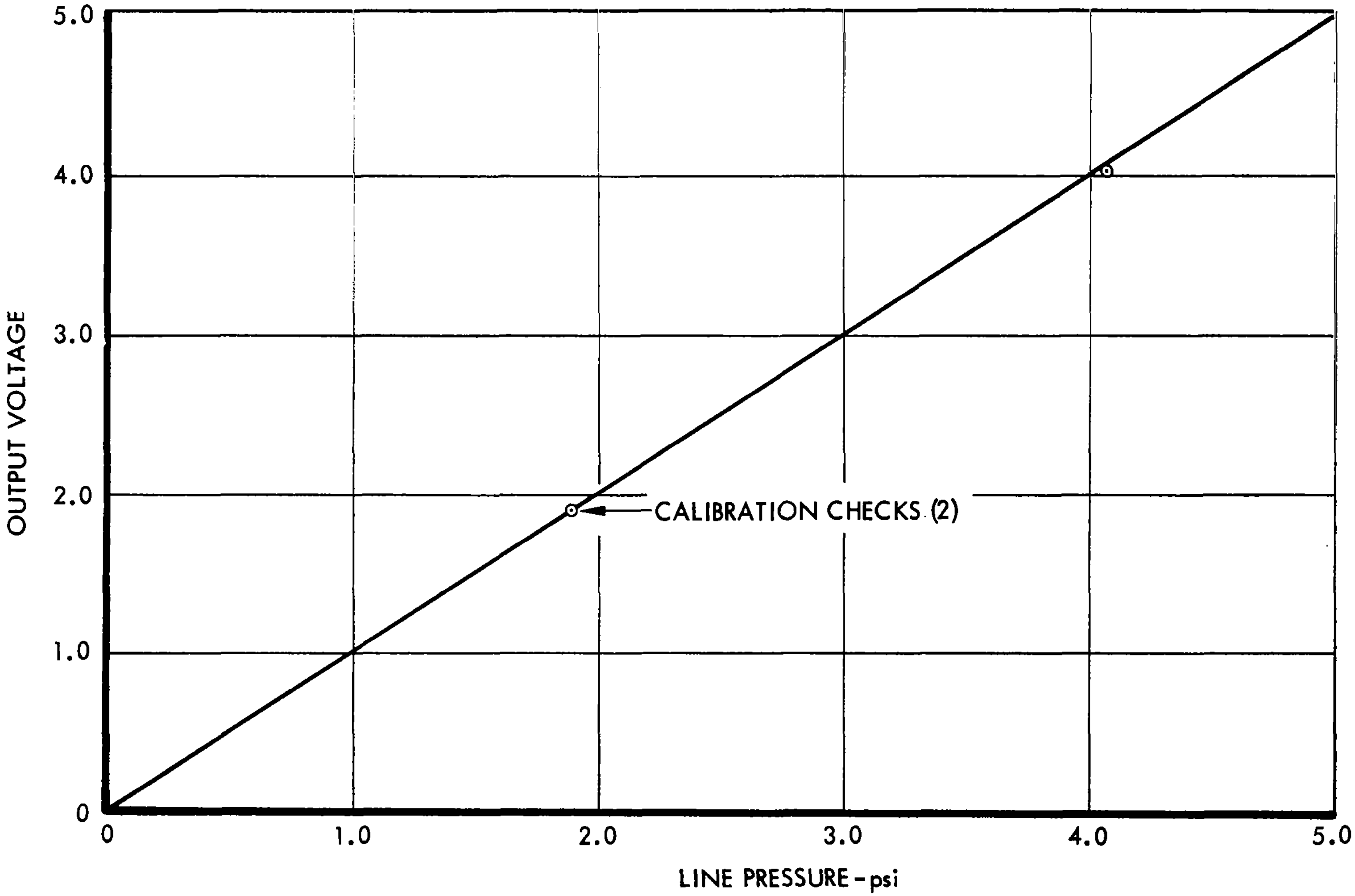


FIGURE 15

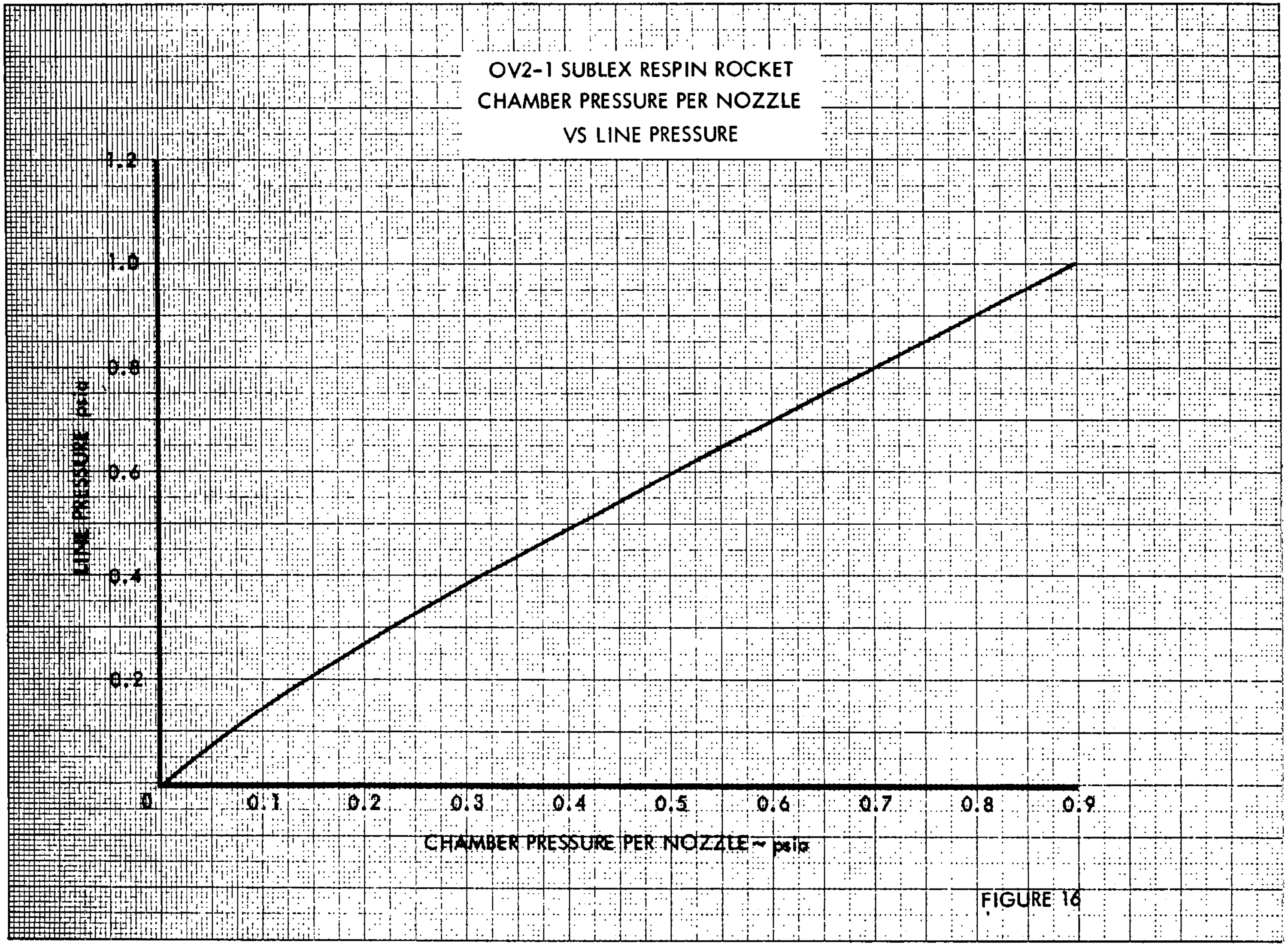
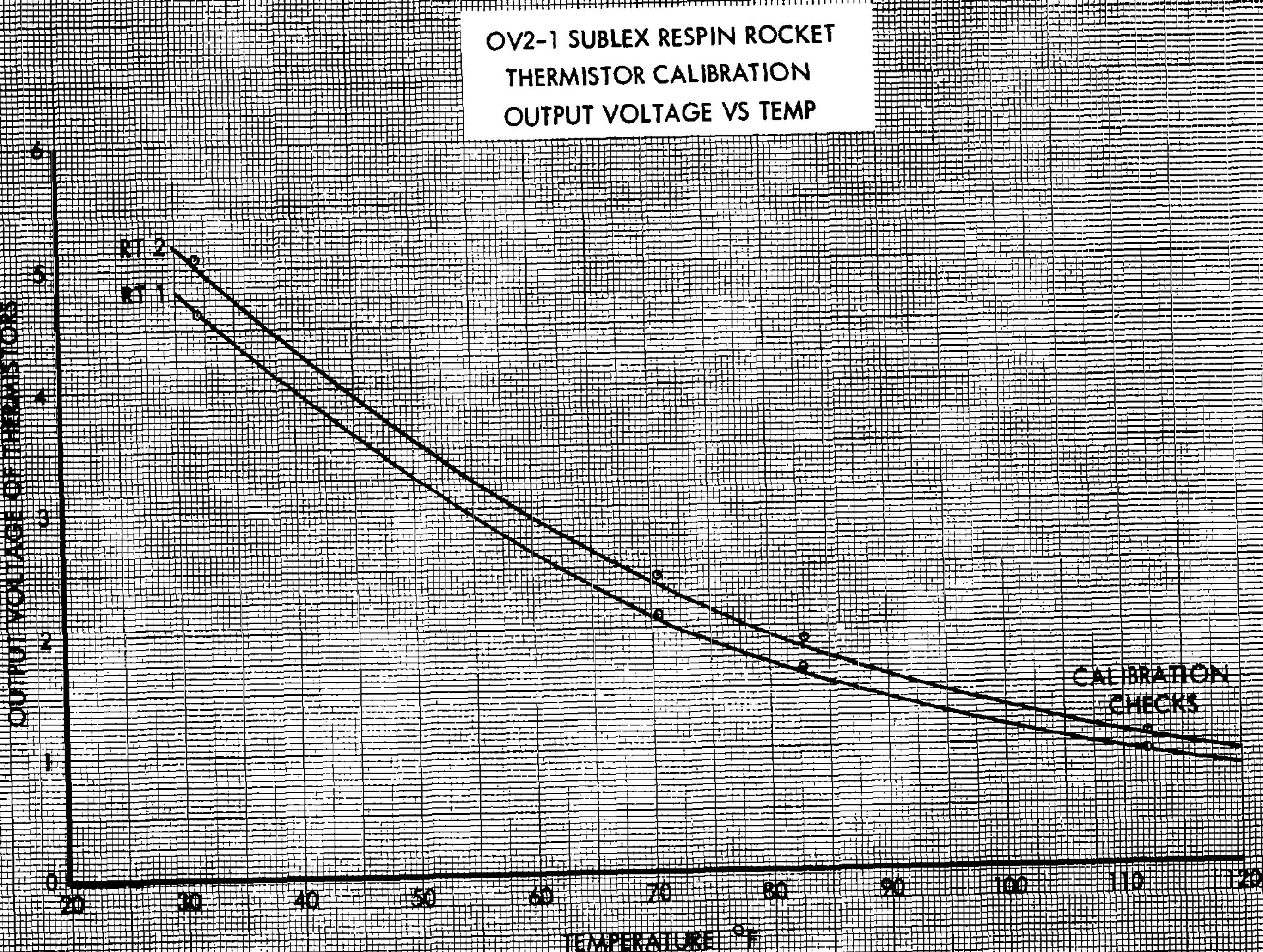

OV2-1 SUBLEX RESPIN ROCKET
CHAMBER PRESSURE PER NOZZLE
VS LINE PRESSURE
1.2
1.0
0.8
0.6
0.4
0.2
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
LINE PRESSURE ~ psia
CHAMBER PRESSURE PER NOZZLE ~ psia

FIGURE 16


OV2-1 SUBLEX RESPIN ROCKET
THERMISTOR CALIBRATION
OUTPUT VOLTAGE VS TEMP
RT 2
RT 1
CALIBRATION
CHECKS
OUTPUT VOLTAGE OF THERMISTORS
TEMPERATURE °F
0
1
2
3
4
5
6
20
30
40
50
60
70
80
90
100
110
120
FIGURE 17

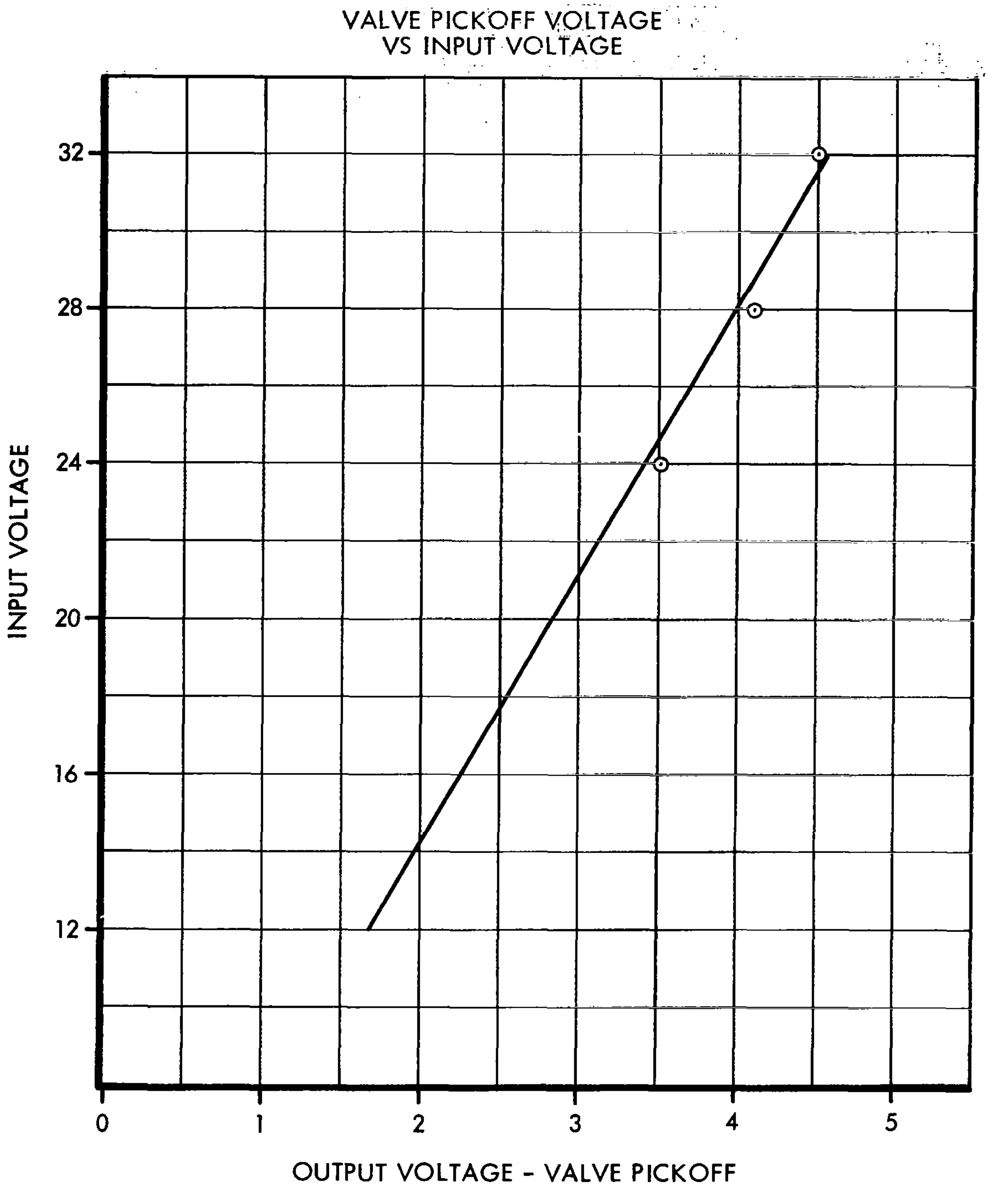

VALVE PICKOFF VOLTAGE
VS INPUT VOLTAGE
INPUT VOLTAGE
32
28
24
20
16
12
0
1
2
3
4
5
OUTPUT VOLTAGE - VALVE PICKOFF

FIGURE 18

VALVE HEATER POWER
VS VOLTAGE INPUT

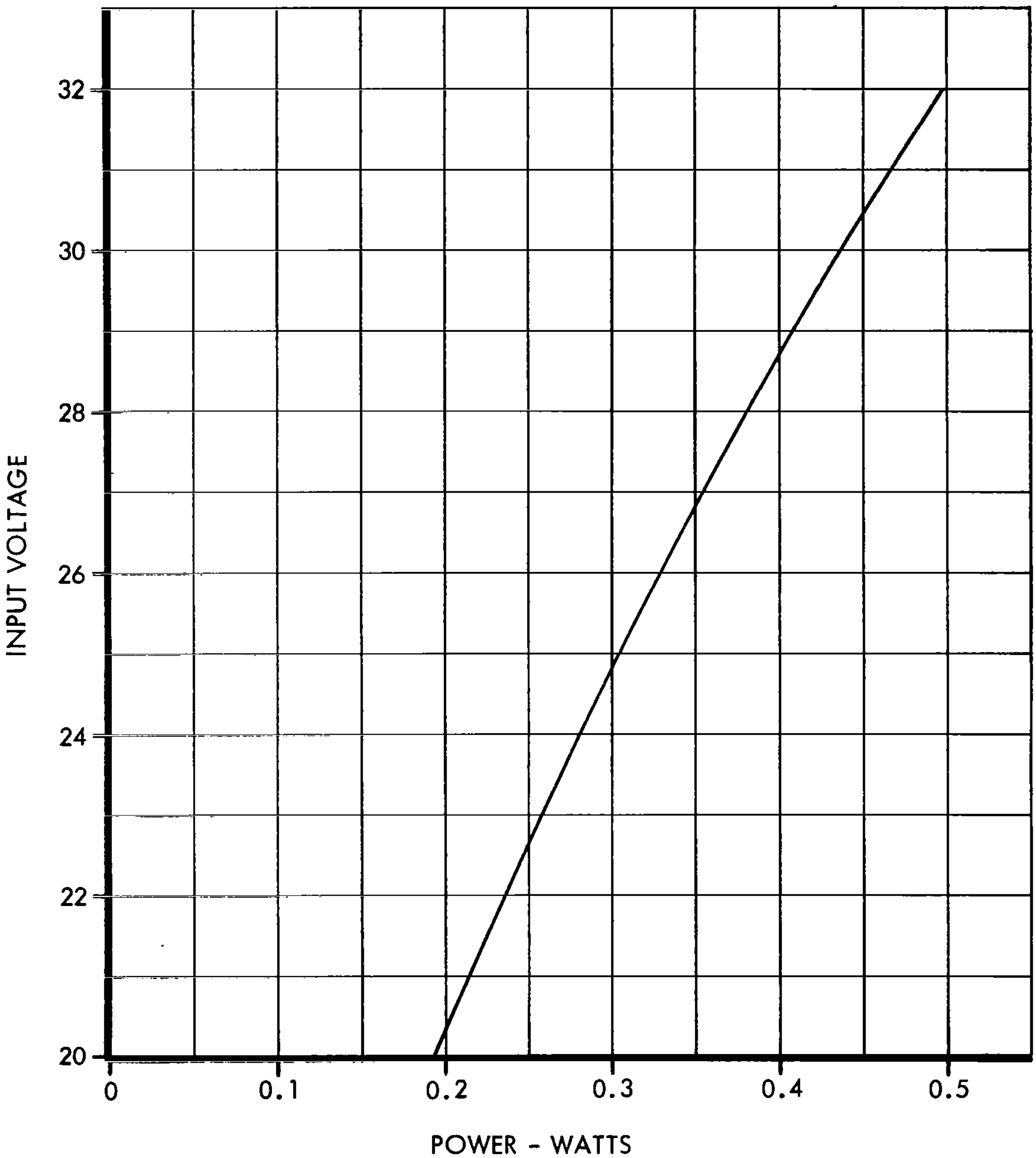


FIGURE 19

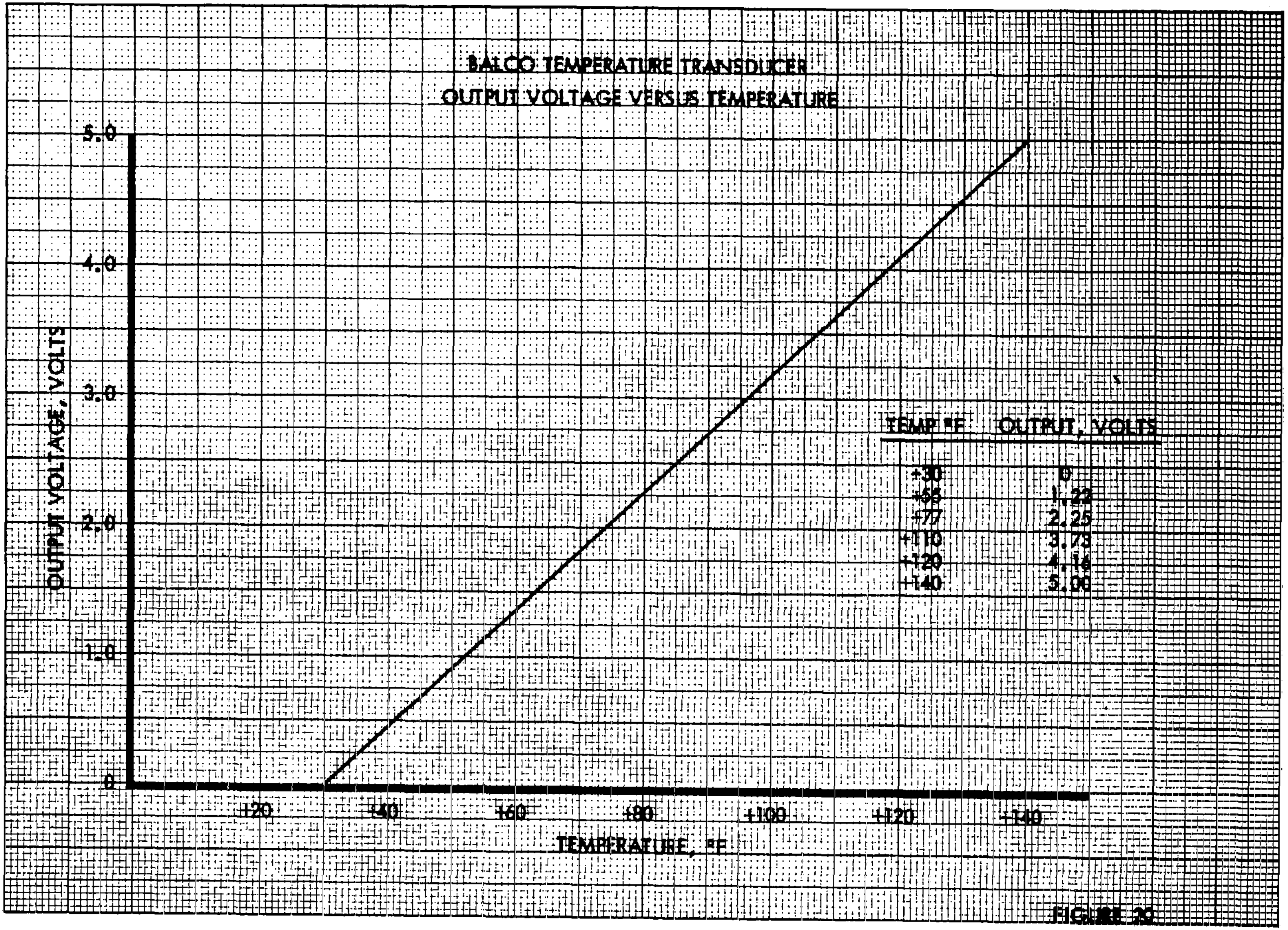

BALCO TEMPERATURE TRANSDUCER
OUTPUT VOLTAGE VERSUS TEMPERATURE
OUTPUT VOLTAGE, VOLTS
5.0
4.0
3.0
2.0
1.0
0
+20
+40
+60
+80
+100
+120
+140
TEMPERATURE, °F
TEMP °F OUTPUT, VOLTS
+30 0
+55 1.22
+77 2.25
+110 3.73
+120 4.16
+140 5.00

FIGURE 22

2.6 Status

The complete SUBLEX Respin Rocket System package was delivered to Northrop Space Laboratories on January 7, 1965, and buy-off was completed on January 13, 1965. The system was installed on the OV2-1 satellite during the week of January 20 to January 24, 1965. The Air Force buy-off of the satellite took place on March 29, 1965, with shipment to Cape Kennedy following immediately. During the period between March 29, and August 15, 1965, the satellite, with the SUBLEX Respin Rocket System installed, was in storage in a clean room facility. Prelaunch operations were begun on August 16, 1965. Launch occurred from Cape Kennedy on October 15, 1965. However, the OV2-1 satellite was not ejected into orbit, due to a failure of the transtage booster. Apparently the transtage, with the OV2-1 satellite attached, began tumbling and disintegrated in space.

3.0 RELIABILITY STUDY

3.1 Conclusions

The use of a solenoid valve for on-off thrust control is the major source of inherent unreliability in a typical subliming solid system. Test data collected to date have not identified a predominant failure mode for the valve models investigated in this report. However, additional testing would be required to demonstrate the reliability of a valve at a reasonable confidence level.

3.2 Reliability Prediction Based on Generic Failure Rates

An analysis of a typical subliming solid system similar to the OV2-1 SUBLEX Respin System (see Figure 1) was based on industry generic component failure rate data. It provides a quantitative indication of potential reliability problem areas for mechanical and electromechanical parts used within the system. Table I lists the parts and generic failure rate values that make up the system and provide the generic failure rate of the system used in Table II. As indicated, the on-off solenoid valve is expected to be the major contributor to system unreliability.

Experience has shown that the control of propellant migration and condensation in flow passages is another major design problem area that must be considered in the design of a subliming solid system. A discussion of the problem and design considerations is included in paragraph 1.10 of the Final Report (Contract NAS 5-3599), and in paragraph 2.4.2 of this Phase II report.

The subliming solid system reliability is limited primarily by the valve reliability. Improving the valve reliability to one tenth of its generic failure rate would reduce the system failure rate to 3.01×10^{-6} failures per hour from the 14.01×10^{-6} failures per hour estimated. The corresponding improvement in the system reliability depends upon the mission time and the level of major stresses occurring. For comparison purposes a hypothetical three-year mission profile was assumed, as shown in Table III. Under these conditions the system reliability would be improved from .953 to .989; i.e., it would eliminate 36 mission failures out of every 1,000 missions.

TABLE I

TYPICAL SUBLIMING SOLID SYSTEM

Parts List and Generic Failure Rate Values (GF_r)

ITEM	GF_r* (Failures Per Million Hours)
1 - Tee Fitting Outlet	.10
1 - Propellant Tank	.15
1 - Flow Restrictor	.08
1 - Fitting	.10
1 - Connector (3 pin @ .035/pin)	.11
4 - Silicon Diodes (@ .2)	.80
2 - Film Resistors	.06
2 - Heater Elements (@ .02)	.04
1 - Solenoid on-off valve	11.00
1 - Connector (2 pins @ .035/pin)	.07
Propellant	1.50**
	14.01

NOTE:

* Generic Failure Rate Values are from the AVCO Failure Rate Tables dated April 1962, except when followed by an asterisk.

** An arbitrary estimate was included for the SUBLEX propellant.

TABLE II

FAILURE MODE AND EFFECT ANALYSES

		FAILURE		EFFECT ON SUBLIMING SOLID REACTION CONTROL SYSTEM
COMPONENT	FUNCTION	MODE	CAUSE	
On-off control valve	To start and stop the flow of sublimate to the expansion nozzle in a subliming solid system	1. Fails "open" when commanded closed	a. Binding	a. Continued high thrust when commanded off
			b. Particle contamination between valve seat and mating port.	b. Exhaust propellant prior to end of mission
		2. Fails "closed" when commanded open	a. Loss of operating power	a. Loss of thrust capability
			b. Electrical short	
			c. Electrical open	
			d. Frozen in place	
			e. Cold welding in vacuum	
		3. Gas flow leak when closed	a. Particle contamination	a. High off-leakage resulting in excessive thrust for off condition.
			b. Seat damage	b. Exhaust propellant prior to end of mission
			c. Seat erosion	
		4. Gas Leak Externally	a. Body rupture	a. Loss of thrust capability
			b. Seal leakage at joints	b. Contamination of spacecraft

TABLE III
RELIABILITY PREDICTION SUMMARY

Mission Profile		Estimate of Mission Failures			Estimate of Reliability
Event	Time (t) Hours	$\lambda = K_E \ \Sigma GF_r$ *		λt	Probability of Success
		K_E	Failures Per Hour	Failures Per Mission	$R_i = e^{-\lambda t}$
I. Generic Failure Ratio for System = 14.01×10^{-6} (includes valve = 11.0×10^{-6})					
1. Launch	.3	900	.012609	.003783	
2. Total power on operating time	500	1	.000014	.007000	
3. Coast and orbit	25,780	0.1	.0000014	.036092	
TOTAL				.046875	.953
II. Generic Failure Rate for System = 3.01×10^{-6} (includes valve = 1.1×10^{-6})					
1. Launch	.3	900	.002709	.00813	
2. Total power on operating time	500	1	.000003	.001500	
3. Coast and orbit	27,780	0.1	.0000003	.008334	
TOTAL				.010647	.989

3.3 Considerations in Improving Valve Reliability

A summary of the potential failure modes, causes, and effects is shown in Table II. A valve design must consider all of these potential failure modes to achieve a high inherent reliability. Manufacturing and handling controls must also consider them to avoid degrading the reliability, and a test program must consider them to maximize its effectiveness.

Paragraph 1.5 of the Final Report (NAS 5-3599) discusses valve requirements, configurations, and design considerations for the subliming solid system application.

3.4 Valve Test Experience in Subliming Solid Application

The use of valves with subliming solid systems has been demonstrated in tests conducted to date. Time/cycle test data have been accumulated as shown in Tables IV and V without a primary valve failure.

Table IV is a summary of tests conducted on the Coaxial Solenoid Valve P/N AF 42-562.

Eckel valve, P/N AF 42-562, is a six-watt coaxial solenoid valve. The demonstrated mean-time between failures (MTBF) at a 90% confidence level is equal to or greater than 15.1 hours. In comparison, the predicted MTBF using the AVCO Tables, failure rate data under orbit conditions is 91,000 hours. This valve may or may not have a reliability comparable to the predicted MTBF in a subliming solid system application. Test results did not establish an upper limit on MTBF. Additional testing would be required to identify any predominant failure modes that may exist, and to increase the demonstrated reliability.

Table V is a summary of tests conducted on the Coaxial Solenoid Valve, P/N AF 77C-A119.

Eckel valve, P/N AF 77C-A119, Rocket Research Corporation Drawing No. 30-1029, is a two-watt coaxial solenoid valve weighing 0.15 pounds with an orifice diameter of .05 inches. It is an all welded, stainless steel valve.

The accumulated test results are not sufficient to determine if these particular valve designs are as reliable as predicted by general industry data (i.e., 11.0×10^{-6} failures per hour), or as desired for a high reliability subliming system (i.e.,

TABLE IV

TIME/CYCLE TEST DATA SUMMARY

PART NAME Valve, Solenoid PART NO. AF42-562 VENDOR Eckel Valve Company

Date	Part Serial Number	Effluent: Sublex Gas	Effluent: N_2	Effluent: Air	Environment: Upstream Pressure 6.5 psia	Environment: Downstream Pressure Vacuum (hrs)	Environment: Downstream Pressure 14.7 psia	Power Applied	Temp. °F	No. of Cycles	Reliability Estimate Data: No. of Failures	Reliability Estimate Data: Hours	Reliability Estimate Data: K_E	Test Data Reference
9-17-64	35458	X			X	120		5 min.	50-70	1	0	.083	1	Report
9-22-64		X			X	120				1	0	.083	1	171-23-3
9-29-64		X			X	168				1	0 (2)	.083	1	
10-5-64		X			X	168				1	0	.083	1	
10-13-64		X			X	192				1	0	.083	1	
10-21-64		X			X	168				1	0	.083	1	
10-23-64	35458	X			X	48		5 min.	50-70	1	0 (2)	.083	1	
9-7-64	35460	X			X	8		5 min.	50-70	1	0	.083	1	
9-16-64		X			X	192				1	0	.083	1	
9-28-64		X			X	288				1	0	.083	1	
9-30-64		X			X	48				1	0	.083	1	
10-8-64		X			X	192				1	0	.083	1	
10-20-64		X			X	288				1	0	.083	1	
10-23-64		X			X	72		5 min.	50-70	1	0 (2)	.083	1	Report
3-25-65	35460	X			X		5 sec	5 sec.	70	1	0 (4)	.001	1	171-23-3
9-17-64	35459	X			X	8		5 min.	50-70	1	0	.083	1	
10-1-64		X			X	336				1	0 (2)	.083	1	
10-14-64		X			X	312				1	0	.083	1	
10-23-64	35459	X			X	264		5 min.	50-70	1	0 (2)	.083	1	
	35480 – 35476 (3)							33.33 hrs		100	0	33.33	1	
TOTAL	5 units							34.83		119	0	34.83		

Footnotes:

1. Calculated by using values from Handbook of Statistical Tables by D. W. Owen, Table 9.4, Page 262. The estimated reliability and corresponding confidence level is equal to, or greater than e-T/MTBF where T is hours of stress during mission.
2. Slow pressure rise indicating partial plugging of valve inlet by recondensation.
3. Operated in Flow Rate & Thrust Vs P_c Tests, approx. 50 tests ea. at 20 min./test, no failures.
4. In storage for approx. 5 months.

DEMONSTRATED MTBF VS CONFIDENCE LEVEL (1)

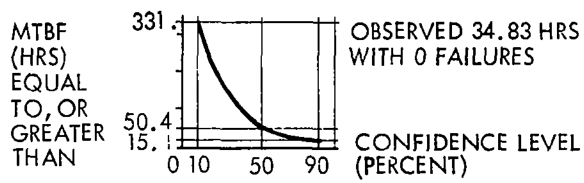


TABLE V

TIME/CYCLE TEST DATA SUMMARY

PART NAME Valve, Solenoid PART NO. AF77C-A119 VENDOR Eckel Valve Company

Date	Part Serial Number	Effluent: Sublex Gas	Effluent: N_2 sec.	Effluent: Air sec.	Environment: Upstream Pressure 6.5 psia	Environment: Downstream Pressure Vacuum (wks)	Environment: Downstream Pressure 14.7 psia	Power Applied	Temp. °F	No. of Cycles	Reliability Estimate Data: No. of Failures[1]	Reliability Estimate Data: Hours[2]	Reliability Estimate Data: K_E	Test Data Reference
11-3-64	001	X	60	10	X	X		70 sec.	70	102	0	.019		
11-3-64	002								70	102	0	.019		
11-16-64	↑	X			X	X			40-100	0	0	.083		RRC Report 171-23-6
11-23-64		X			X	X		5 min.	↑	1	0(4)	.083		↑
12-8-64		X			X	X		5 min.		1	0	.083		
12-16-64		X			X	X		5 min.		1	0	.083		
12-29-64		X			X	X		5 min.		1	0	.083		
1-12-65		X			X	X		5 min.		1	0	.083		
3-2-65		X			X	6			↓	10	0(3)			↓
3-24-65	↓	X			X	2			40-100	1	0			RRC Report
7-9-65	002	X			X	16			70	1	0			171-23-6
11-3-64	003		60	10				70 sec.	70	102	0	.019		
11-64/1-65	003(5)													
11-3-64	004		60	10				70 sec.	70	102	0	.019		

Footnotes:

1. Failure Definition: Valve fails to open or close upon command, or exhibits 10cc/hr @ approximately 7 psia leakage when closed.
2. No. of hours with voltage, acceleration, vibration, or temperatures greater than 100°F applied.
3. Valve did not indicate open until tenth pulse.
4. Slow pressure rise indicating partial plugging of valve inlet by recondensation.
5. Operated in test without failure, approximately 50 tests @ 20 minutes per test.

3.0×10^{-6} failures per hour or less). To demonstrate a failure rate of 3.0×10^{-6} failures per hour at a 90% confidence level requires a total of 767,000 test hours under mission usage conditions with zero failures. This is equivalent to operating 88 valves for one year.

3.5 Test Program for Demonstrating Valve Reliability

Many hours of test and/or usage operation are required to demonstrate that a part has achieved a high mean-time between failures (i.e., low failure rate) with an associated high confidence level.

Figure 21 illustrates an operating characteristic curve for three different demonstration plans. The symbols used are:

n = the number of sample units

c = the number of failures that occurred during test.

The curves represent the corresponding confidence level for any selected MTBF value. Thus a sample of 57 valves operated for six months under simulated mission conditions would have a MTBF value $\geq$91,000 hours ($\leq 11.0 \times 10^{-6}$ failures/hours) with a 50% confidence level. If zero failures occur, the design goal of MTBF $\geq$333,000 hours ($\leq 3.0 \times 10^{-6}$ failures/hour) will have been demonstrated with a confidence level of approximately 50% and MTBF $\geq$91,000 hours with a confidence of approximately 94%.

A demonstration test plan should include the use of a SUBLEX propellant, various combinations of on-off duty cycles, temperature extremes, applied voltage extremes, and provisions for detecting valve open failures, valve closed failures, and gas leakage.

A suggested test program for obtaining a high reliability valve for subliming solid application would consist of:

a. A preliminary screening of available valve types, using engineering judgment and available test information to select a valve type from each of two valve manufacturers.

OPERATING CHARACTERISTIC CURVE
DEMONSTRATION TESTING

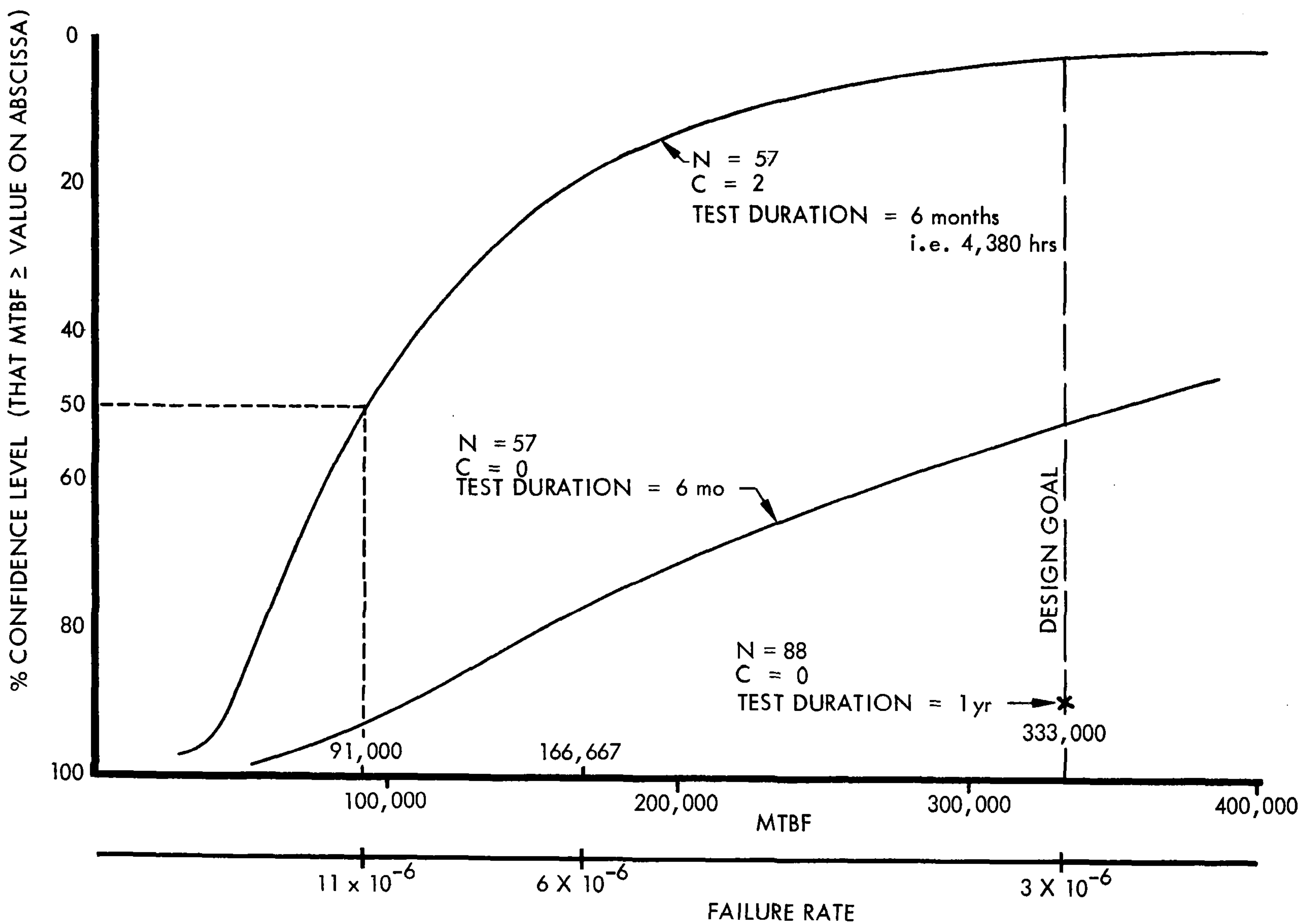


FIGURE 21

b. Defining the acceptance tests requirements, to include:

1. Particle count test using a Freon flush
2. Internal leakage at a specified pressure
3. Pressure drop at specified flow conditions
4. Pull in voltage
5. Drop out voltage
6. Response time at specified voltage, open and close
7. Coil resistance
8. Coil to case insulation resistance at 500 VDC
9. Dielectric strength test at 1,000 VDC
10. External leakage test
11. Pressure proof test
12. Visual examination of valve seat for imperfection at 30 power magnification

c. Conduct development tests that include:

1. Materials Evaluation
   (a) Material compatibility test
   (b) Tensile strength of condensed propellant to various materials considered for exposed surface of moving valve part
2. Prototype valve evaluation, one unit, redesign for failure
   (a) Vibration, humidity, shock
   (b) Repeat acceptance test
   (c) On-off cycle during altitude temperature test; temperature varied between high and low extremes
   (d) Repeat acceptance tests
   (e) Particulate flow test using known particle sizes of propellant

(f) Plot current (I) versus voltage (E) at 500 V increments to insulation breakdown at maximum-rated working temperature

(g) Torque to two times specified torque for installation

(h) Burst test

3. Prototype valve evaluation, one unit

   (a) Six-month vacuum test using propellant and high vacuum

d. Conduct qualification tests on three units in following sequence:

   1. Vibration
   2. Acceleration
   3. Shock
   4. Temperature altitude
   5. Humidity
   6. Salt spray
   7. Repeat acceptance test
   8. Cycle test to failure or completion of reliability demonstration test using same cycle conditions as demonstration test
   9. Tear down and visually examine all parts of valve at completion

e. Reliability Demonstration

   1. Select one of the two valve designs and cycle test 57 valves for six months using the postqualification configuration, propellant gases as effluent, vary on-off times, and vary the valve ambient temperature.

4.0 NOZZLE OPTIMIZATION

4.1 Introduction

Nozzle geometry optimization is a classical problem encountered in the design of all rocket engines. Optimum nozzle area ratios can usually be determined accurately by utilizing isentropic flow calculations and the thermochemical combustion or decomposition parameters of the propellant. These ideal gas computations are normally accurate to within a few percent. However, recent information generated by several different sources, including California Institute of Technology's Jet Propulsion Laboratory, Electro-Optical Systems, Incorporated, NASA-AMES, and NASA-LEWIS indicates that inviscid flow assumptions are no longer accurate for design of small nozzles where low nozzle throat Reynolds numbers are below about 1,000.

More important, however, is the fact that at low Reynolds numbers large energy losses are encountered in conversion of enthalpy to kinetic energy. In the case of the subliming solid, these losses are primarily nozzle friction losses, incomplete expansion of the exhaust gases, and directional flow losses (cosine losses).

Nozzle performance, therefore, may be expected to be sensitive to configuration (length, half-angle, expansion ratio), although continuum behavior will dominate throughout the thrust range of primary interest for reaction control systems. The above considerations suggest that there may be important trade-offs in nozzle expansion ratio and half-angle in order to achieve optimum performance. The optimum performance will be obtained with that nozzle design for which the sum of various losses due to friction, unused chemical and internal energy, heat transfer, and flow divergence are a minimum. Unfortunately, there has not yet been an intensive study of small nozzle optimization, although data are beginning to accumulate.

The nozzle optimization studies conducted during this program were intended to add further to the existing data by experimentally comparing the performance of several different nozzle configurations. In addition, a literature survey was conducted for the purpose of collecting and comparing results of various experimental studies of small nozzle performance.

4.2 Approach

The performance of a nozzle (that is, its ability to convert the internal energy of a gas into useful thrust) is usually specified as a nozzle coefficient, C_f, which is defined by the equation:

$$F_{vac} = C_f P_c A_t$$

where

F_{vac} = vacuum thrust

C_f = thrust coefficient

P_c = stagnation pressure at nozzle entrance

A_t = geometric area of throat

For an ideal nozzle in which isentropic expansion occurs infinitely to a vacuum, this coefficient, C_{f_∞}, is a function only of the gas specific heat ratio. For real nozzles, however, the actual coefficient is less than C_{f_∞} due to the losses previously mentioned.

The real nozzle thrust coefficient, C_f, is related to the maximum theoretical thrust coefficient, C_{f_∞}, by the equation:

$$\frac{C_f}{C_{f_\infty}} = C_v C_d$$

where

C_v = velocity (or performance) coefficient

C_d = discharge coefficient

The velocity coefficient, C_v, is the ratio of the average effective exhaust velocity achieved in a nozzle to the ideal, one-dimensional isentropic exhaust velocity achieved with an infinite area ratio. This coefficient, which is also known as the performance coefficient, is approximately equal to the ratio of the delivered specific impulse to the maximum theoretical specific impulse obtainable. The discharge coefficient, C_d, is the ratio of the actual mass flow passed by a nozzle of geometric throat area, A_t, to the ideal mass flow passed by a similar, one-dimensional, isentropic nozzle under the same initial conditions.

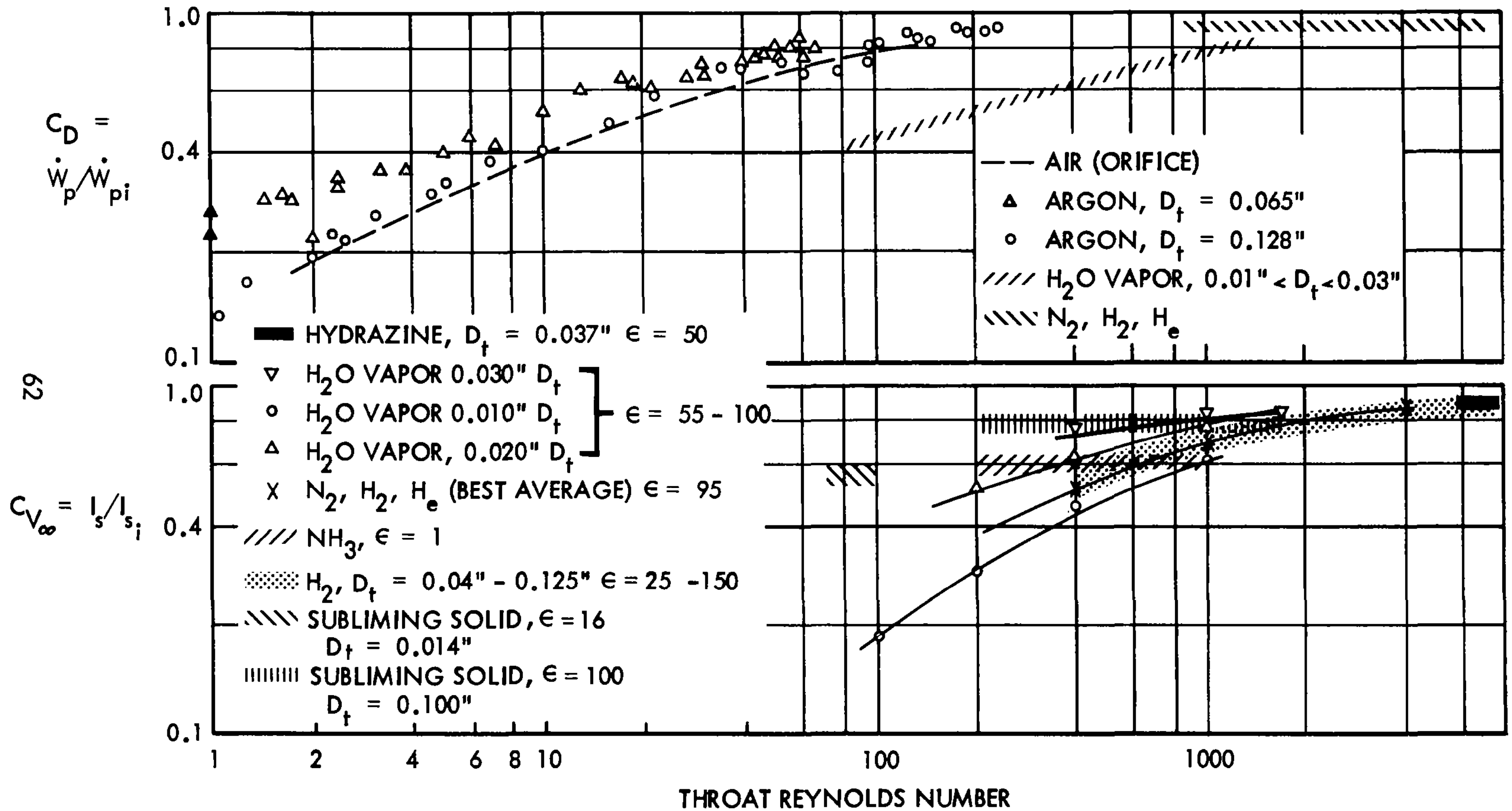
During the nozzle optimization study described herein, C_f data were obtained and correlated with the Reynolds number corresponding to nozzle throat conditions to enable comparison of data obtained for various nozzle sizes and for various working fluids and operating conditions. In addition, limited data were obtained correlating C_v and C_d to Reynolds number.

These data were obtained for a variety of working fluids, nozzle geometries, and initial conditions, and were evaluated in an attempt to define the trade-offs to be considered in the design of small nozzles.

4.3 Literature Survey

A literature search was conducted in an attempt to gather and compare the results of various experimental studies of small nozzle performance. Much data was taken from Reference 9. These data are shown in Figure 22 for C_d and C_v, respectively, wherein efficiency is plotted as a function of throat Reynolds numbers (Re). It should be kept in mind that differences in working fluid, nozzle contours, and surface finish are responsible for some of the data spread, whereas experimental difficulties in accurately measuring very low flow rates and thrusts also contribute to the uncertainty. The data for discharge coefficient show good correlation except for the results obtained with water vapor. The velocity (or performance) coefficient is not as well-behaved, and more data are obviously needed. Nevertheless, it appears that, depending to some extent on nozzle design and working fluid, velocity coefficients between 0.6 and 0.8 can be obtained for throat Reynolds numbers from about 75 to 2,000.

There is some evidence, based on work with water vapor and ammonia, that expansion vapors from initially saturated conditions may result in losses not experienced with highly superheated gases. For example, one investigator has observed condensation of NH_3 within a nozzle with an expansion ratio of less than 2. Further, it was observed that, using the same nozzle at throat Reynolds numbers well in excess of 100,000, performance of NH_3 was only about 60% of I_s, whereas GN_2 produced



$C_D = \dot{W}_p/\dot{W}_{pi}$
$C_{V_\infty} = I_s/I_{s_i}$
1.0
0.4
0.1
AIR (ORIFICE)
ARGON, D_t = 0.065"
ARGON, D_t = 0.128"
H_2O VAPOR, 0.01" < D_t < 0.03"
N_2, H_2, H_e
HYDRAZINE, D_t = 0.037" ∈ = 50
H_2O VAPOR 0.030" D_t
H_2O VAPOR 0.010" D_t
H_2O VAPOR, 0.020" D_t
∈ = 55 - 100
X N_2, H_2, H_e (BEST AVERAGE) ∈ = 95
NH_3, ∈ = 1
H_2, D_t = 0.04" - 0.125" ∈ = 25 -150
SUBLIMING SOLID, ∈ = 16 D_t = 0.014"
SUBLIMING SOLID, ∈ = 100 D_t = 0.100"
1
2
4
6
8
10
100
1000
THROAT REYNOLDS NUMBER

FIGURE 22

about 77% at the low expansion ratio employed. Data shown in Figure 23 for water vapor, as compared to other gases, suggest that the same mechanism was encountered. Thus, one point which needs further study is the effect of degree of superheat on performance. Due to the small nozzle dimensions involved and the resulting short residence times, no recondensation in the nozzle proper has been observed with subliming solid rockets, but it is not clear that vaporizing liquid rockets (which do not break up into two or more different types of molecules as do many subliming solid materials) are equally immune from this effect.

In addition to the above data, Electro-Optical Systems, Incorporated, (Reference 10) performed an analysis to determine the magnitude of nozzle losses in the low Reynolds number regime and came up with some significant trends. The following theoretical results are based upon these simplifying assumptions:

a. Gas flow was assumed to be (1) fully frozen or (2) in a complete equilibrium expansion (calculated from Mollier Chart). Frozen flow losses, if any, are thus assumed independent of nozzle shape and size.

b. Core flow assumed isentropic, and viscous boundary layer assumed to occupy a small fraction of the cross-section (this assumption questionable).

c. No heat transfer.

d. Nozzle conical, exit flow spherically symmetric (source flow).

e. Viscous boundary layer taken to "start" at nozzle throat. Any losses up to the nozzle throat were not considered, and the flow at the throat was assumed uniform.

f. Effects of pressure gradient and streamline divergence on the friction coefficient were neglected. The laminar flat plate friction law was used.

Typical results from these analyses for an exhaust gas specific heat ratio of 1.4 are presented in Figures 23 and 24 for Reynolds numbers ranging from 100 to 2,000. These results are preliminary in nature and show trends only. From Figure 23 it

NOZZLE THRUST RATIO VERSUS EXPANSION HALF-ANGLE
FOR DIFFERENT REYNOLDS NUMBERS

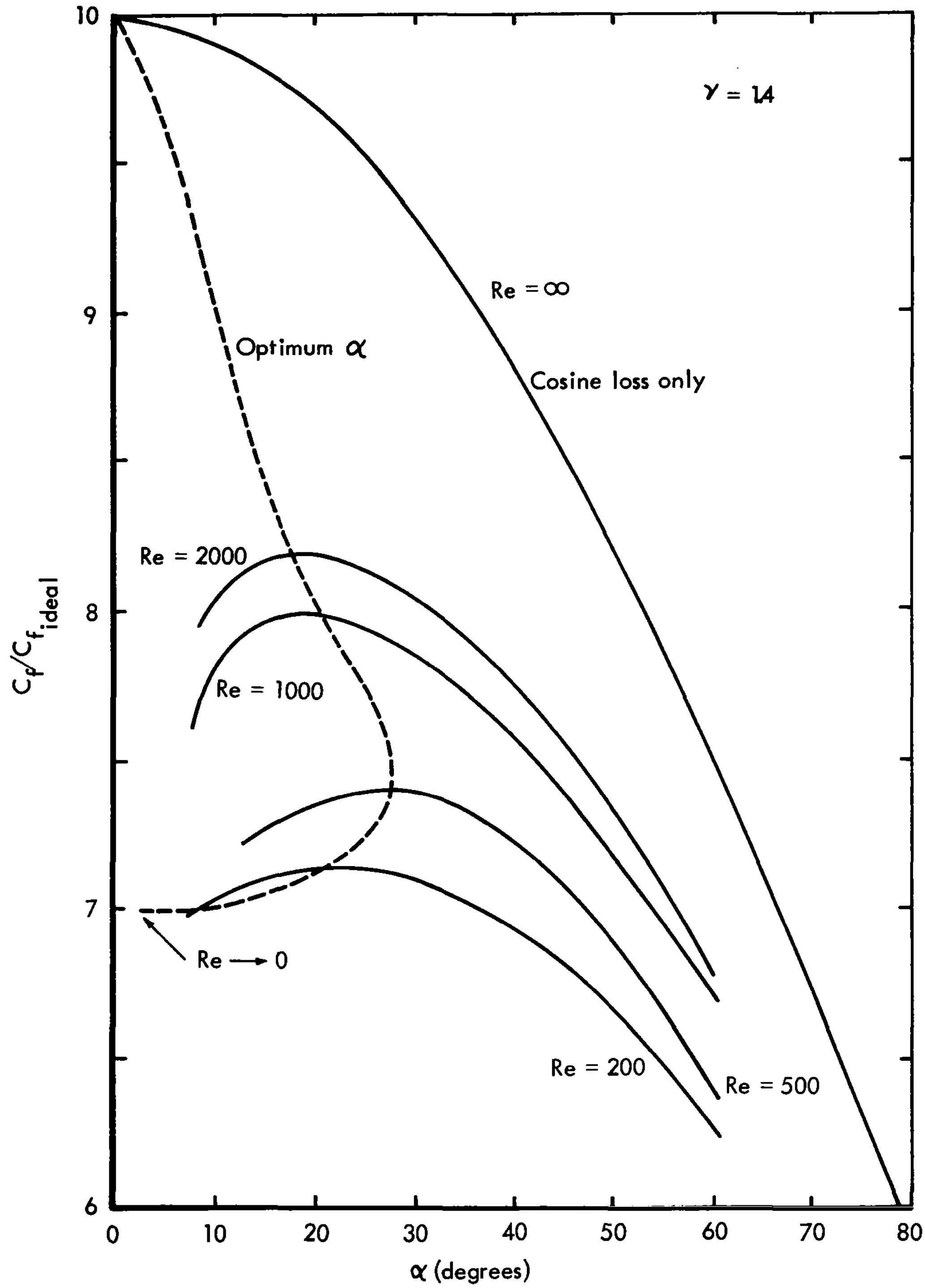


FIGURE 23

OPTIMUM EXPANSION ANGLE AND AREA RATIO AND MAXIMUM THRUST COEFFICIENT VERSUS NOZZLE THROAT REYNOLDS NUMBER FOR $\gamma = 1.40$

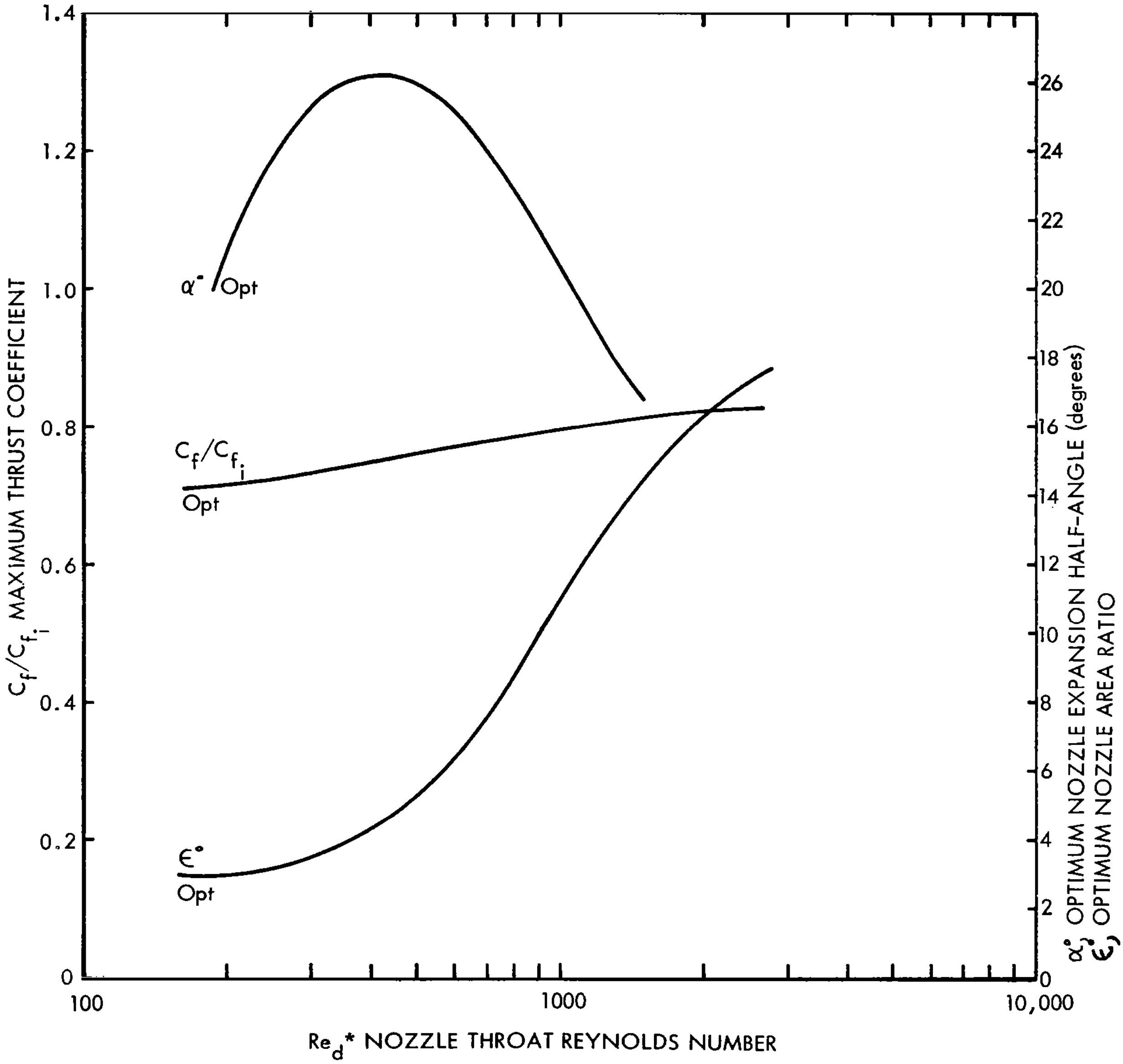


FIGURE 24

may be seen that as throat Reynolds numbers decrease, $C_f/C_{f_{ideal}}$ decreases and the optimum nozzle half-angle shifts from 16° to the 25° to 30° range. (Note that C_{f_i} is, for the EOS data, defined as the ideal thrust coefficient at the corresponding area ratio and not at an infinite area ratio.) At lower Reynolds numbers (200 to 500) the choice of half-angle is not sensitive as long as it is roughly in the proper range (within approximately ± 3° of optimum). The important consideration in determining the optimum nozzle design is not the selection of the precise optimum angle, but rather, for a chosen angle (near optimum) the use of the correct A/A* (or nozzle length).

Figure 24 summarizes results for a diatomic gas ($\gamma = 1.4$) under frozen flow conditions. The optimum expansion angle, area ratio, and thrust coefficient performance are plotted against nozzle throat Reynolds number. At very low Reynolds numbers the "optimum" nozzles are quite short and do not give appreciably more thrust than a sonic throat alone ($C_f/C_{f_i} = 0.7$ for $\gamma = 1.4$). For very short nozzles the calculations are pessimistic, however, as the initial friction factors close to the throat are too high. Results from these curves indicate that at low Reynolds numbers (100 to 2,000) the optimum area ratio is quite small and the nozzle has a large half-angle.

4.4 Test Program

A method often used in comparing nozzle performance is to determine the thrust coefficient ratio versus the throat Reynolds number. The thrust coefficient ratio is defined as the ratio of the actual measured thrust coefficient of the nozzle tested to the theoretical maximum thrust coefficient for an infinite area ratio nozzle. The theoretical thrust coefficient (C_{f_∞}) can be determined from the following equation:

$$C_f = \sqrt{\left(\frac{2K^2}{K-1}\right)\left(\frac{2}{K+1}\right)^{\frac{K+1}{K-1}}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{K-1}{K}}\right]} + \left(\frac{P_e}{P_c}\epsilon - \frac{P_o}{P_c}\epsilon\right)$$

Where:

K = ratio of specific heats of gas

P_e = nozzle exit pressure

P_c = nozzle chamber pressure

P_o = ambient pressure

ϵ = area ratio

If $\epsilon = \infty$, the terms P_e/P_c and P_o/P_c go to zero. Therefore:

$$C_{f_\infty} = \sqrt{\left(\frac{2K^2}{K-1}\right)\left(\frac{2}{K+1}\right)^{\frac{K+1}{K-1}}}$$

For SUBLEX A, $C_{f_\infty} = 1.947$.

The actual vacuum thrust coefficient can be determined from the equation:

$$C_f = \frac{F_{meas} + P_o A_e}{P_c A_t}$$

Where:

F_{meas} = measured thrust, lbf

A_t = nozzle throat area, in^2

P_c = measured chamber pressure, psia

P_o = measured vacuum chamber pressure, psia

A_e = nozzle exit area, in^2.

Thrust is measured on the Rocket Research Corporation Compound Pendulum Balance, while chamber pressure is measured with a pressure transducer.

Throat Reynolds number (Re) is defined as follows:

$$Re = \frac{\rho V D}{\mu}$$

Where:

ρ = density

V = velocity

D = throat diameter

μ = viscosity

Since

$$\dot{w} = \rho AV$$

$$Re = \frac{4\,\dot{w}}{\pi D\,\mu\,g}$$

or since

$$\dot{w} = \frac{P_c A_t g}{c^*}$$

$$Re = \frac{12\,P_c D_t}{c^* \mu}$$

Where:

P_c = chamber pressure, psia

D_t = nozzle throat diameter, in

c^* = characteristic velocity, ft/sec

μ = viscosity, lb-sec/ft^2

c^* is dependent upon absolute temperature and is shown plotted against temperature in Figure 25. Since D_t and μ approximately are known, Reynolds number can be determined by measuring only P_c and determining c^* from the temperature.

The velocity correction factor, C_v, is equal to the ratio of the actual specific impulse, I_s to the maximum theoretical specific impulse, I_{s_∞}; it can be determined as follows:

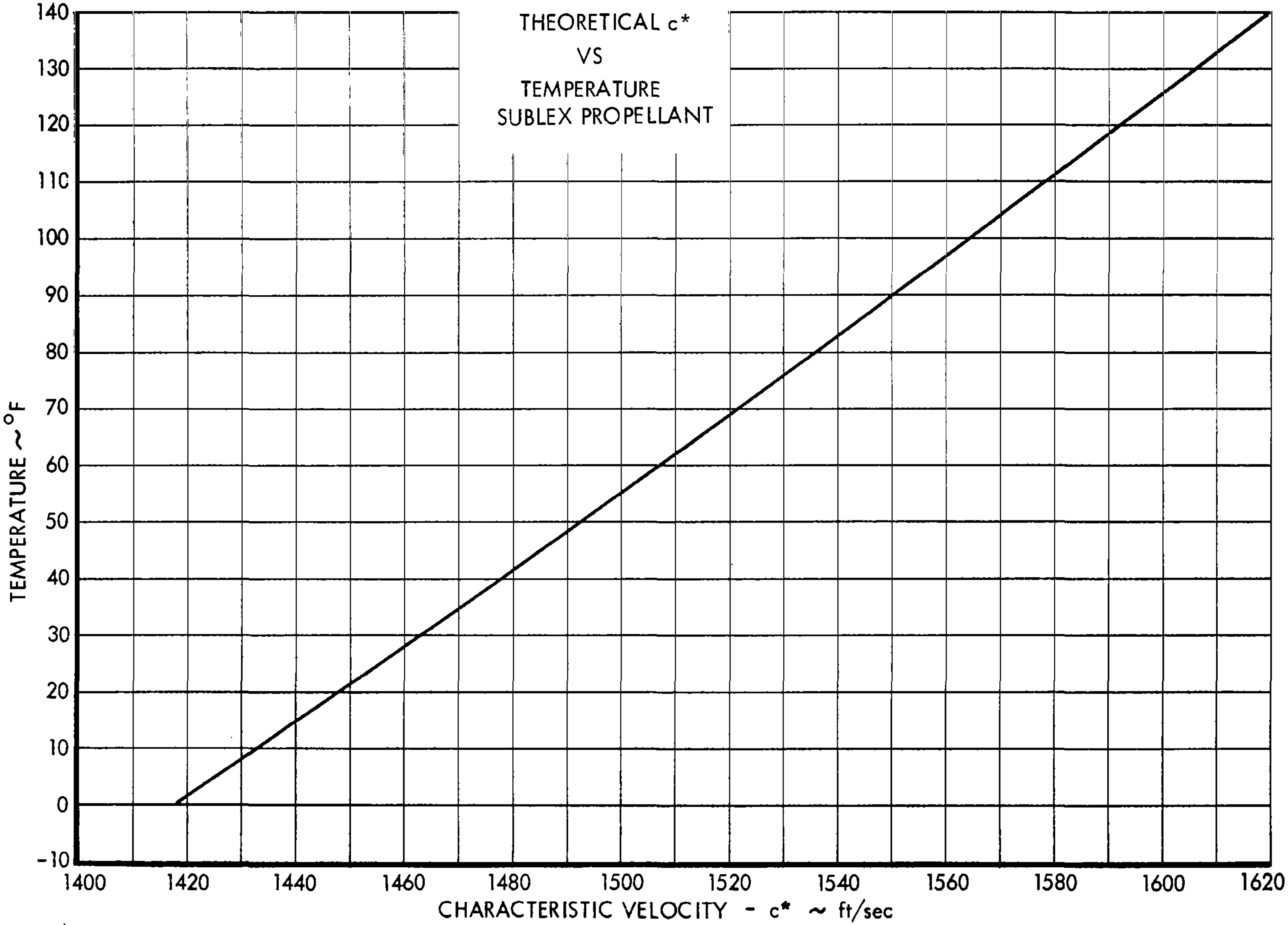

THEORETICAL c*
VS
TEMPERATURE
SUBLEX PROPELLANT
TEMPERATURE ~ °F
140
130
120
110
100
90
80
70
60
50
40
30
20
10
0
-10
1400
1420
1440
1460
1480
1500
1520
1540
1560
1580
1600
1620
CHARACTERISTIC VELOCITY - c* ~ ft/sec

FIGURE 25

Thrust, F, is measured on the compound pendulum balance over a certain time period, t. The amount of propellant flowed, W, over that time period is determined by weighing the tank before and after each run. Therefore the actual specific impulse, I_s can be determined:

$$I_s = \frac{\int F dt}{W}$$

An average velocity correction factor, $\bar{C}_v$, can then be determined:

$$\bar{C}_v = \frac{I_s}{I_{s\infty}} = \frac{\int F dt}{I_{s\infty} W}$$

Where:

$$I_{s\infty} = \frac{C_{f\infty} c^*}{g} = \frac{1.947\ (1523)}{32.2} = 92.2 \text{ sec}$$

The value of $\bar{C}_v$ is an average value corresponding to some average thrust level, which can be determined by:

$$F_{ave} = \frac{\int F dt}{t}$$

The discharge correction factor, C_d, is defined as the ratio of the actual mass flow to the maximum theoretical mass flow rate. It can be determined similarly to C_v. The actual average mass flow rate, $\dot{w}_a$, can be determined from measuring the amount of propellant lost over the total test time. The ideal mass flow rate can be determined from the equation:

$$\dot{w}_i = \frac{P_c A_t g}{c^*}$$

Then:

$$C_d = \frac{\dot{w}}{\dot{w}_i} = \frac{W c^*}{P_c t A_t g}$$

Since P_c is not constant

$$C_d = \frac{Wc^*}{A_t g \int P_c dt}$$

Again, C_d is an average value corresponding to an average pressure which can be determined by $P_{ave} = \int P_c dt/t$.

A schematic diagram of the test apparatus is shown in Figure 11. The entire system was installed within a vacuum chamber. The test procedure was as follows:

a. The propellant tank was weighed.

b. The desired nozzle was installed and the chamber evacuated.

c. The propellant valve was opened for 90 seconds.

d. Thrust and nozzle pressure were recorded with time.

e. The above procedure was repeated 4 to 10 times.

f. After a series of runs were completed, the propellant weight lost was determined by again weighing the tank.

All pertinent information could therefore be obtained with two measurements: thrust and nozzle pressure. The vacuum chamber pressure was below 1.0 micron at all times, thereby virtually eliminating the effect of back pressure on nozzle performance. Since no thrust control mechanism was applied, thrust and nozzle pressure dropped rapidly with time. Therefore, data points at several different Reynolds numbers could be calculated for each run. Run durations were nominally 90 seconds in length.

Several different nozzle configurations were tested, as described in Table VI. They were designed to yield comparisons between half-angle (α), throat diameter (D_t), and area ratio (A_c/A_t). Two different half-angles were tested, 15° and 20°. Throat diameters ranged from 0.142 to 0.01 inch. Area ratios ranged from 100 to 10. The SUBLEX system used was designed to operate at an initial thrust level of 1×10^{-2} lbf, with a drop-off to approximately 1×10^{-3} lbf in 5 minutes. In addition, some tests were conducted at a steady state thrust level of 1×10^{-3} lbf.

TABLE VI

NOZZLE CONFIGURATIONS

NOZZLE DASH NUMBER	HALF ANGLE DEGREES	THROAT DIAMETER INCHES	AREA RATIOS TESTED IN^2	REYNOLDS NUMBER RANGE
-3	15	0.142	100	400-1,400
-7	15	0.10	100,50	200-2,200
-11	15	0.0703	100,50,25,10	400-3,500
-13	20	0.0729	100,50,25,10	400-3,500
-15	15	0.045	100	200-5,000
-19	15	0.032	100	500-5,500
-21	20	0.032	100	500-5,500

4.5 Results and Conclusions

An analysis of the data obtained during the test program described in paragraph 4.4 verified the existence of significant performance losses that occur in small nozzles, which become particularly critical in the design of low flow rate, low pressure systems. Sufficient data are not available to determine optimum nozzle configurations for design purposes; however, the data indicate that optimum expansion ratios are much larger than those predicted by theoretical analysis, as shown in Figure 24. Also, significant trends are evident which indicate the direction to be taken during subsequent testing.

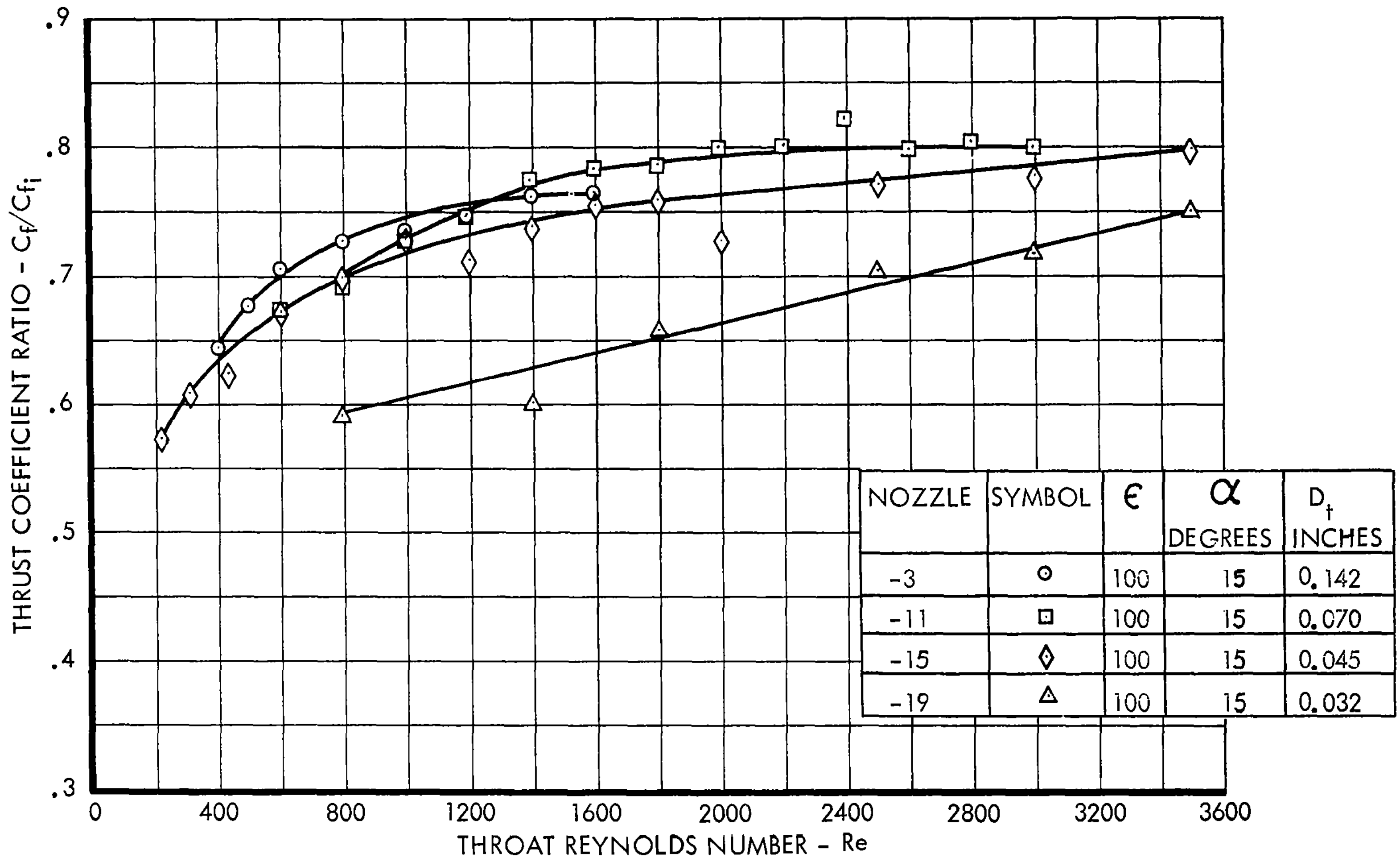
The most significant trend noted during the nozzle optimization tests was the drop-off in nozzle efficiency with decreasing throat Reynolds number (Re). This effect is best illustrated in Figures 26 and 27. It should be noted that in every case the nozzle efficiency decreased with decreasing Re. It appears that, if the curves were to be expanded to include higher Re data, they would converge at an Re value of between 5,000 and 10,000 at a C_f/C_{f_∞} value of between 0.8 and 0.85. From this apparent maximum value, the efficiency drops off gradually with decreasing Re, until a point is reached wherein C_f/C_{f_∞} begins to drop more rapidly. This point seems to occur at increasingly large Re as throat diameter is reduced, as can be seen in Figure 26. For the -3 nozzle (0.142 D_t), the drop-off occurs at Re $\cong$ 1,000; for the -11 nozzle (0.07 D_t) the drop-off starts at Re $\cong$ 1,800; for the -15 nozzle (0.045 D_t), the drop-off starts at Re $\cong$ 3,500; and for the -19 nozzle (0.032 D_t), the drop-off starts somewhere above Re = 4,000. This effect may occur because, for a given Re, the boundary layer thickness is a larger percentage of the throat diameter as the throat diameter is reduced. There were not sufficient data to make conclusions as to the effect of throat size at Re below 600, but it is expected the above effect will be greatly magnified and have therefore an even greater effect on performance.

An attempt was made to compare the effect of area ratio on nozzle performance to determine if there was an optimum area ratio below 100. A theoretical curve, published by Electro-Optical Systems, Incorporated, (see Reference 9), indicated that the optimum area ratio for Re below 2,000 was considerably below 100 for a half-angle between 15 and 20. Figure 24 indicates that the optimum nozzle configuration for a Reynolds number of 2,000 has a 16° half-angle and/or area ratio

THRUST COEFFICIENT RATIO

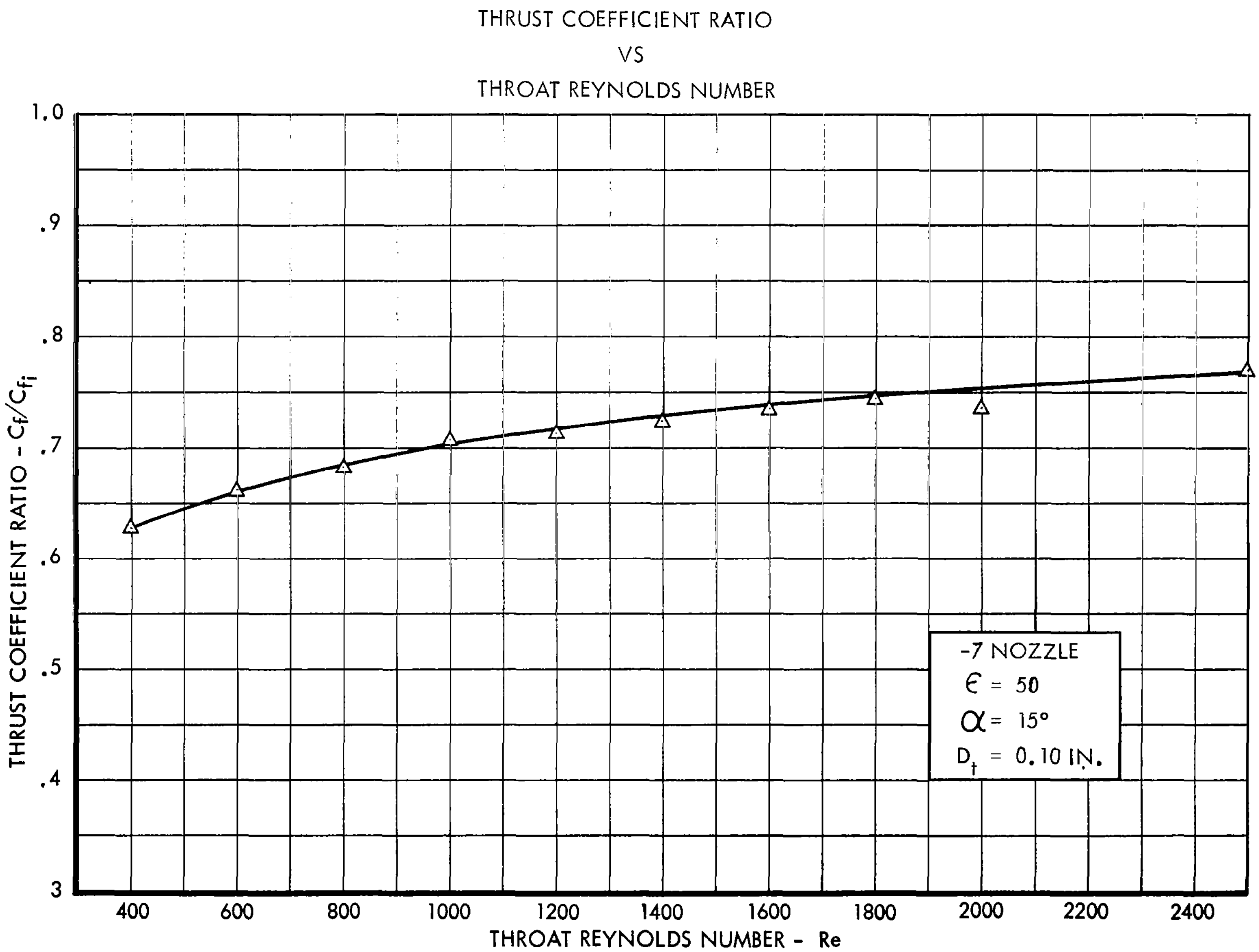
VS

THROAT REYNOLDS NUMBER



NOZZLE	SYMBOL	ϵ	α DEGREES	D_t INCHES
-3	⊙	100	15	0.142
-11	⊡	100	15	0.070
-15	◊	100	15	0.045
-19	△	100	15	0.032

FIGURE 26



THRUST COEFFICIENT RATIO
VS
THROAT REYNOLDS NUMBER
1.0
.9
.8
.7
.6
.5
.4
3
THRUST COEFFICIENT RATIO - C_f/C_{f_i}
400
600
800
1000
1200
1400
1600
1800
2000
2200
2400
THROAT REYNOLDS NUMBER - Re
-7 NOZZLE
ϵ = 50
α = 15°
D_t = 0.10 IN.

FIGURE 27

of 16. At Re = 1,000 the optimum configuration is indicated to be a 20° half-angle with an area ratio of only 11. (It should be noted here that the C_{f_i} used in Figure 25 is the ideal thrust coefficient at the corresponding actual area ratio, not at an infinite area ratio. This method is misleading and gives no basis for comparison of the absolute value of C_f.) The test data accumulated during this program indicates there is no optimum area ratio below 100 at any Re. Figure 28 is a graph showing the -13 nozzle for area ratios of 100, 50, 25, and 10. As indicated, the smaller area ratio yields the lowest performance in all cases. The same is true of the -11 nozzle as shown in Figure 29, except for the crossover between the 100 and 50 area ratio data below Re = 1,000. This crossover, however, is probably due to minor variations in test conditions or in nozzle surface finish, and is not a true indication of an optimum point. More data are clearly required before any conclusions can be drawn as to the existence at an optimum expansion ratio. It is interesting to note that the percent reduction in performance is much smaller than predicted by isentropic flow calculations as the area ratio becomes smaller. For example, the predicted drop-off in performance between ϵ = 100 and ϵ = 50 is 1.5 percent, as compared to a measured value about the same (from Figure 29). However, the predicted drop-off in performance between ϵ = 100, and ϵ = 10 is 7.6 percent, as compared with only about 3 percent measured.

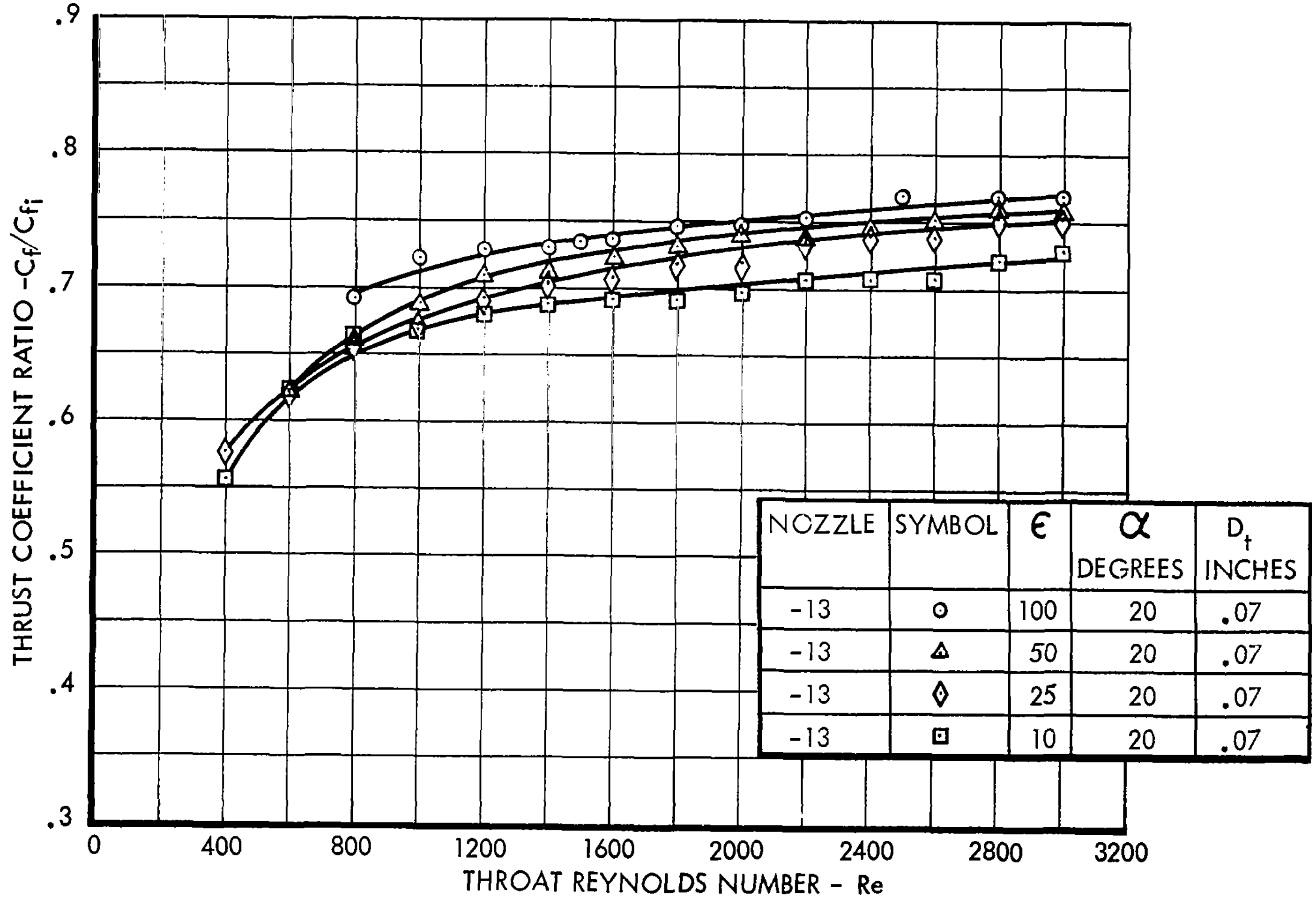
It was also attempted to determine the effect of half-angle on nozzle performance. Two half-angle configurations were tested: 15° and 20°, both at a throat diameter of 0.07 inches and at area ratios of 100, 50, 25, and 10. The results indicate some inconsistencies, as shown in Figures 28, 29, and 30, that make it difficult to draw any absolute conclusions. The -11 (15°) nozzle yielded better performance than the -13 (20°) for area ratios of 100 and 50. At an area ratio of 10, however, the situation was reversed, and the -13 yielded higher performance. More data are required to determine if an actual crossover point exists. The -19 and -21 nozzle data comparing the 15 and 20 degree half-angles indicates essentially no difference in performance for the Reynolds numbers tested.

In addition to determining the thrust coefficient ratio, C_f/C_{f_∞}, several measurements were made of the two coefficients comprising this ratio: C_v and C_d. This was attempted in order to obtain not only a discharge coefficient, C_d, but also

THRUST COEFFICIENT RATIO

VS

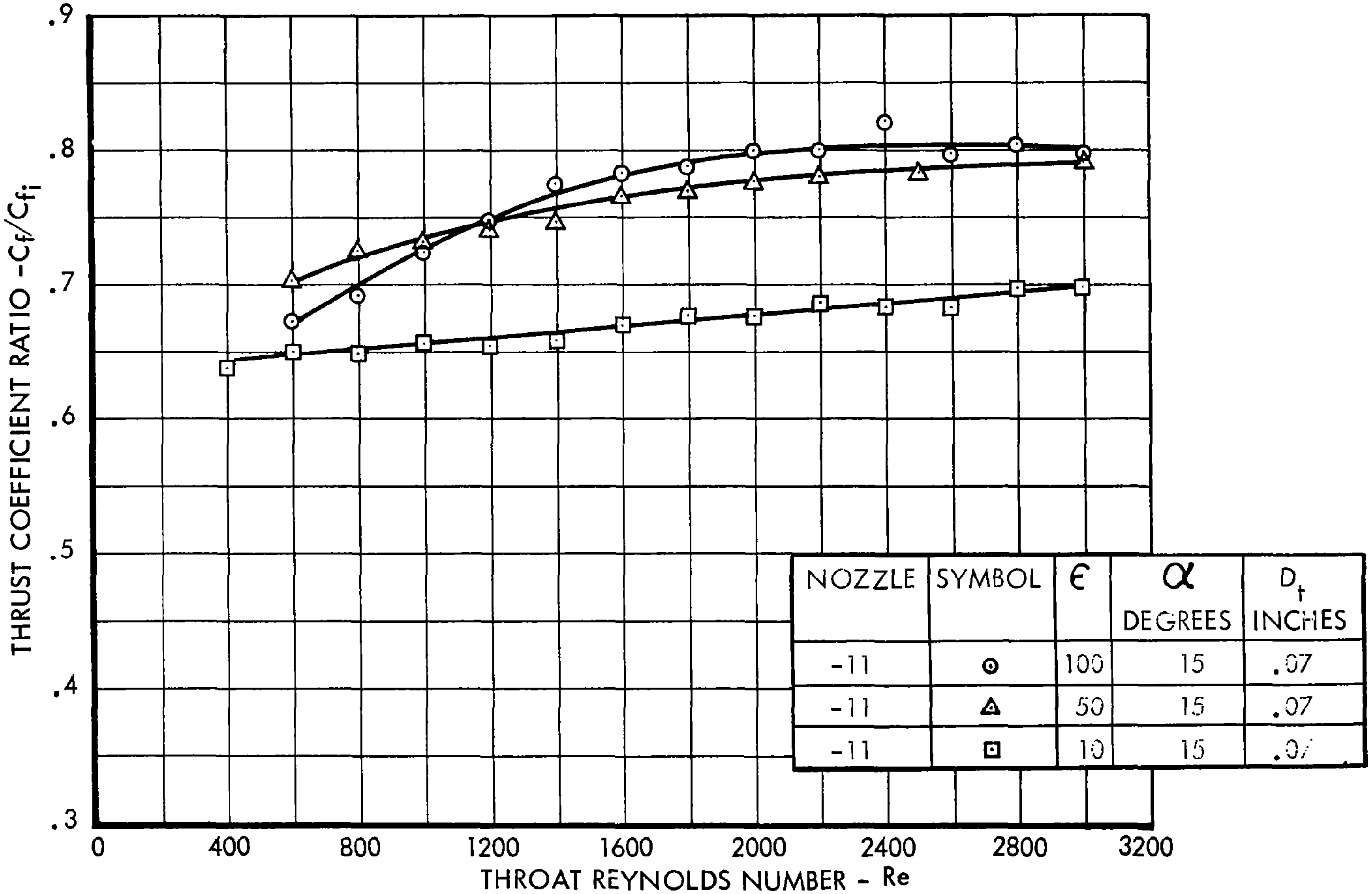
THROAT REYNOLDS NUMBER



NOZZLE	SYMBOL	ϵ	α DEGREES	D_t INCHES
-13	⊙	100	20	.07
-13	△	50	20	.07
-13	◊	25	20	.07
-13	⊡	10	20	.07

FIGURE 28

THRUST COEFFICIENT RATIO
VS
THROAT REYNOLDS NUMBER



NOZZLE	SYMBOL	ϵ	α DEGREES	D_t INCHES
-11	⊙	100	15	.07
-11	△	50	15	.07
-11	⊡	10	15	.07

FIGURE 29

THRUST COEFFICIENT RATIO

VS

THROAT REYNOLDS NUMBER

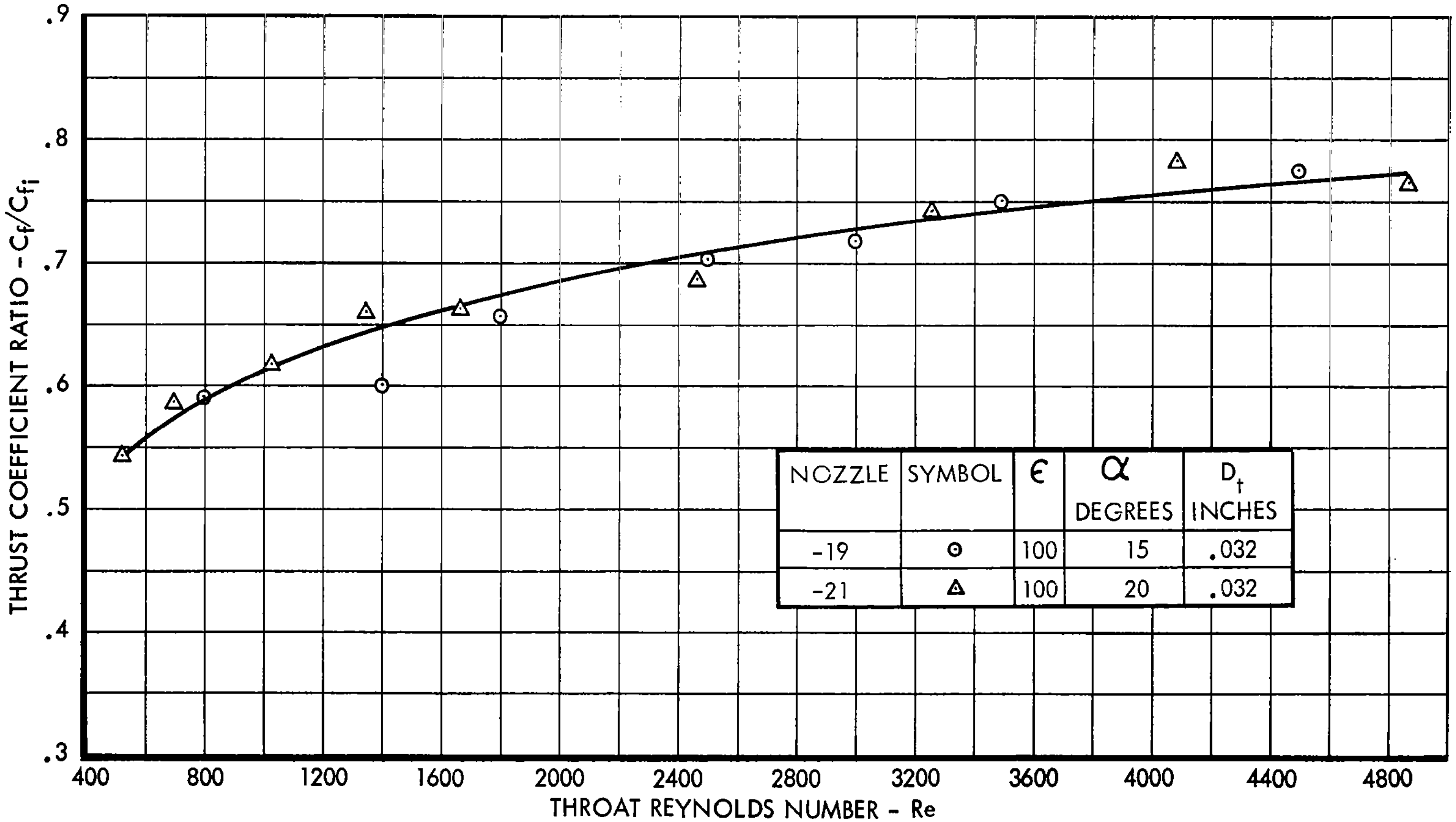


FIGURE 30

to obtain further measured specific impulse (I_s) data on SUBLEX A. As described earlier, C_v, the velocity coefficient, can be defined as the ratio of actual I_s to ideal I_s. I_s can be determined by integrating the thrust versus time trace and dividing by the total weight lost over that time. Nine I_s values were determined, yielding an average I_s equal to 74 lbf-sec/lbm. This corresponds to an average throat Reynolds number of approximately 800. Therefore, the number closely represents the performance that would be realized in a typical subliming solid system operating at a thrust level between 1×10^{-2} and 1×10^{-3} lbf. Specific impulse values measured by other methods have also yielded a value of approximately 74 seconds. An average C_v can also be determined by dividing by the ideal specific impulse. Therefore:

$$C_v = \frac{74}{92} = 0.80$$

C_d was also measured by the method described earlier. Four values were measured yielding an average $C_d = 0.91$. Since $C_f/C_{f_\infty} = C_v C_d$, an average thrust coefficient ratio can then be determined.

$$C_f/C_{f_\infty} = 0.9\ (0.80) = 0.72$$

This number represents closely the value obtained by direct measurement at a throat Reynolds number of 800. It must be kept in mind that the number represents an average value of results obtained from three different nozzle throat diameters. The significant fact is the specific impulse value obtained. Much more data is necessary to obtain accurate values of C_v and C_d. Based on Figure 23, it is believed that C_d will remain relatively constant down to very low throat Reynolds numbers, but this must be determined experimentally for each nozzle throat size. Then C_v can be determined simply by measuring C_f.

5.0 FLOW RATE MEASUREMENT INVESTIGATION

5.1 Introduction

The accurate measurement of very low subliming solid propellant flow rates (below 10^{-4} lbm/sec) is a major problem involved in microrocket performance measurement. Various methods have been used at Rocket Research Corporation with some success. Two methods involve the use of the perfect gas law, wherein an average mass flow rate is determined by measuring the pressure change in a plenum chamber of known volume. Two other methods involve the actual weighing of the amount of mass flowed over a selected time interval. In order to obtain a better understanding of the accuracy, along with a comparison of each of these methods, and to characterize each method as to system application, a flow rate measurement study was conducted. During this study, each of the above methods of measuring flow rate was investigated, tested when necessary, and the results then compared to the predicted flow rate. An OV2-1 prototype SUBLEX Respin Rocket System was used during this study. It should be noted that not all methods available are included herein, only those used often at Rocket Research Corporation. Following is a discussion of those methods.

5.2 Evacuated Plenum - Method I

Method I, the evacuated plenum method of measuring flow rate, consisted of flowing propellant into a known plenum volume and measuring the change in pressure in the plenum. The average flow rate during the time interval of flow was then calculated by use of the perfect gas law. Thus, from paragraph 2.4.4.2:

$$\dot{w} = \frac{\Delta PV}{RTt}$$

Where:

V	=	measured plenum volume, in^3
ΔP	=	rise in plenum pressure, psia
R	=	gas constant, FT/°R
T	=	absolute temperature, °R
t	=	time, seconds

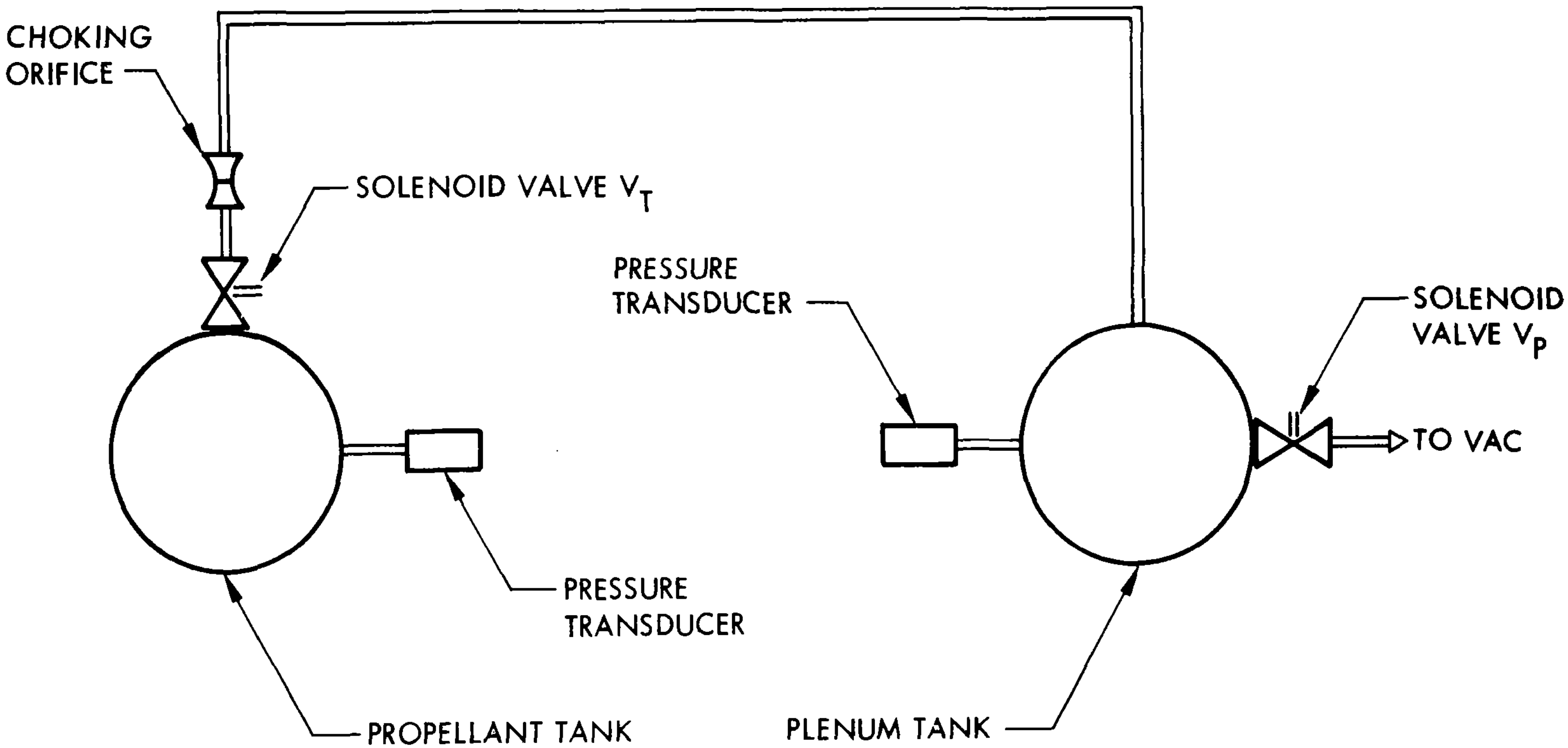
A schematic of the test apparatus is shown in Figure 31. The volume of the plenum tank and all lines up to the propellant valve V_t is measured. The plenum tank valve V_p is opened, allowing evacuation of the plenum volume. After V_p is closed, V_t is opened, allowing propellant to flow into the plenum volume. Tank and plenum pressure are recorded with time up to the point where unchoking occurs. Flow rate can then be determined by the above equation. Because of the drop of propellant pressure with time, flow rate also drops with time. The flow rate value determined is an average value corresponding to an average tank pressure. Data must therefore be taken over small time increments and correlated to tank pressure.

Figure 32 shows the results of the Method I flow rate versus tank pressure data. There is reasonably close agreement between the predicted curve and the measured data points. As expected, data taken in the higher tank pressure range show much more scatter than that taken in the lower range, the reason being that tank pressure drops rapidly at the higher pressures. This means that data must be averaged over a greater pressure range. It can be concluded that the evacuated plenum method is best suited for measuring relatively short duration (depending on plenum volume) flow rates and gives reasonably accurate results.

5.3 Differential Pressure - Plenum Method (Method II)

The differential pressure-plenum tank method is the same method used to measure the OV2-1 SUBLEX Respin Rocket System flow rate. This method is discussed thoroughly in paragraph 2.4.4.2. Only data in the lower flow rate ranges could be obtained during these tests due to limitations in the particular system apparatus. However, the data obtained fell closely along the theoretical line in much the same manner as the Method I data points. (See Figure 32.) This system has the advantage of always flowing from a pressure source to a vacuum, thus ensuring choked flow at all times. However, the duration of flow time is dependent upon the plenum chamber size. The data plotted in Figure 32 were first plotted against nozzle pressure, then later correlated back to tank pressure so that comparisons could be made with the other flow rate methods.

From the data obtained, it can be concluded that the differential pressure-plenum method is also an excellent way of obtaining flow rate data. Further, there is no



SCHEMATIC DIAGRAM - EVACUATED PLENUM METHOD I FLOW RATE

FIGURE 31

FLOW RATE
VS
TANK PRESSURE

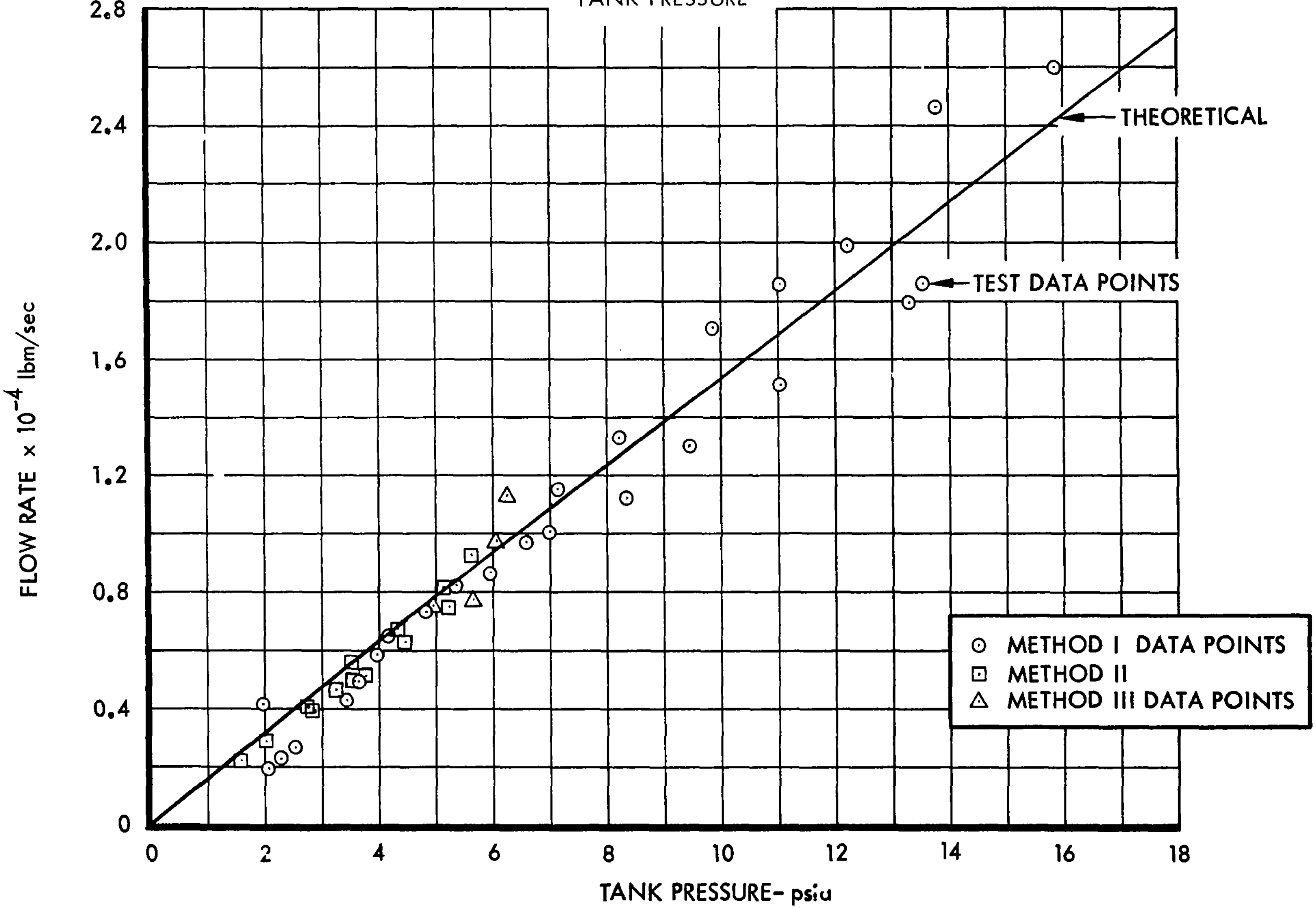


FIGURE 32

noticeable advantage for this system over the evacuated plenum method. They are both suited for flow rates between 10^{-4} and 10^{-6} lbm/sec.

5.4 Cold Trap - Method III

Method III of the flow rate measurement study consisted of actually weighing the amount of propellant flowed during a certain time period and then correlating this mass flow rate back to tank pressure. The propellant was passed through cold traps which recondensed all the propellant vapor. The traps are weighed before and after each run to determine the mass of propellant expended.

Figure 33 is a schematic diagram of the test apparatus. The propellant tank is connected directly to the two preweighed evacuated cold traps placed in series, which are connected to a vacuum pump. The traps are placed in the vacuum dewars filled with liquid nitrogen. The valve is opened, allowing propellant to flow through the traps, at which point all propellant vapor is recondensed. Tank pressure is recorded with time. At the end of the test the traps are again weighed and the amount of propellant calculated. Flow rate is then:

$$\dot{w} = \frac{\Delta W}{t}$$

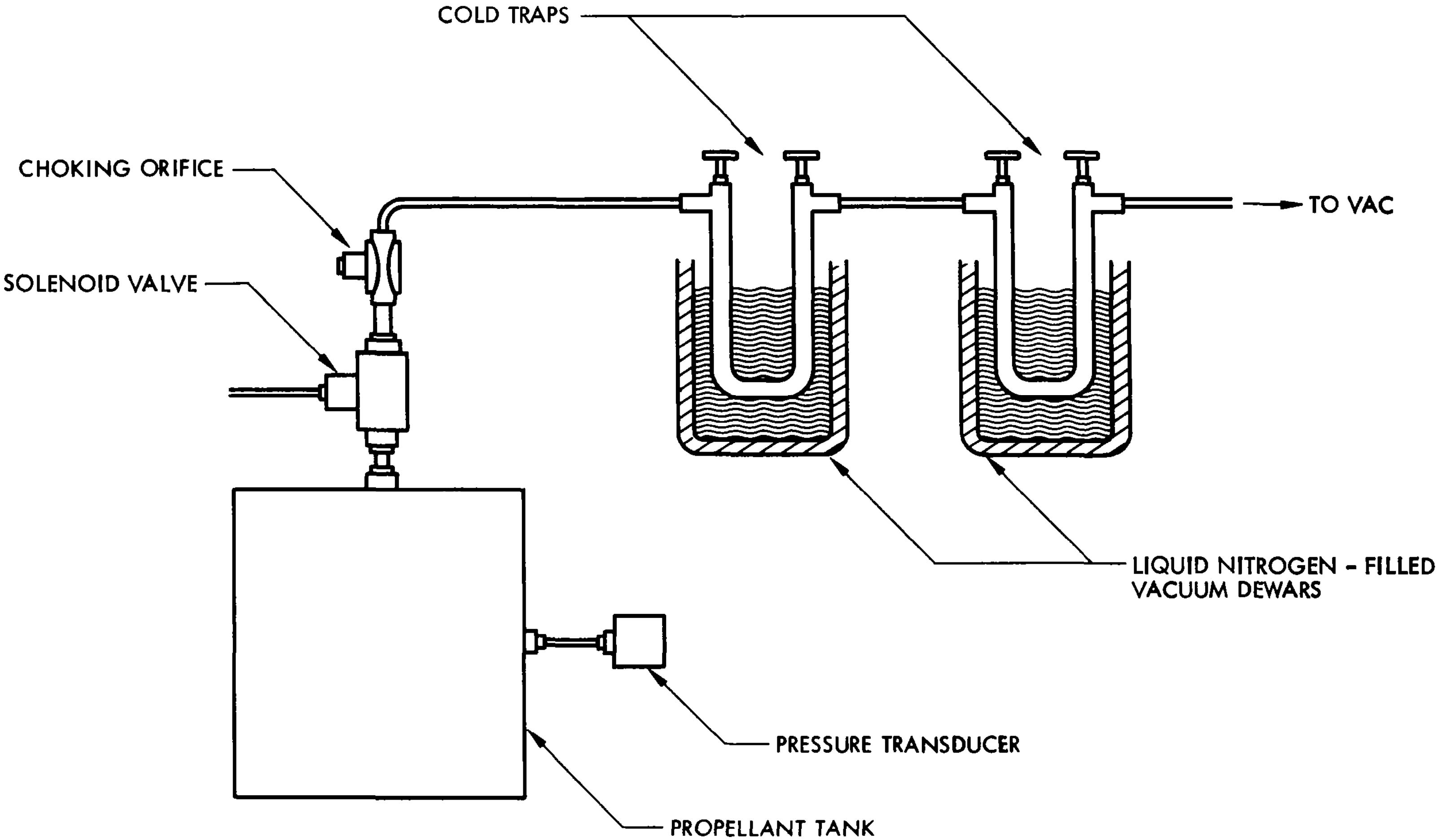
Where:

ΔW = propellant lost, lb

t = time of run, seconds

The tank pressure versus time curve is integrated, and the average tank pressure is found by dividing by the time.

The results of the test are plotted in Figure 32. As shown, the data fall closely along the theoretical curve. It is difficult, however, in a rapidly decreasing flow rate system such as the one tested, to correlate the flow rate with tank pressure due to the long duration runs necessary to obtain enough propellant to weigh. Tank pressure varies so much during this period that it is difficult to obtain an average tank pressure value. However, for subliming solid systems operating at constant pressure, this method can be an extremely accurate way in which to determine flow rate. It could be operated for long periods of time where a



SCHEMATIC DIAGRAM -COLD TRAP
METHOD III FLOW RATE

FIGURE 33

sizeable amount of propellant could be trapped and weighed accurately. Constant pressure subliming solid rockets can be easily constructed at the lower thrust levels, i.e., 10^{-3} to 10^{-6} lbs.

5.5 Differential Tank Weight - Method IV

Another method similar to the cold trap method (Method III) was investigated. This method involves weighing the propellant tank before and after a run and determining the amount of mass lost over the run period. The arguments listed under paragraph 5.4 apply again here. This method is suited for constant pressure systems such as those operating at low thrust levels. As an example, Rocket Research Corporation used this method to obtain flow rate data on another program. A system operating at constant pressure and at a flow rate of approximately 5×10^{-6} lbm/sec was pulsed continuously for a period of eight hours. The amount of mass lost was determined simply by weighing the tank before and after the run. Flow rate determined in this manner is very accurate.

5.6 Instantaneous Flow Rate - Method V

The last method of measuring flow rate yields instantaneous values corresponding to a measured pressure. Flow rate can be determined from the equation:

$$\dot{w} = \frac{C_d A_t P_c}{c^*}$$

Where:

C_d	=	discharge coefficient
A_t	=	area of choking orifice, in^2
P_c	=	pressure upstream of orifice, psia
c^*	=	characteristic velocity, ft/sec

It is first necessary to determine the discharge coefficient (C_d) for the system being operated. C_d should remain constant for all practical purposes over the entire pressure range. However, to determine C_d accurately, the method described in paragraph 5.5 can be applied. Thus:

$$C_d = \frac{Wc^*}{A_t \int P_c dt}$$

Where:

W = propellant lost, lb

$\int P_c dt$ = integral of P_c versus time curve

Once C_d has been determined accurately, then instantaneous flow rate can be determined by use of the first equation simply by measuring tank pressure versus time. The characteristic velocity is a theoretical number dependent only upon the characteristics of the gas and absolute temperature. No tests were conducted on this method due to limited time and money.

5.7 Conclusions

As a result of the flow rate study, several conclusions can be reached. First, the evacuated plenum (Method I) and the differential pressure-plenum (Method II) methods are very similar in concept, operation, and application. Both are indirect methods of measuring flow rate; that is, they are dependent upon the assumption that the fluid is a perfect gas. This assumption reduces the accuracy of the data obtained. They are ideally suited for short duration flow measurement at almost any flow rate. (The run time is dependent upon plenum volume.) Also, flow rate data can be determined by Methods I and II over very short time increments, thereby making it easier to correlate data to tank pressure thus increasing the accuracy of the data. Second, the cold trap (Method III) and differential tank weight (Method IV) methods are again similar in concept, operation, and especially application. Both are direct methods of measuring flow rate, which is desirable. However, they are most suited for flow rate measurement of constant pressure systems, which limits their use in subliming solid systems to flow rates below 10^{-5} lbm/sec. Generally, Methods III and IV require long duration steady state runs to obtain accurate data. Third, Method V is an indirect method of measuring flow rate that is dependent upon a direct method for calibration of the discharge coefficient, C_d. However, once C_d is determined accurately, this method becomes an accurate, simple, and straightforward means of obtaining instantaneous flow rate for any system simply by measuring pressure. This method is recommended where large quantities of data are required.

It is recommended that further work be done in the area of flow rate measurement. New and better methods should be found and investigated. A detailed error analysis should also be conducted on all methods so that the most accurate methods can be recognized. An extensive literature search should be conducted to determine other flow measuring methods being used in industry, and to compare them with the methods described herein.

6.0 PROPELLANT PROPERTIES

6.1 Discussion

Some propellant properties of SUBLEX A were determined experimentally during the Phase I Program (reference Final Report, Contract NAS 5-3599). However, with the addition of the chemistry laboratory at Rocket Research Corporation, it was advantageous to again determine propellant properties using more refined equipment and to obtain more accurate data. The properties of SUBLEX A that were determined are:

a. Vapor pressure versus temperature

b. Heat of sublimation

c. True density

d. Bulk density for several mesh sizes

The vapor pressure versus temperature relationship determined experimentally by Rocket Research Corporation agrees very closely with data in the literature (see Figures 34 and 35). The heat of sublimation determined was 741 Btu/lb, which is lower than the published value of 782 Btu/lb.

The true density (that is, the density of the pure SUBLEX A crystal), was found to be .0417 lb/in^3. The bulk density was determined for four different particle sizes. The results indicate a maximum density of .022 lb/in^3, which is lower than the previously used minimum bulk density of .027 lb/in^3. There is, however, an explanation for this occurrence. The bulk densities determined here were for propellant grains of nearly equal size. If several different propellant grain sizes were combined, the bulk density would increase due to more efficient particle packing. It is recommended that further work be done in this area to determine the optimum bulk density packing by measuring bulk densities of different combinations of propellant grain sizes.

6.2 Vapor Pressure versus Temperature

The vapor pressure-temperature relationship for SUBLEX A was determined from -60° to 90°F, covering the pressure range of 0.019 to 14.24 psia.

VAPOR PRESSURE
VS TEMPERATURE
FOR SUBLEX A

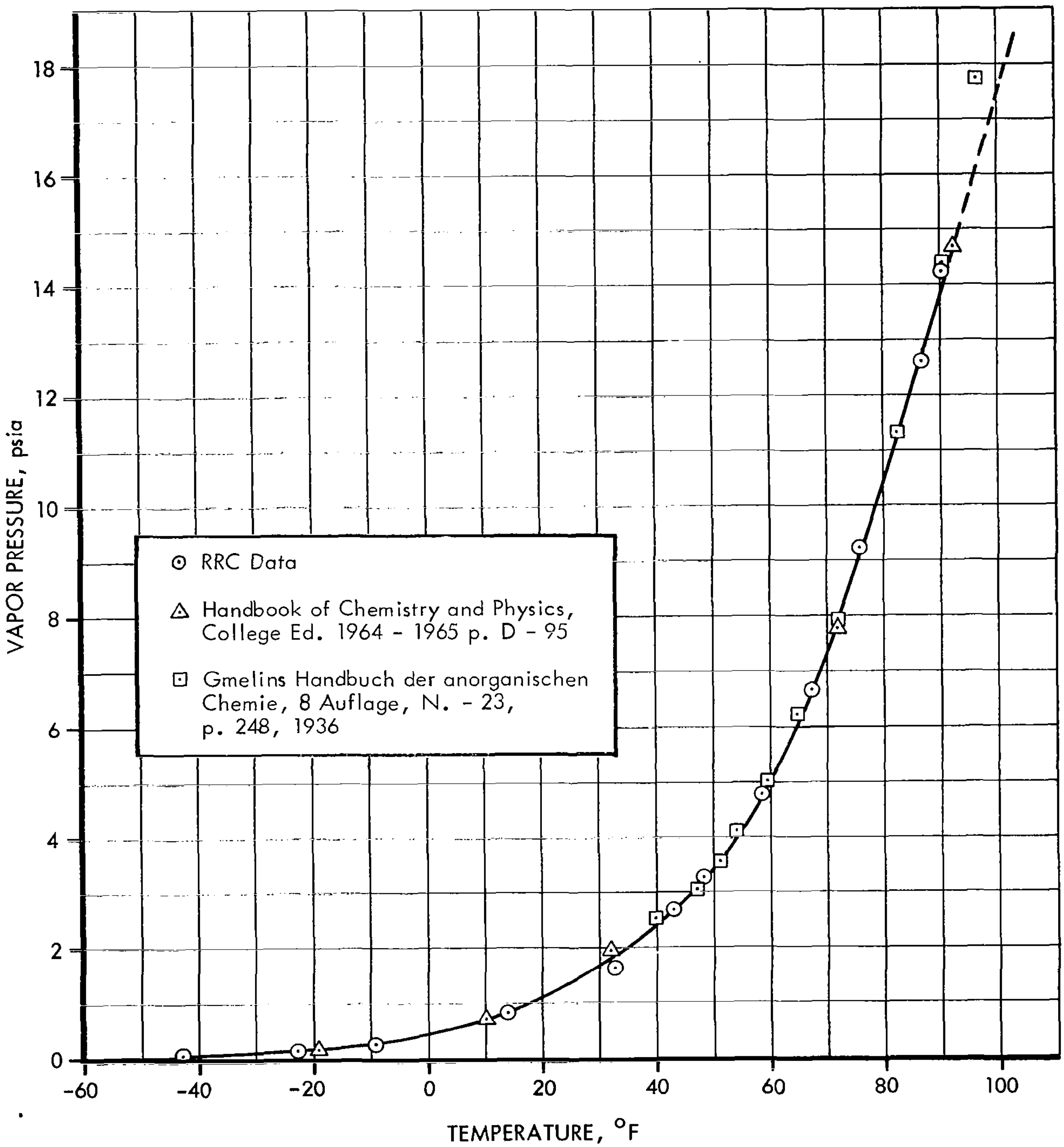
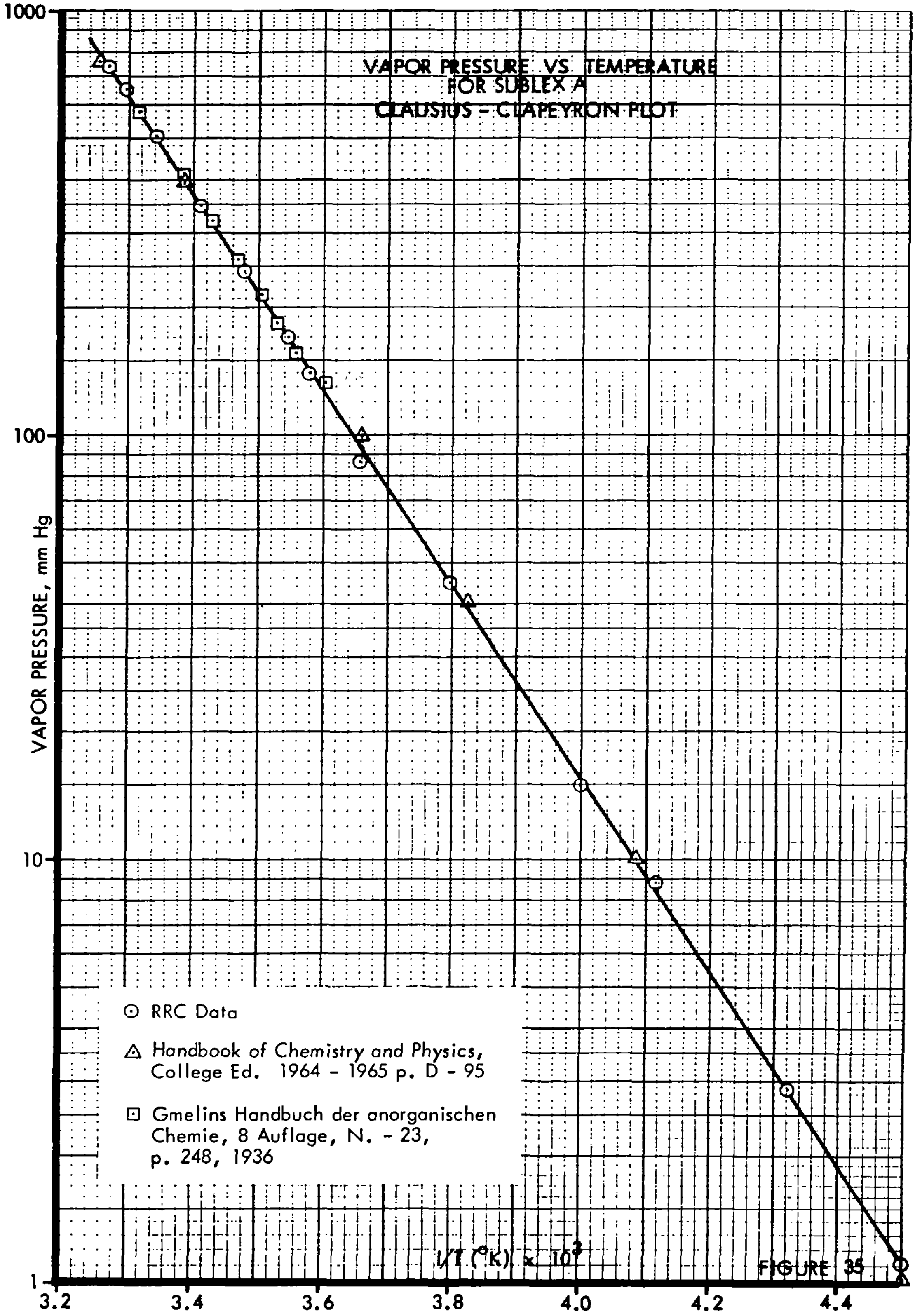


FIGURE 34


VAPOR PRESSURE VS TEMPERATURE FOR SUBLEX A
CLAUSIUS - CLAPEYRON PLOT
VAPOR PRESSURE, mm Hg
1000
100
10
1
3.2
3.4
3.6
3.8
4.0
4.2
4.4
1/T (°K) x 10³
RRC Data
Handbook of Chemistry and Physics, College Ed. 1964 - 1965 p. D - 95
Gmelins Handbuch der anorganischen Chemie, 8 Auflage, N. - 23, p. 248, 1936
FIGURE 35

From -60° to 77.7°F, the vapor pressure was measured using a high vacuum line (see Figure 36). A sample of resublimed SUBLEX A was transferred to trap A, where it was frozen at -196°C (liquid nitrogen temperature). All of the stopcocks were then closed. The liquid nitrogen bath was replaced by a controlled temperature bath, and the vapor pressure of the SUBLEX A was measured after thermal equilibrium was attained. (The vapor pressure remained constant.)

Above 77.7°F (room temperature), a modified vapor tensimeter (see Figures 37 and 38) connected to a high vacuum line (replaces traps B, C and D in Figure 36) was used to determine the vapor pressure. SUBLEX A was transferred to the sample bulb of the vapor tensimeter by high vacuum techniques (similar to the operation described in the previous paragraph). The tensimeter was then immersed in a controlled temperature bath, and the vapor pressure readings were taken with the aid of a cathetometer after thermal equilibrium was attained.

The vapor pressure of SUBLEX A at various temperatures is listed in Table VII. Vapor pressure-temperature values reported in the literature are also listed in Table VII for comparative purposes. The data are illustrated graphically in Figures 34 and 35.

6.3 Heat of Sublimation

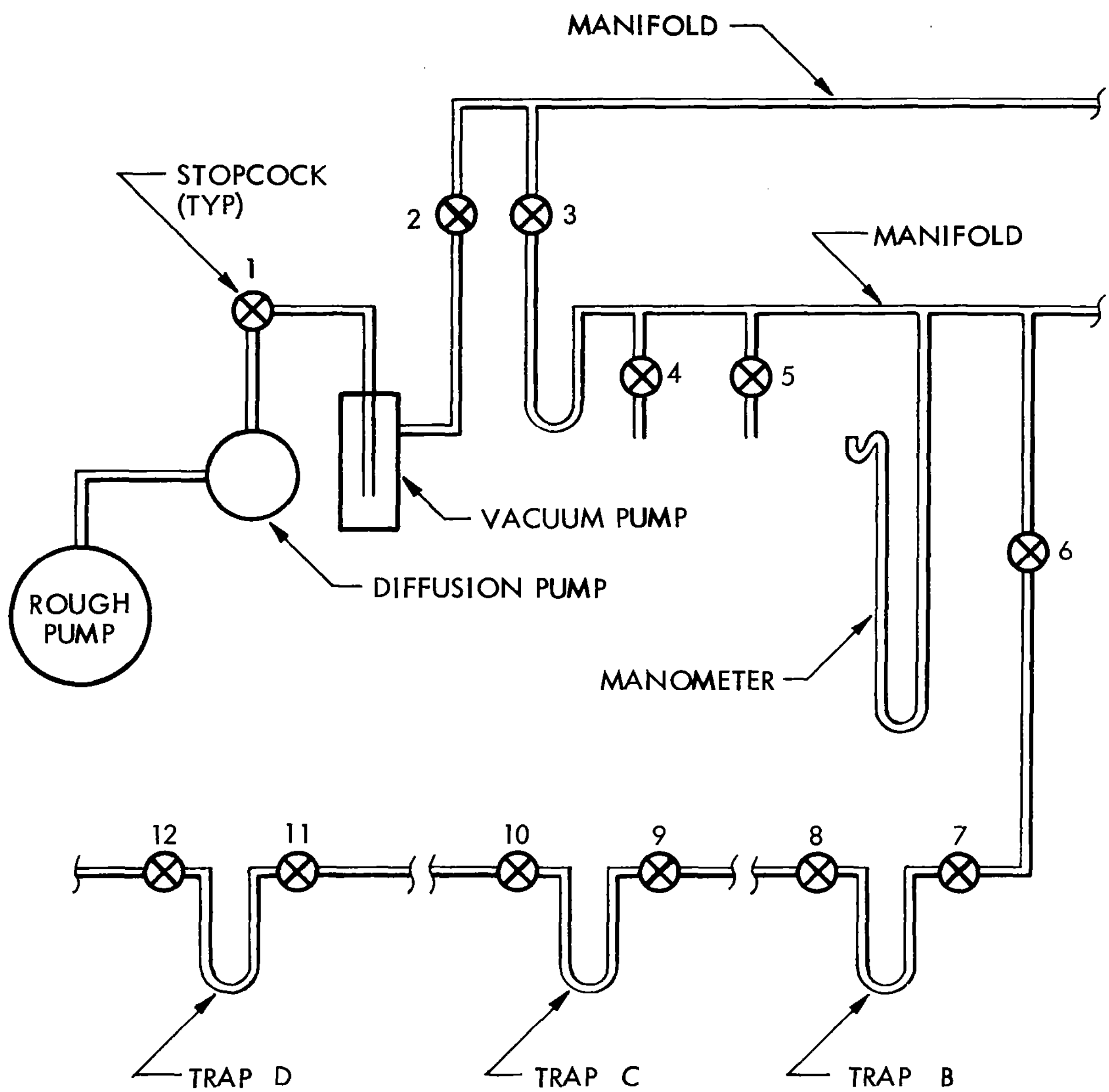
The vapor pressure-temperature data were plotted to fit the Clausius-Clapeyron equation (Figure 35). Data from the literature were also used in the plot (done in metric system units, mm Hg and °K). The heat of sublimation, ΔH_{sub}, was determined from the slope of the straight line fitted to the data by use of the following equation:

$$\Delta H_{sub} = -(\text{slope of line} \times 2.303 \times 1.987)$$

The heat of sublimation was calculated to be 10,500 (741 Btu/lb) calories per mole (see Table VIII for the calculations).

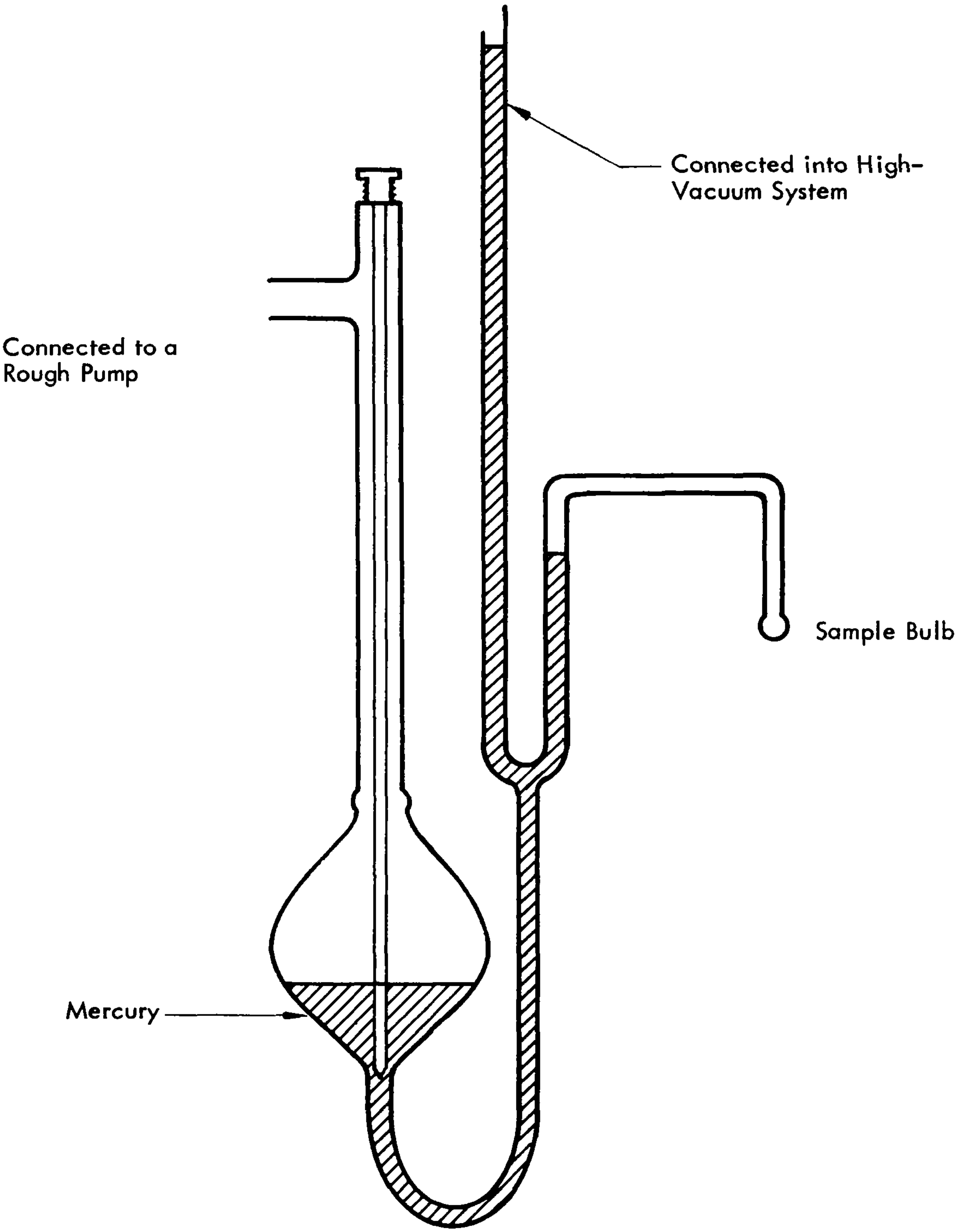
6.4 True Density

The density of SUBLEX A was determined by the following technique. Two tared, calibrated, 50 ml volumetric flasks were approximately half-filled with SUBLEX A (done in a glove box under a dry nitrogen atmosphere). The stoppered flasks



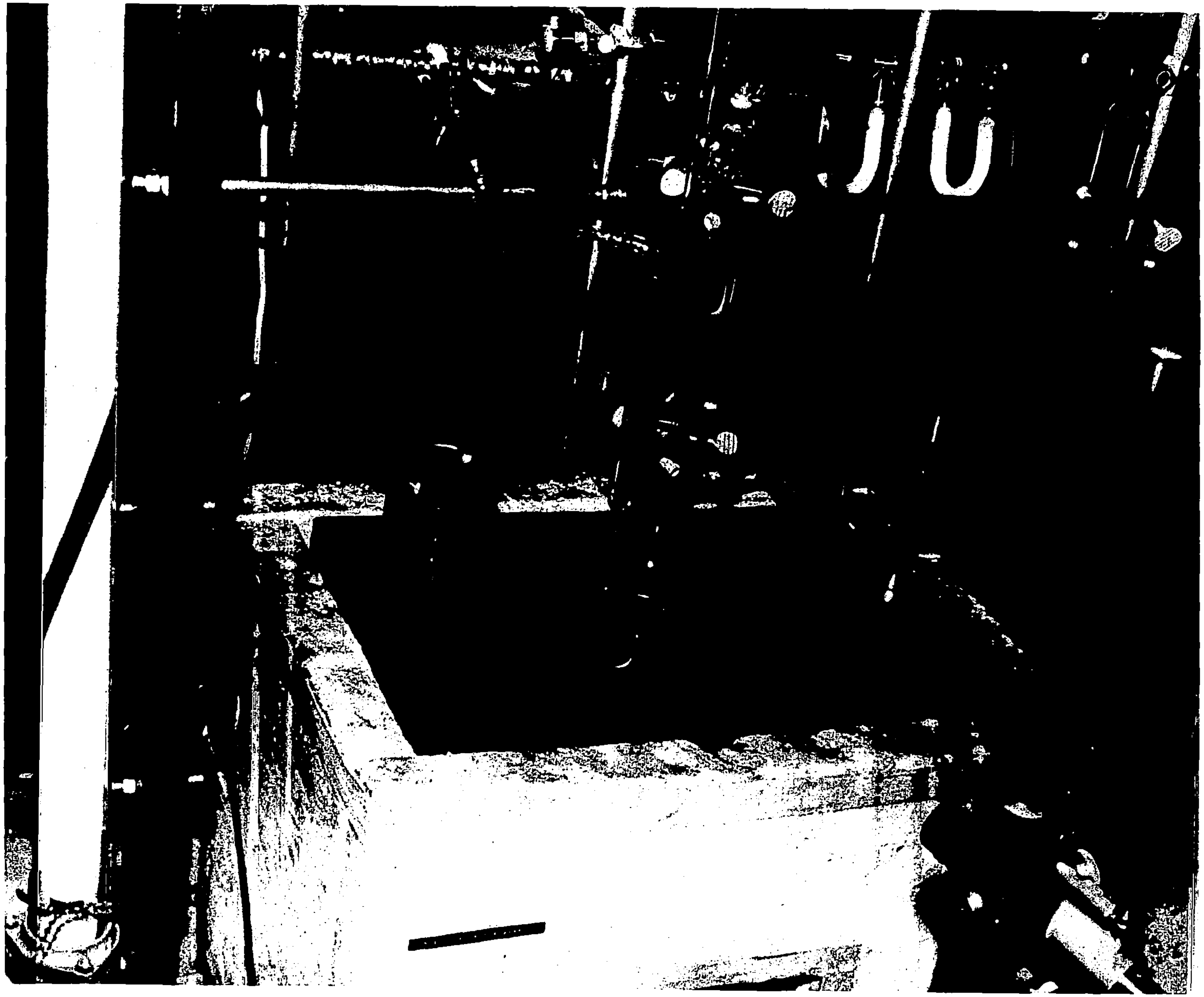
HIGH VACUUM LINE SCHEMATIC

FIGURE 36



SCHEMATIC OF MODIFIED VAPOR TENSIMETER

FIGURE 37



PHOTO, VAPOR TENSIMETER

FIGURE 38

TABLE VII

VAPOR PRESSURE VERSUS TEMPERATURE FOR SUBLEX A

Temperature, °F	Pressure, psia
Rocket Research Corporation Data	
-60.0	0.0193
-43.0	0.0541
-23.1	0.1702
- 9.6	0.2862
14.0	0.8530
32.4	1.675
42.8	2.698
48.2	3.336
58.3	4.811
67.3	6.739
77.7	9.732
86.5	12.650
90.0	14.239
Handbook of Chemistry and Physics, College Edition, 1964-1965, p D-95.	
-60.0	0.01934
-19.7	0.1934
10.0	0.7735
32.0	1.934
71.2	7.735
91.9	14.696
Gmelins Handbuch der anorganischen Chemie, 8 Auflage, N. -32, p 248, 1936.	
39.6	2.552
43.0	2.746
46.2	3.075
50.2	3.558
53.6	4.099
59.0	5.008
64.4	6.227
71.6	7.928
77.2	9.688
82.4	11.370
89.8	14.464
96.1	17.771
102.7	22.354
111.9	30.166

TABLE VIII

CALCULATION OF THE HEAT OF SUBLIMATION OF SUBLEX A

$$\text{Slope of line in Figure 36} = \frac{\log P_2 - \log P_1}{\frac{1}{T_2} - \frac{1}{T_1}}$$

$P_1 = 1.9$ mm, $\log P_1 = 0.27875$

$P_2 = 640$ mm, $\log P_2 = 2.80618$

$1/T_1 = 3.30 \times 10^{-3}$ °K

$1/T_2 = 4.40 \times 10^{-3}$ °K

$$\text{Slope} = \frac{0.27875 - 2.80618}{(4.40 - 3.30)\ (10^{-3})} = -2.2977 \times 10^3$$

$\Delta H_{sub} = -(\text{slope of line} \times 2.303 \times 1.987)$

$\Delta H_{sub} = -(-2.2977 \times 10^3 \times 2.303 \times 1.987)$

$\Delta H_{sub} = 10{,}500$ calories

$\Delta H_{sub} = 741$ Btu/lb

containing the samples were then weighed, and the flasks were returned to the glove box, where they were filled within an inch of the calibration line with toluene. After all the gas bubbles had been removed from the flasks by gently tapping them, the flasks were transferred to a 25.0°C constant temperature water bath. The toluene level was then adjusted to the calibration line after thermal water bath. The toluene level was then adjusted to the calibration line after thermal equilibrium was attained. The dried, stoppered flasks were then weighed. A density of 1.1534 g/ml (.0417 lb/in^3) was determined (see Table IX for calculations).

6.5 Bulk Density for a Range of Mesh Sizes

The bulk density ("as loaded" density) for four particle size distributions of SUBLEX A was determined as follows (all work being done in a glove box under a dry nitrogen atmosphere). A sample of SUBLEX A was ground, using a mortar and pestle, and the material was run through a series of Tyler sieves to obtain SUBLEX A of four particle size distributions. The bulk density of SUBLEX A of each particle size range was determined by adding the material to a tared, graduated cylinder, vibrating the cylinder, and adding more material as the SUBLEX A settled until a constant volume of50 ml was obtained (see Figure 39).

The sample was then weighed (results are listed in Table X). The phenomenon of decreasing bulk density from 20 mesh to 100 mesh and then increasing bulk density from 100 to 150 mesh was also noted on another Rocket Research Corporation Program. The bulk density figures may be different if another method of grinding and sieving is used.

The use of a vibrator to obtain settling was compared with the "tap" method of settling, in which the solid is packed by tapping the container instead of vibrating it. After the bulk density of the +100, -150 mesh range material had been determined by using a vibrator, the same material was used to determine the bulk density by the tapping technique. The results were identical.

PHOTO, APPARATUS IN GLOVE BOX

FIGURE 39

TABLE IX

DENSITY OF SUBLEX A

Density of toluene at 25.0° C = 0.86230 g/ml
Volume of Flask #2 (Run #1) = 49.899 ml
Volume of Flask #3 (Run #2) = 49.890 ml

	Run #1	Run #2
Weight of SUBLEX A + flask	39.6638 g	40.8368 g
-Weight of flask	30.7116 g	32.1778 g
Weight of SUBLEX A	8.9522 g	8.6590 g
Weight with toluene added	75.9983 g	77.3836 g
-Weight of SUBLEX A + flask	39.6638 g	40.8368 g
Weight of toluene	36.3345 g	36.5468 g
Volume of added toluene	$\frac{36.3345\ g}{0.86230\ g/ml} = 42.137 ml$;	$\frac{36.5468\ g}{0.86230\ g/ml} = 42.383\ ml$
Volume of flask	49.899 ml	49.890 ml
-Volume of toluene	42.137 ml	42.383 ml
Volume of SUBLEX A	7.762 ml	7.507 ml
Density of SUBLEX A	$\frac{8.9522 g}{7.762 ml} = \frac{1.1533 g}{ml}$;	$\frac{8.6590\ g}{7.507\ ml} = \frac{1.1535\ g}{ml}$

TABLE X

BULK DENSITY OF SUBLEX A

Run No.	Mesh	gm/ml	lbs/in^3	Microns	
1	+20-40	.662	.0239		
2	"	.662	.0239		
3	"	.644	.0233		
4	"	.646	.0233		
		Av. 0.654	0.0236	+840-420	
					Av. 630
5	+40-60	.590	.0213		
6	"	.594	.0215		
		Av. .592	.0214	+420-250	
					Av. 335
7	+60-100	.532	.0192		
8	"	.534	.0193		
9	"	.532	.0192		
		Av. .532	.0192	+250-149	
					Av. 200
10	+100-150	0.608	.0220		
11	"	0.610	.0220		
		Av. 0.619	.0220	+149-105	
					Av. 125

6.6 Recommendations

It is recommended that further tests be conducted on SUBLEX A to completely characterize its properties. The following additional properties should be determined:

a. Heat capacity

b. Thermal conductivity

c. Evaporation coefficient

d. Storage stability

e. Thermal stability

f. Hygroscopicity

g. Surface area as a function of particle size distribution

BIBLIOGRAPHY

1. Northrop Space Laboratories, "OV2-1 Satellite Design Review," NSL 64-220, August 25-26, 1964.

2. Rocket Research Corporation, "Model Specification for the OV2-1 SUBLEX Respin Rocket System," 171-23-1, December 29, 1964.

3. Rocket Research Corporation, "Recondensation Test Report, OV2-1 SUBLEX Respin Rocket System", 171-23-2, November 20, 1964.

4. Rocket Research Corporation, "OV2-1 Valve Life Test Report, Eckel Valve Company Model AF421562," 171-23-3, November 25, 1964.

5. Rocket Research Corporation, "Qualification Test Report, OV2-1 SUBLEX Respin Rocket System," 171-23-4, January 25, 1965.

6. Rocket Research Corporation, "Acceptance Test Report, OV2-1 SUBLEX Respin Rocket System," 171-23-5, January 25, 1965.

7. Rocket Research Corporation, "Operating Environment Test Report, OV2-1 SUBLEX Respin Rocket System," 171-23-6, March 31, 1965.

8. Rocket Research Corporation, "Final Report for Subliming Solid Control Rocket, November 22, 1963, through September 30, 1964," prepared for Goddard Space Flight Center under Contract NAS5-3599.

9. Rocket Research Corporation, "A Review of Microrocket Technology: 10^{-6} to 1-lb Thrust," George S. Sutherland and M. E. Maes, Seattle, Washington, AIAA Paper No. 65-620.

10. Rocket Research Corporation, "Development of the Subliming Solid Control Rocket, Phase 2, " Monthly Progress Reports 1 through 10, October, 1964, through July, 1965.

11. Rocket Research Corporation, "Test Specification OV2-1 SUBLEX Respin Rocket System - Qualification and Acceptance," October 24, 1964.

12. Northrop Space Laboratories, NSL-64-211A, "General Specification of Invironmental Requirements and Environmental Testing for the OV2-1 Satellite".

NASA-Langley, 1967 CR-712