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BOEING

NUMBER (14) D2-114118-2 Vol 1

(6)

TITLE ROCKET ENGINES - LIQUID PROPELLANT.

VOLUME I - SMALL ENGINES,

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CODE IDENT. NO. 81205

NUMBER (14) D2-114118-2-9 Vol[redacted]-1

(6)

TITLE: ROCKET ENGINES - LIQUID PROPELLANT.

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MODEL Research CONTRACT ______

ISSUE NO. ______ ISSUED TO. ______

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(11) 30 Oct 68

(12) 185p.

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ABSTRACT

This document contains information on rocket engines using propellants stored as liquids or gases.

Volume I contains information on engines used for reaction control systems and low velocity increment propulsion systems. Cold (ambient) gas, heated gas, monopropellant and bipropellant engines are considered. Estimates are made of performance, size and power requirements. Operating principles, hardware details and systems consideration are discussed.

Volume II has data on engines with thrusts of 100 pounds and larger.

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KEY WORDS

Bipropellant	Reaction Control Systems
Cold Gas	Thrusters
Hot Gas	Resistojets
Monopropellant	Radioisotope Thrusters
Propulsion System	Rocket Engine

VOLUME I

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1.0 NOMENCLATURE

A_c	catalyst bed frontal area
A_e	nozzle exit area
A_t	nozzle throat area
a^*	sonic velocity
C_D	discharge coefficient
C_F	nozzle thrust coefficient
C_v	velocity coefficient
c_p	specific heat at constant pressure
c_v	specific heat at constant volume
D_M	maximum diameter of thruster
D_{TC}	thrust chamber diameter
d_c	catalyst pack diameter
d_e	nozzle exit diameter
d_p	catalyst particle diameter
d_t	nozzle throat diameter
E_i	dissociation potential (volts) of the propellant molecule
F	thrust
G	catalyst bed loading, propellant weight flow ÷ catalyst bed frontal area
g_c	conversion constant
h_c	enthalpy of gas in combustion chamber
h_e	enthalpy of gas at nozzle exit
I_{sp}	specific impulse
I_t	total impulse
$I_{t_{min}}$	minimum impulse bit

L^*	characteristic length of combustion chamber, chamber volume/throat area
L_b	catalyst bed length
L_E	engine length
L_N	nozzle length
L_T	thruster length
L_{tc}	chamber length
$\dot{m}$	mass flow
N_T	number of injector orifices
P_a	ambient pressure
P_c	chamber pressure
P_e	nozzle exit pressure
P_i	input power
P_{in}	inlet pressure
P_j	jet power
P_{ss}	steady state pressure
ΔP_{CH}	catalyst bed pressure drop
ΔP_i	injector pressure drop
R	gas constant
Re	Reynolds number
S_o	spacing between orifice hole centers
T_c	gas temperature in the chamber
T_e	ga temperature at the exit
V_c	gas velocity in the chamber
V_C	chamber volume
V_e	gas velocity at the exit
W	mass flow parameter

W_a	ablative engine weight
W_{cg}	cold gas thrustor weight
W_E	engine assembly weight
W_R	radiation engine weight
$\dot{w}$	weight flow
α	nozzle divergence half angle
α	degree of dissociation
β	nozzle convergence half angle
γ	ratio of specific heats, c_p/c_v
δ^*	boundary layer displacement thickness
ϵ	nozzle expansion ratio, A_e/A_t
ϵ_b	porosity of catalyst bed
η_e	engine efficiency
η_f	frozen flow efficiency
η_p	heater power efficiency
η_N	nozzle efficiency
λ	nozzle divergence correction factor
ρ_b	catalyst bed density
ρ_p	catalyst particle density
ρ_T	gas density at throat
ϕ	catalyst sphericity

2.0 SUMMARY

Information on rocket engines using propellants stored as liquids or gases for use in attitude control and low velocity increment applications is presented. The thrust range for these engines is from micropounds to around 100 pounds.

Parameters which are important in the design or selection of a rocket engine for such applications are discussed and data from engines developed for spacecraft applications and research purposes presented. These include performance (both steady state and transient), weight, size envelope, reliability, life, power requirements, duty cycle, response, repeatability, cost and interface considerations.

The various types of rocket engines are discussed. These include cold gas, heated gas, monopropellant and bipropellant engines. The various propellants for each type of thrustor are also discussed. Rocket engine performance is presented from both the theoretical standpoint and from compilation of delivered performance data for existing systems. A tabulation of transient performance for existing engines operating in the pulse mode is included.

3.0 INTRODUCTION

To determine the performance of the various types of thermal and chemical propulsion systems, specific design data for the elements of the system are required as well as information on overall system performance. Similar information is required on the other systems - electric and nuclear in order to determine the best system for any particular application. Previous Boeing studies related to propulsion systems are indicated on the Figures 3.0-1 and 3.0-2 by document title and number. Future system work will be incorporated into the document series shown on Figure 3.0-2.

This document concerns itself with state-of-the-art and design data of one element in the system - the rocket engine. Previous work relating to all types of engines is shown in Figure 3.0-3. Future engine and thrustor work will be incorporated into the document system shown on Figure 3.0-4. Access to information on the remaining propulsion elements beyond the overall systems and engines may be obtained through the Space Propulsion Systems - Concepts document (D2-113549-1).

Thermal and chemical propulsion technology has been and will continue to be the subject of considerable investigation by government agencies, educational institutions, and aerospace contractors. The purpose of this series of documents is to provide a central source of information generated by outside agencies as well as to provide a repository for related Boeing research.

The purpose of volume one of this document is to present information on engines used for reaction control systems and low velocity increment propulsion systems. The properties of the gases and liquid propellants used in these engines and some system considerations are also included.

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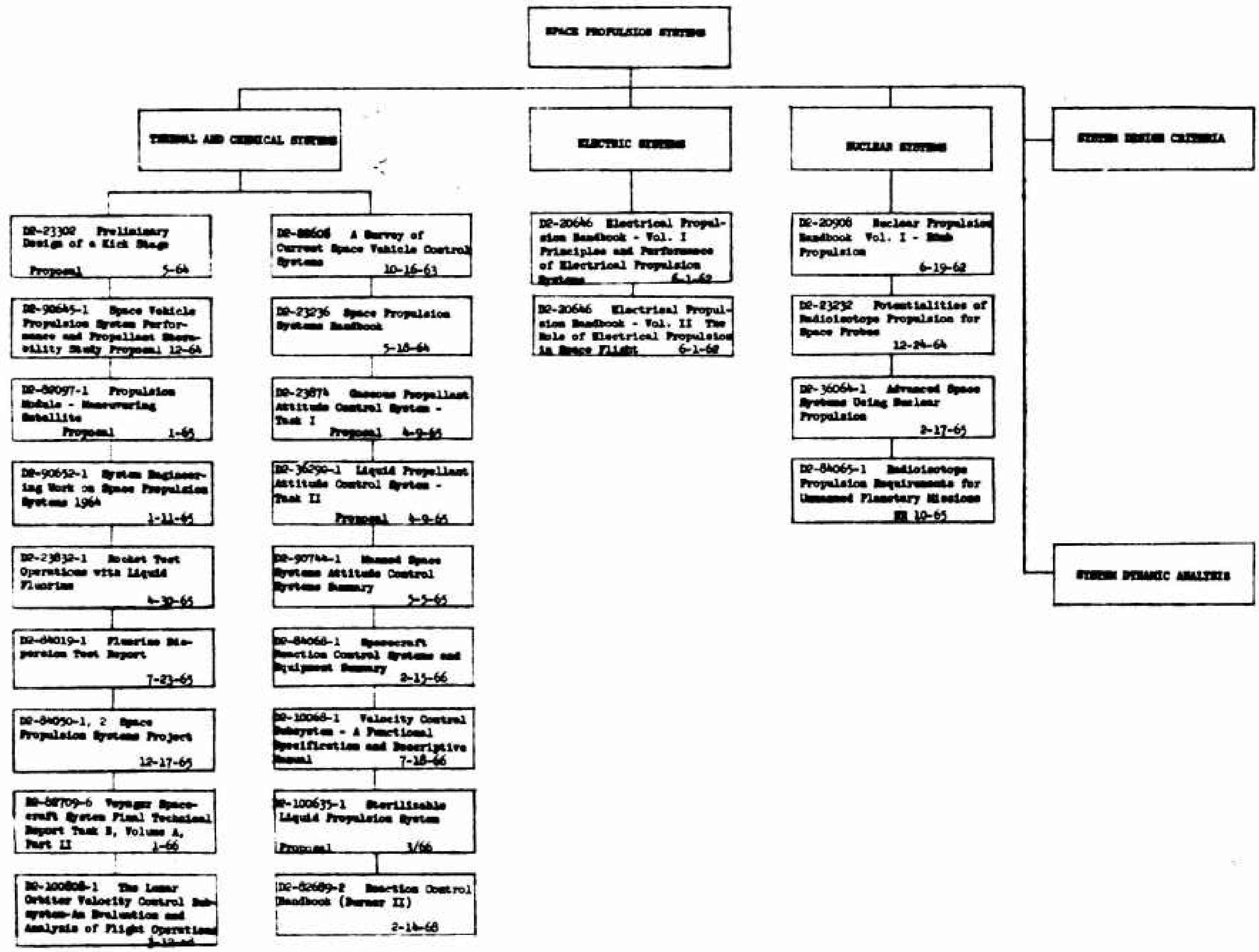

FIGURE 3.0-1 SPACE PROPULSION SYSTEMS

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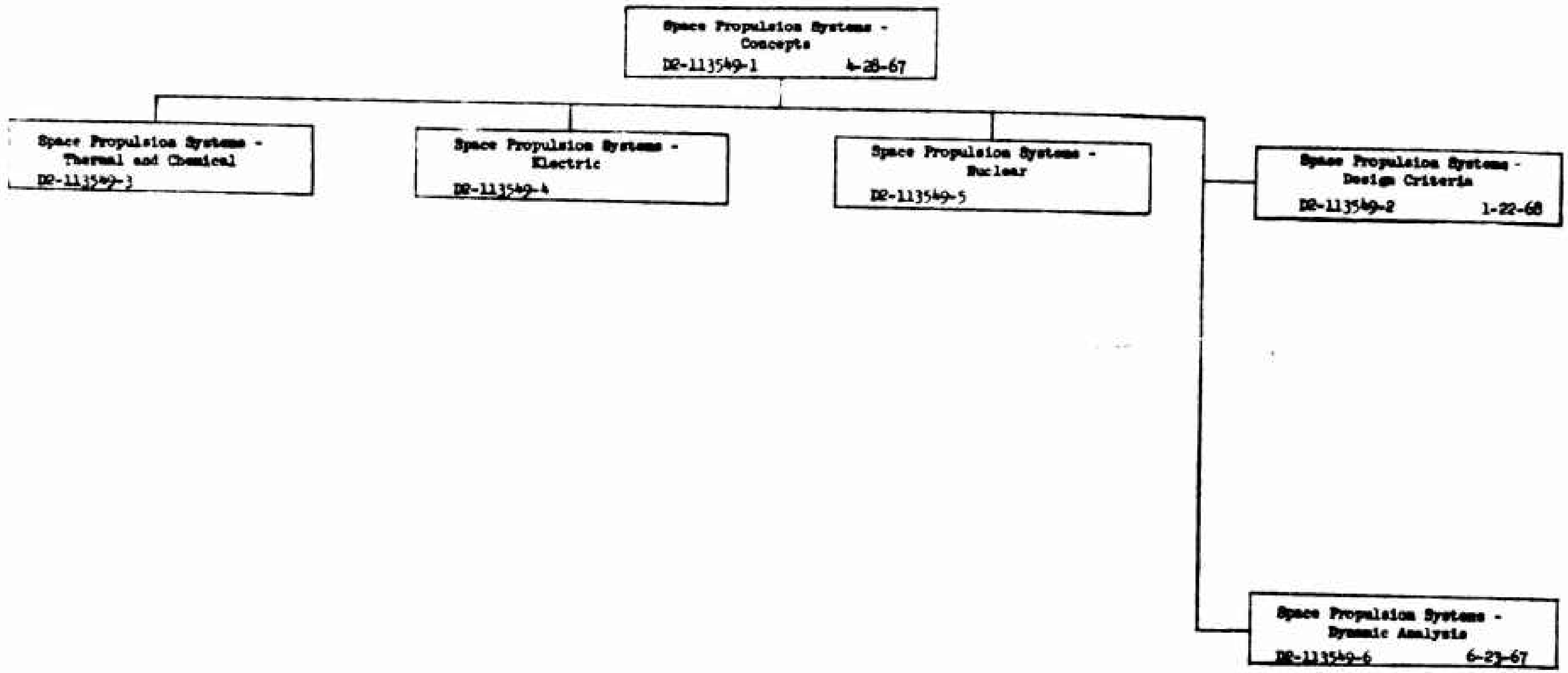

FIGURE 3.0-2 SPACE PROPULSION SYSTEMS

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ROCKET ENGINES

LIQUID PROPELLANT

- D2-12349 A 7090 Program for Computing the Thermal Conductivity and Viscosity of Gas Mixtures 1-3-62
- D2-12041 Heat Transfer in Ramjets and Liquid Rockets 5-23-63
- D2-23653-1 Performance and Physical Characteristics of Liquid Rocket Engines 11-19-64 *
- D2-23687-1 Reaction Control Jet Engine Test Facility Development and Engine Test Results 1-5-65
- D2-36335-1 Preliminary Design Handbook for Large Liquid Rocket Engines 1-5-66

SOLID PROPELLANT

- D2-9449 A Parametric Design Study of Solid Propellant Rocket Motors 1-15-61
- D2-12333 Preliminary Design of Solid Propellant Rockets 12-4-61
- D2-22055 Computer Programs for Large Solid Propellant Motor Design [illegible] 62
- D2-22057 Large Solid Propellant Grain Stress Analysis [illegible] 62
- D2-20456-1 Ignition and Thrust Termination Study 7-31-64
- D2-113583-1 Voyager Spacecraft System Final Technical Report - Task C Thrust Vector Control Study 7-67

ELECTRIC THRUSTORS

- D2-20907 A Stabilized Induction Process, Planar-Mirror Driven Plasma Accelerator 5-15-62
- D2-20960 Stabilized Shunt-Induction Electromagnetic Engine 6-62
- D2-20646-3 Electrical Propulsion Handbook Vol. III-Characteristics of Ion Propulsion 2-19-63
- D2-20646-4 Electrical Propulsion Handbook Vol. IV - Characteristics of Electrothermal Propulsion 6-26-63
- D2-84132-1 SOA in Electrical Propulsion-1965 7-18-66

NUCLEAR

- D2-20698 A Method of Computing Intermediate Nuclear Reactor Dimensions 5-27-62
- D2-20971 Some Properties of the Gaseous Homopole-Application to Fuel Retention in the Gaseous Core Nuclear Rocket 7-10-62
- D2-20908 Nuclear Propulsion Handbook - Vol. III Gaseous Fission Reactors 8-10-62
- D2-22049 System and Fuel for the Nuclear Rocket 10-18-62
- D2-22652 Gaseous Core Nuclear Propulsion 8-19-63
- D2-22816 Radioisotope Propulsion 12-16-63
- D2-84131-1 Radioisotope Propulsion SOA - 1965 1-10-66
- D2-84130-1 Graphite Core Nuclear Propulsion SOA - 1965 2-14-66

* Superseded by D2-114118-2 Vol. II

FIGURE 3.0-3 ROCKET ENGINES

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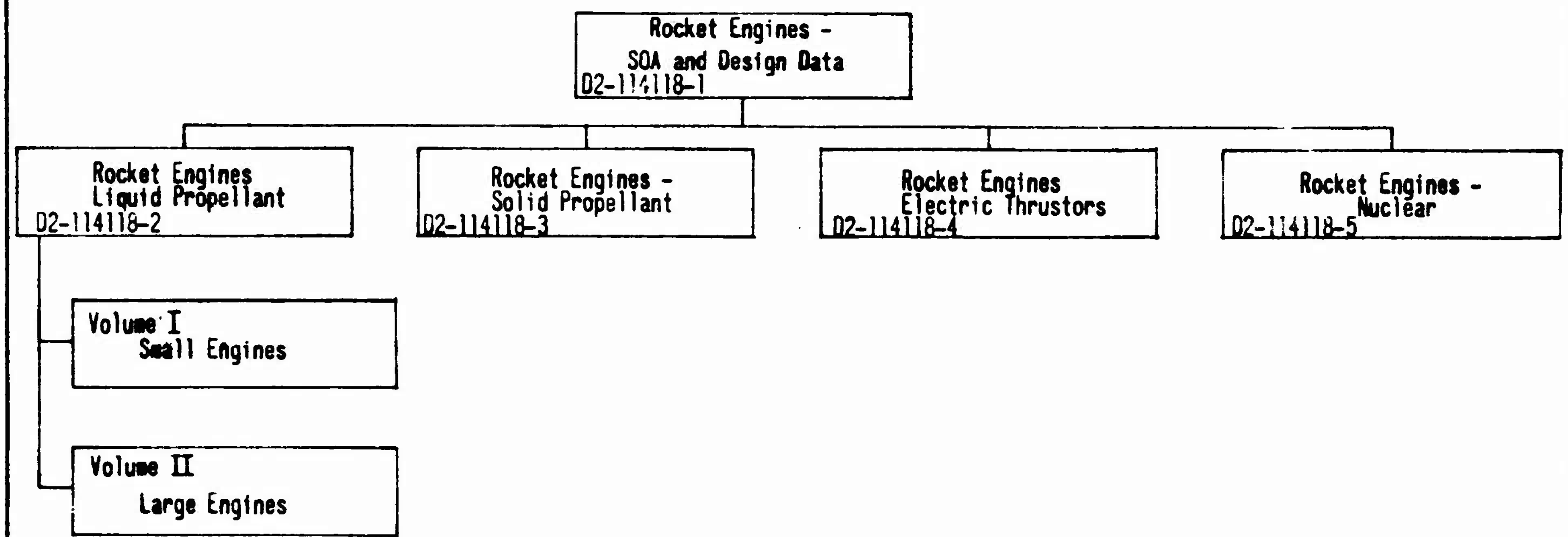

FIGURE 3.0-4 ROCKET ENGINES

4.0 MISSION REQUIREMENTS

Small rocket engines, or thrustors, are reaction devices used for translational and rotational control of spacecraft and boosters. Translational control is concerned with imparting a translational velocity in a specific direction. It is usually referred to as "velocity control" or "delta-v (ΔV) control". Rotational control, directed around a vehicle axis, is generally referred to as "reaction control" and is used in conjunction with other devices to provide vehicle attitude orientation and control. Velocity control and reaction control systems use similar components though their selection, design, installation and operating requirements may be considerably different.

4.1 VELOCITY CONTROL

Spacecraft velocity control systems are propulsion systems used for maneuvering, modifying trajectories, and for changing orbital characteristics. These applications entail certain unique features involving duty cycle, operating conditions, performance and the duration of space storage.

The number and duration of discrete operating sequences can usually be predicted from mission plans. For example, midcourse corrections are usually budgeted in a transfer trajectory to remove launch errors and to bias aiming point or arrival date. This budgeting involves relating the velocity corrections required, to propellant expenditures, while at the same time minimizing the number of maneuvers, commands and spacecraft functions required. Since velocity control systems usually operate for more than one second, thrustor operation is considered to be steady state. Steady state performance is desirable since it minimizes transient effects, permits design for peak performance, and allows the thrustor to reach operating temperature. Consequently, velocity control systems often make good use of the higher performance attained with monopropellant and bipropellant thrustors.

Propulsion system hardware is located, as much as possible, within the spacecraft. The thrustors, however, must protrude in some fashion from the spacecraft. Velocity control thrustors are usually closely coupled to the spacecraft along a major axis since the translational maneuvers they accomplis not require a moment arm. This eases the thermal control requirements concerned with preventing propellant freezing at the inlet lines. But it can complicate spacecraft thermal control requirements associated with engine, nozzle and exhaust plume heat loads.

Very small maneuvers are occasionally required from a velocity control system for midcourse correction, orbit trim, or similar uses. These small maneuvers require very small, precise and repeatable pulses (impulse bits) from the thrustors. These factors are controlled by thrustor size, valve response and valve location.

The spacecraft must be properly oriented prior to a translational maneuver, and subsequently reoriented for the coast phase. This is a function of the attitude control system (ACS). During the maneuver, engine thrust vector control (TVC) is required to prevent it from introducing undesired moments about the spacecraft center of gravity.

Velocity control propulsion systems are selected on the basis of performance, weight, duty cycle and lifetime requirements, reliability and hardware availability. Velocity control thrustors and their valves are designed or selected on the basis of performance, response, repeatability, weight, size, lifetime, reliability, thermal characteristics, availability and various materials consideration. Primary criteria generally are specific impulse, minimum impulse bit, availability and weight.

4.2 REACTION CONTROL

Reaction control systems (RCS) are spacecraft propulsion systems used to provide spacecraft pitch, yaw and roll control. They are also used for attitude positioning when conducting major maneuvers, directing sensors, or aiming photographic systems. Their unique requirements concern system response, accuracy, duty cycle and component location.

Reaction control system thrustors usually operate in a pulsed mode to impart small impulse increments to the spacecraft about a specific axis. Consequently they are particularly characterized by transient and pulse mode factors including thrust rise and decay times and impulse bit size and shape. Overall performance is also quite important.

Reaction control thrustors are operated by attitude control system (ACS) commands. This system processes information about spacecraft orientation and relates it to desired orientation and maneuver rate to issue thrustor "ON" and "OFF" signals. Hence, reaction control impulse bits are sized and scheduled against an assumed duty cycle based on anticipated disturbances, response rate limits, and scheduled spacecraft attitude positioning events. A "limit cycle" mode is usually used to accommodate induced disturbances by applying cycling response of the thrustors to a control band established by spacecraft pointing limits and control rate limits. These conditions are subsequently reduced to impulse bit size, duty cycle and propellant allowances. The pulse mode operation of reaction control thrustors with chemically reactive propellants yields lower performance than under steady-state conditions because of transient operation effects, engine thermal conditions, and single-versus-multiple design point considerations. The specific impulse of chemically

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reactive propellants approaches that of cold gas propellants with decreased pulse frequency and pulse width, as shown in Figure 4.2-1. Consequently, reaction control thrustors will not benefit from the higher performance propellants to the extent that velocity control thrustors do.

The reaction control system is usually installed in the spacecraft in a manner similar to that of velocity control systems. However, the thrusters are mounted at a distance from the spacecraft primary axes to provide moment arms for the rotational motions desired. Large moment arms permit small (lower thrust) engines which, in turn, are capable of smaller and more precise impulse bits. Propellant line length and electrical cabling increases correspondingly. These remote thrustor locations also require some thermal protection whenever propellants are used which can freeze.

Reaction control propulsion systems are selected on the basis of response, repeatability, performance, reliability and availability. Primary criteria are specific impulse, minimum impulse bit, pulse width, thrust rise and decay time, and operating tolerances.

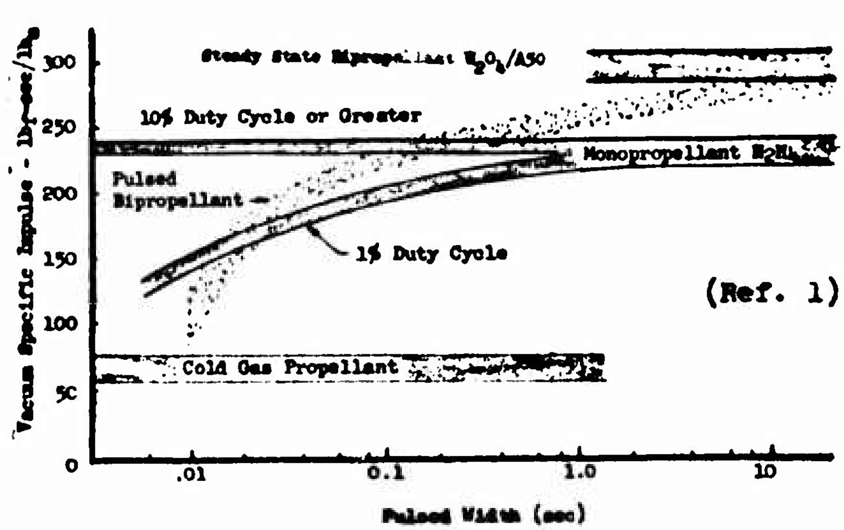

FIGURE 4.2-1 [illegible]

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4.3 APPLICATIONS FOR LOW ΔV THRUSTORS

Small rocket engines or thrustors, have been extensively used for velocity and reaction control of satellites and boosters. This will increase with the increasing exploration and exploitation of space. Thrustor selection is based on mission related factors (impulse, duty cycle measurements), system factors (thrust, weight, size), and engine characteristics (performance, response, induced environment). Hence, certain thrustor types are appropriate to particular applications. These circumstances are discussed below.

4.3.1 Thrust and Impulse Effects

Engines used for translational maneuvering are commonly larger and operate less frequently than do engines used for attitude stabilization. Their operating duration per firing is longer, so steady state conditions are usually attained. Impulse requirements for translational maneuvering are large, so the higher performance of steady state operation helps to minimize propellant expenditures. Transient characteristics are less important with these engines, so they are generally designed for high performance rather than response.

Translational engines are often related to thrust and total impulse as shown in Figure 4.3-1. This figure shows that propulsion applications involving 100 to 100,000 lb-sec impulse and 0.01 to 100 lb thrust can be further defined by mission and thrustor type. The boundaries to these regions are not hard and fast. They were derived from numerous system, mission, and engine studies some of which undoubtedly were resolved by other factors such as reliability. Nevertheless, this figure serves a useful function in simplifying a complex situation to the extent that application trends are apparent.

Impulse (i.e. propellant) requirements are a strong function of maneuver velocity. They increase with velocity capability to the extent that system weight becomes

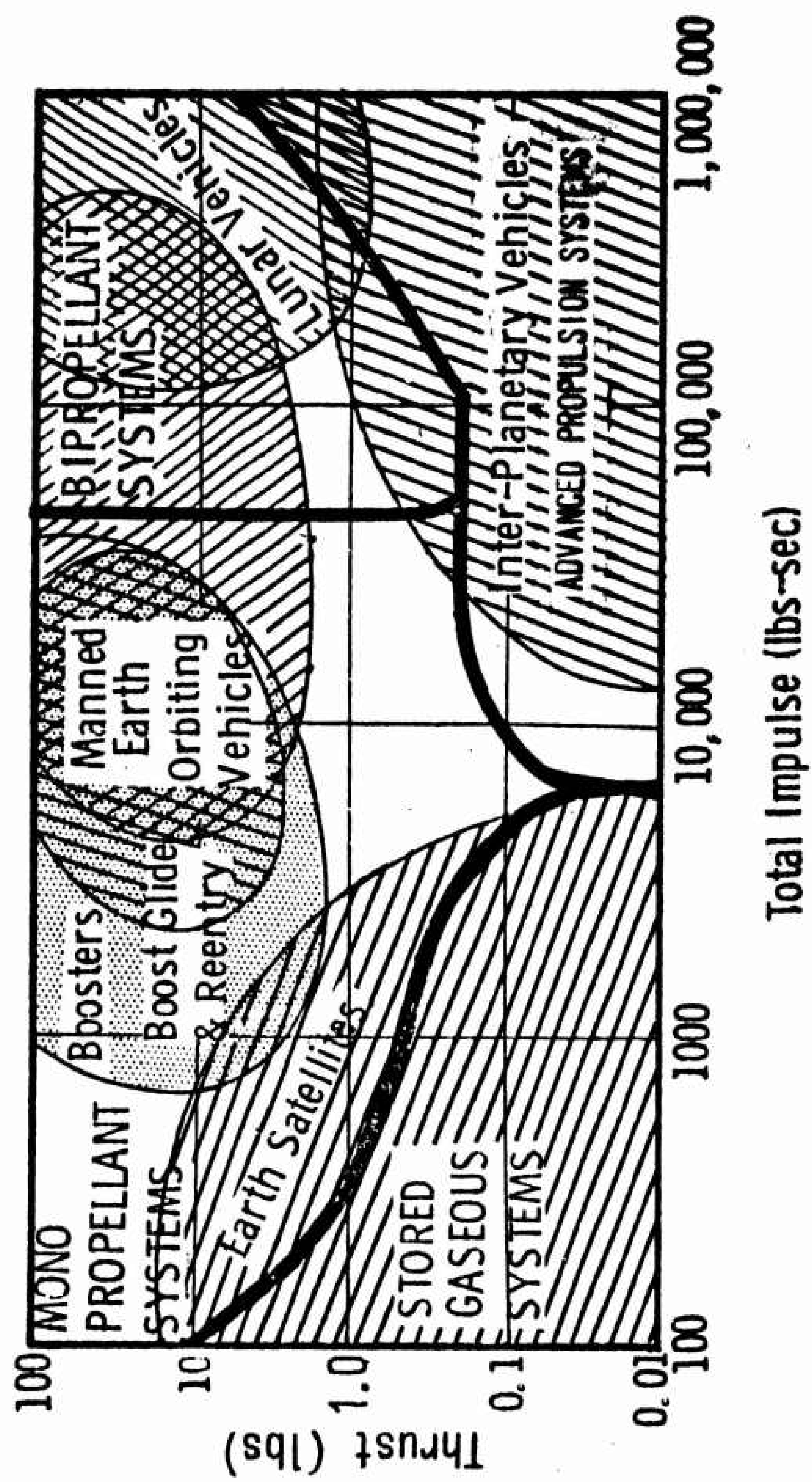

FIGURE 4.3-1 ROCKET ENGINE APPLICATION REGIONS - STEALY STATE

(Ref. 2)

increasingly a function of propellant loading. Under these circumstances, the higher performance propulsion systems may offer significant weight advantages. Propulsion systems are sometimes used for several maneuvers which may permit the use of higher energy systems. For example, Lunar Orbiter performed both midcourse correction (small ΔV) and orbit insertion (large ΔV) with the same bipropellant engine. Surveyor uses three bipropellant engines for midcourse correction, attitude control during solid motor firing, and vernier control for maneuvering and soft landing. Ranger and Mariner, however, used their monopropellant hydrazine systems only for midcourse correction maneuvers. The performance advantage of monopropellant engines in the Ranger program and bipropellant engines in Surveyor were enough to offset their development. Mariner hardware and technology was derived from Ranger. Lunar Orbiter used Apollo program bipropellant technology. Thus, velocity control systems used in the 150-4000 fps velocity and 550-2500 lb spacecraft weight range currently use the higher performance monopropellant or bipropellant engines since the technology, hardware, and space experience are available.

Impulse requirements are proportional to spacecraft size. It has been mentioned that propulsion system weight becomes increasingly a function of propellant weight with larger systems. Thus, propulsion systems for large spacecraft can derive a distinct weight advantage from using the higher performance monopropellant and bipropellant engines. The larger spacecraft currently occur in the manned Apollo and Manned Orbiting Laboratory (MOL) programs. Maneuvering and attitude control engines in the Mercury Spacecraft (Apollo Program) used monopropellant hydrogen peroxide thrustors. The succeeding Gemini spacecraft used bipropellant engines for maneuvering and attitude control as will the Apollo Command (CM), Service (SM) and Lunar (LM) modules.

Less impulse is usually required for attitude positioning and control than for performing major maneuvers. The impulse capability provided to satellites, especially earth satellites, is primarily for attitude control and is thus not large in magnitude. The pulsing duty cycle employed for reaction control reduces the performance advantage of monopropellant or bipropellant thrustors over stored gas thrustors (cold or heated gas). Also, the weight of small propulsion systems is strongly influenced by components which are not strongly weight-sensitive to thrust level or propellant loading such as standard components sizes, fittings, bosses, mounts, cabling and wiring. As a consequence, the lower performance cold gas systems have been competitive on a weight basis and have distinct advantages in simplicity and hardware availability. When velocity requirements increase, such as for velocity control purposes, a weight advantage accrues to monopropellant and bipropellant propulsion systems, since engine performance is highest when operating under steady-state conditions.

Monopropellant engines will increase in use as hardware experience, performance and availability increases. Hydrogen peroxide (H_2O_2) has been recently used as a monopropellant with good success on several satellites, including Syncom and Comsat. It had previously been used as a monopropellant for reaction control and auxiliary power on the X-1 and X-15 rocket aircraft, and the Centaur upper stage booster.

Boosters and maneuvering stages have frequently had pulsed bipropellant engines for reaction control using the primary propellants. Recently, monopropellant engines have been introduced into booster system use with the Burner II upper stage using hydrogen peroxide for reaction control. Significantly, the bipropellant attitude control and maneuvering engines on Titan II Transtage have been changed to monopropellant hydrazine engines using the Shell 405

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spontaneous catalyst.

4.3.2 Response and Impulse Bit Effects

Thrustors used for very small velocity corrections, attitude positioning, and attitude control are particularly affected by engine pulsing characteristics, especially thrust response and impulse bit size. Good pulsing characteristics are measured in terms of good repeatability, small minimum impulse bits, rapid thrust rise and decay times, high performance, and long operating life. Figure 4.3-2 is a generalized guide for selecting engines to be used in pulsing applications in which minimum bit size is important. This figure shows discrete regions of thrust and minimum impulse in which particular types of engines (i.e., stored gas, monopropellant or bipropellant) are appropriate. The boundary describing the lowest minimum impulse bit in each area indicates the best pulsing capability of close-coupled, fast response thrustor systems. Adjacent to these regions are arrows indicating the particular type of mission to which these engines are suited. This figure shows especially that unmanned earth orbital, lunar and interplanetary satellites are the spacecraft which require

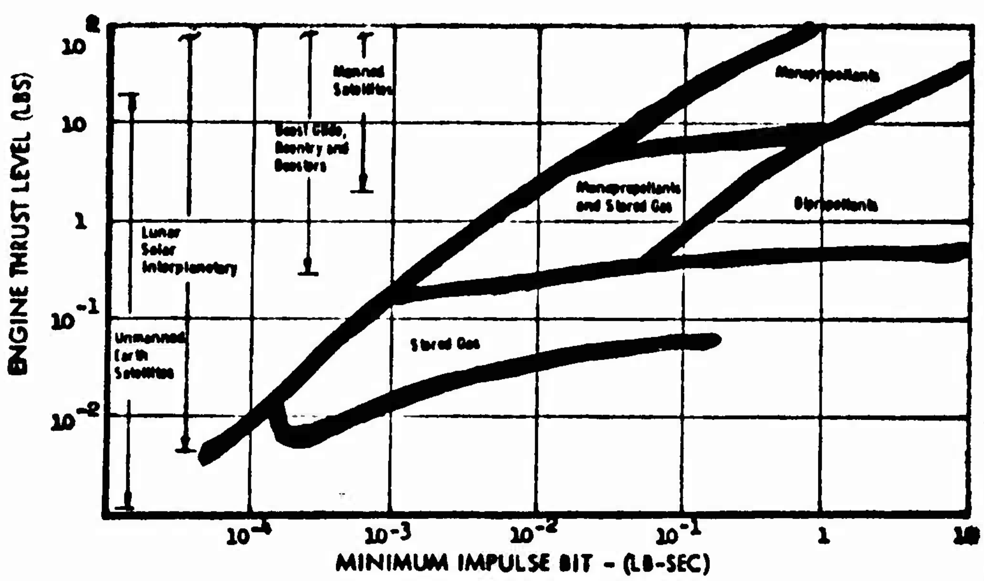

FIGURE 4.3.2 ROCKET ENGINE APPLICATION REGIONS-PULSED OPERATION

engines of less than 1 pound thrust and 10^{-2} lb-sec minimum impulse bit capability. Boosters, reentry and boost glide vehicles and manned satellites use larger engines which have larger minimum impulse bits. Future large unmanned satellites may use larger engines, but they may still need relatively small minimum impulse bits. Small engines will be used in applications involving the smaller satellites, low maneuver rates, extremely fine pointing requirements or where they are used in conjunction with other attitude stabilization devices.

Response characteristics are transient conditons involving chamber pressure, thrust, impulse and time during the start-up and shutdown phases. Transient effects are more pronounced with small pulses, slow acting valves, large thrust chambers, low propellant temperatures and, sometimes, with increased pulse spacing. Transient conditions including ignition delay, thrust build-up and thrust decay are shown in Figure 4.3-3. Some differences exist in the industry concerning the meaning of these terms, especially as they apply to different engine types. Ignition delay is used herein for that period of time consumed between initiation of the electrical ignition signal and first detection of a corresponding reaction in chamber pressure. This includes time increments involved in electrical and electro-mechanical delays, propellant transport lag and delays in chemical reaction initiated by hypergolic, electrical, thermal or catalytic means. Bipropellant engine manufacturers sometimes use the term "ignition delay" for what is actually the chemical reaction delay. Thrust build-up and decay periods are defined by the time required to reach a specific thrust level. They are usually given as a percent of maximum steady state thrust and measured in terms of equivalent chamber pressure.

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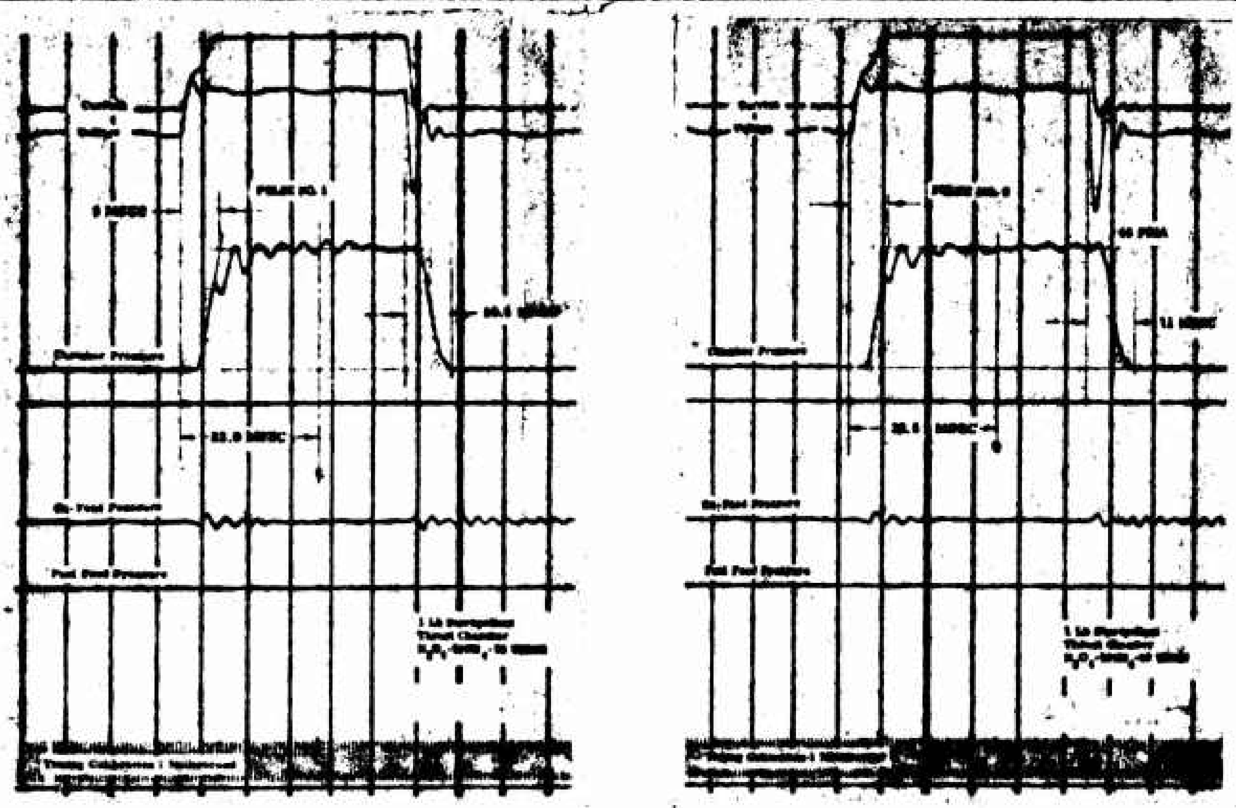

FIG. 4.3-3 TYPICAL ROCKET ENGINE TRANSIENT CONDITIONS (Ref. 3)

Figure 4.3-3 shows two pulses from a sequence of ten obtained with a small bi-propellant thrustor operated at a nominal duty cycle of 9%. This sequence shows transient characteristics and reproducibility of chamber pressure, propellant inlet pressure, and valve current and voltage as the valve is actuated for each pulse. The chamber pressure trace shows that eleven (11) msec were consumed between the application of valve voltage and reaching rated thrust. Approximately four (4) msec. of this were taken up by the electrical delay and valve poppet opening time. The remaining seven (7) msec. include the injector and feed tube filling time, propellant ignition delay, and the time required to fill the chamber volume with combustion gases. The shutdown transient consumed approximately eleven (11) msec. between removing valve power and attaining essentially 0% thrust. This time was consumed in electrical delay, valve poppet closing, emptying the "dribble volume", and emptying the chamber of gases.

Engine response characteristics are primarily controlled by the chamber design and the valve-chamber relationship. These characteristics depend on the valve actuation time, and the propellant volume contained between the valve and the injector ("hold-up volume"). When quick response is desired for rapid thrust build-up or pulsing operation, fast acting valves are closely mated to the engine to minimize propellant "hold-up volume." Such engines are referred to as "fast-acting" and "close-coupled." Thrust chamber volume is important because small thruster chambers have faster response characteristics. For quick response, chamber volume should be as small as possible, though not so small that performance is reduced through incomplete combustion. Chamber size is described in terms of L*, the chamber characteristic length, which is equal to the chamber volume per unit of nozzle throat area. Figure 4.3-4 shows how ignition delay increases with L*. Figure 4.3-5 shows how (1) performance decreases with decreasing L* and (2) that there is a performance difference in chemical engines designed for pulse-mode operation as opposed to those desired for single point, steady-state operation.

Characteristic length (L*) affects response in cold gas systems, but it does not affect performance since chemical reactions are not involved. Chamber temperature of cold gas thrustors is constant except as it is influenced by variations in state or source temperatures. Consequently, cold gas thrustor performance, though comparatively low, is constant and predictable throughout system operation, simplifying the design of attitude control logic.

Monopropellant engines do not behave in this fashion. Since they are designed to operate at elevated temperature, performance is variable until stable operating conditions exist. This happens when the engine and catalyst have been raised to operating temperature by heat obtained from the

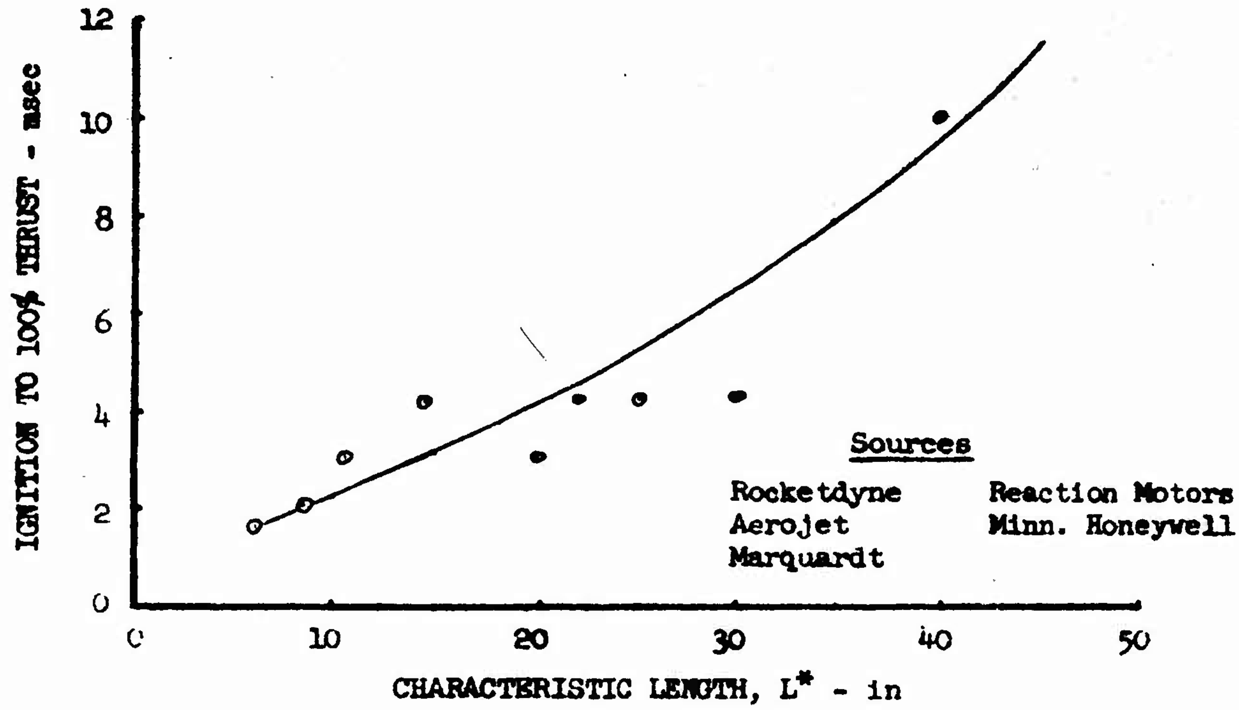

Figure 4.3-4 THRUST RESPONSE (Ref. 4)

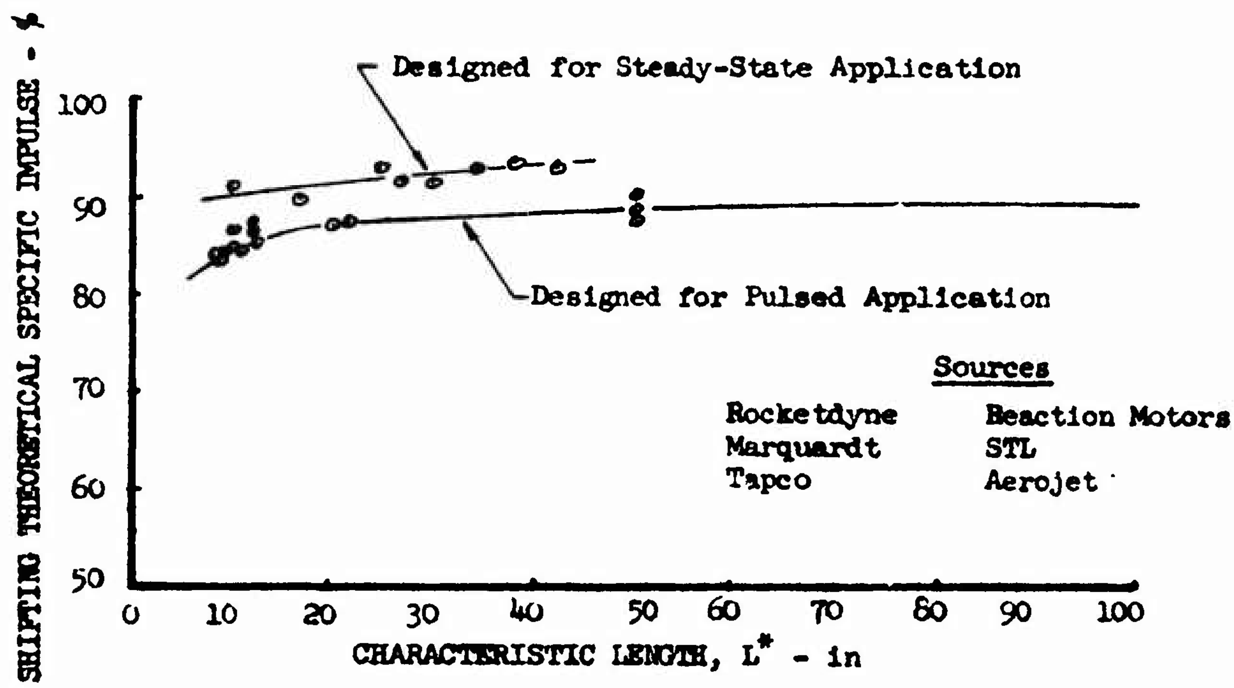

Figure 4.3-5 ENGINE PERFORMANCE STEADY-STATE (Ref. 4)

decomposition process. Performance is reduced, and thrust rise times extended, until stable operating conditions are attained. Thus, monopropellant engines do not exhibit uniform performance and response characteristics under all operating conditions.

Translational maneuvers are controlled by either a timer or an accelerometer. Maneuver rate data is used to control attitude orientation maneuvers. Timed maneuvers accumulate deviations from scheduled performance as a total error at end burn. Maneuvers controlled by accelerometer or rate data correct for these deviations accumulating only those errors attributable to instrumentation tolerance and thrust tailoff variations. An additional error may develop from interference in pulse scheduling when response is so slow that additional pulses are commanded before previously ordered pulses are detected. This is particularly appropriate to "cold" (ambient) monopropellant engines. Manuever error is greatest with timed maneuvers, being generally on the order of 10-30% for very small maneuvers ($\approx$0.10 m/sec) and 3-7% for large maneuvers ($\approx$1.0 m/sec). In most installations accelerometer control can be justified for maneuvers exceeding 0.1 to 10.0 m/sec.

Bipropellant engines lose little performance to engine warm-up since combustion temperatures are so much higher (5000-6000°F) than with monopropellants (1800-3300°F) that it is necessary to keep the chamber walls from overheating. These engines depend, to some extent, on film cooling by using a fuel rich barrier at the chamber walls. Since no catalyst is required there is little interior chamber mass to be brought up to operating temperature. Pulsed bipropellant engines do lose some performance due to mixture ratio variations, though impulse variations appear to be small and reasonably linear. When a bipropellant engine is required to operate in a pulse mode it is specifically designed for these

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conditions. It usually has lower steady state performance, as shown in Figure 4.3-5, but for a particular design, performance and response is consistent over a wide range of operating conditions. It can thus be used in timed, accelerometer controlled, and rate controlled maneuvers with relatively simple control logic.

Fast acting propellant valves are necessary to quick engine response. These valves must be capable of rapid and positive response. Direct acting solenoid valves or torque motor actuated valves are usually preferred. Typical valve actuation periods are in the 5-10 millisecond range.

Cold-gas thrustors have superior response capability in that fast acting, close-coupled hardware can be used, thrust characteristics are relatively insensitive to propellant or hardware temperature, and simple control logic can be used. Response of cold gas engines is limited by valve capability which is currently on the order of 5-7 milliseconds. Transport lag can be as brief as 1-3 milliseconds. Time to 90% thrust then can be less than 7 milliseconds. Repeatability of cold gas engines is quite high.

Monopropellant systems have larger hold-up volumes due to valve and injector design requirements so that transport lag with the liquid monopropellant is somewhat higher. Monopropellant engines will, in pulsing operation, be somewhat less repeatable since the volume uncertainties involve liquids instead of gases, and response characteristics are influenced by thermal conditions. Response to 90% thrust exceeds 7 milliseconds for small monopropellant thrustors designed for pulsing operation and operated with ambient propellants and a hot catalyst bed. Performance is higher than with most cold gas engines.

Bipropellant systems have hold-up volume and hydraulic lag effects in both the fuel and oxidiser circuits and, in addition, they require some degree of mixture ratio control. In small engines, transport lag may consume 4-5 milliseconds. Smaller, fast-response, close-coupled bipropellant engines are capable of 15-25 milliseconds response to 90% thrust.

4.3.3 Packaging Effects

Propulsion system installed volume may, at times be an important factor due to limitations imposed by the booster, shroud, spacecraft structure, or experiments. Cold gas systems are inferior in this regard since a relatively large amount of propellant is stored in the gaseous state. However, cold gas equipment may be easier to position on the spacecraft since there is no danger of propellant freezing. Additionally, cold gas thrustors have minimum thermal interference with adjacent spacecraft equipment.

5.0 ROCKET ENGINE PERFORMANCE

The basic steady-state performance equations for rocket engines are developed in detail in numerous textbooks among which are References 5 and 6. These equations are briefly developed herein to support subsequent discussions of delivered steady-state performance, and of transient performance.

5.1 THEORETICAL STEADY STATE PERFORMANCE

The theoretically ideal rocket is based on adiabatic, steady state flow of homogeneous, compositionally invariant propellants (in chemical equilibrium) which obey the perfect gas laws, which develop no friction, and which have uniform, axially directed velocity. These assumptions permit one-dimensional analyses of the rocket engine.

Rocket engine thrust is defined by a momentum plus pressure-time-area term:

$$F = \dot{m} V_e + [P_e - P_a] A_e \tag{5.1-1}$$

The principle of conservation of energy is used for gas velocity at the nozzle exit. For an adiabatic gas expansion (no heat transfer between the gas and nozzle) having no friction between the gas and the nozzle wall, the conservation of energy requires the decrease in gas enthalpy to be equal to the increase in gas kinetic energy. That is,

$$h_c - h_e = \Delta h = [V_e^2 - V_c^2] / 2g_c \tag{5.1-2}$$

For a perfect gas

$$h_c - h_e = \Delta h = C_p [T_c - T_e] \tag{5.1-3}$$

$$C_p = \frac{\gamma R}{\gamma - 1} \tag{5.1-4}$$

For an isentropic process,

$$\frac{T_c}{T_e} = \left[\frac{P_c}{P_e}\right]^{\frac{\gamma-1}{\gamma}} \qquad (5.1\text{-}5)$$

Assuming that the chamber cross-sectional area is large compared to the throat area, v_c is small and can be neglected:

$$V_e = \left\{\left[\frac{2g\gamma}{\gamma-1}\right] RT_c \left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]\right\}^{1/2} \qquad (5.1\text{-}6)$$

The mass flow rate equals:

$$\dot{m} = \frac{\dot{\omega}}{g_c} = \frac{\rho_T}{g_c} A_t V_t \qquad (5.1\text{-}7)$$

The throat velocity v_t is equal to the speed of sound at the throat, which is

$$V_t = \left[\left(\frac{2g\gamma}{\gamma+1}\right) RT_c\right]^{1/2} \qquad (5.1\text{-}8)$$

The flow density at the throat is related to the chamber density by

$$\frac{\rho_t}{\rho_c} = \left[\frac{2}{\gamma+1}\right]^{1/(\gamma-1)} \qquad (5.1\text{-}9)$$

Using the ideal gas relationship,

$$\rho_c = \frac{P_c}{RT_c} \qquad (5.1\text{-}10)$$

and substituting equations (5.1-8) and (5.1-9) into equation (5.1-7) gives

$$\dot{m} = \left[\frac{P_c A_t \gamma}{(\gamma g_c R T_c)^{1/2}} \left(\frac{2}{\gamma+1}\right)^{\frac{(\gamma+1)}{2(\gamma-1)}}\right] \qquad (5.1\text{-}11)$$

Combining equations (5.1-6) and (5.1-11) into (5.1-1)

$$F = P_c A_t \gamma \left[\frac{2}{\gamma+1}\right]^{\frac{(\gamma+1)}{2(\gamma-1)}} \left\{\frac{2}{\gamma-1}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]\right\}^{1/2} + P_e A_e \qquad (5.1\text{-}12)$$

Equation 5.1-12 is generalized, non-dimensionally, to:

$$C_F = \frac{F}{P_c A_t} = \gamma \left[\frac{2}{\gamma+1}\right]^{\frac{\gamma+1}{2(\gamma-1)}} \left\{\left(\frac{2}{\gamma-1}\right)\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]\right\}^{1/2} + \frac{P_e A_e}{A_t P_c} \qquad (5.1\text{-}13)$$

The term A_e/A_t is evaluated by applying the principle of conservation of matter:

$$\rho_t A_t V_t = \rho_e A_e V_e \qquad (5.1\text{-}14)$$

Noting that,

$$\frac{\rho_t}{\rho_e} = \frac{\rho_t}{\rho_c} \cdot \frac{\rho_c}{\rho_e} \qquad (5.1\text{-}15)$$

and applying the isentropic flow relations,

$$\frac{\rho_t}{\rho_c} = \left(\frac{P_t}{P_c}\right)^{1/\gamma} \quad \text{and} \quad \frac{P_c}{P_t} = \left(\frac{\gamma+1}{2}\right)^{\gamma/(\gamma-1)} \qquad (5.1\text{-}16)$$

the term ρ_t/ρ_e becomes

$$\frac{\rho_t}{\rho_e} = \left(\frac{P_t}{P_c}\right)^{1/\gamma}\left(\frac{P_c}{P_e}\right)^{1/\gamma} = \left(\frac{2}{\gamma+1}\right)^{1/(\gamma-1)}\left(\frac{P_c}{P_e}\right)^{1/\gamma} \qquad (5.1\text{-}17)$$

Combining equations (5.1-17), 5.1-3), (5.1-6) and (5.1-14), the area ratio equals

$$\frac{A_e}{A_t} = \left(\frac{2}{\gamma+1}\right)^{1/(\gamma-1)}\left(\frac{P_c}{P_e}\right)^{1/\gamma}\left\{\left(\frac{\gamma-1}{\gamma+1}\right)\left[\frac{1}{1-\left(\frac{P_e}{P_c}\right)^{(\gamma-1)/\gamma}}\right]\right\}^{1/2} \qquad (5.1\text{-}18)$$

Thus

$$C_{F_i} = \gamma \left\{ \left(\frac{2}{\gamma-1}\right)\left(\frac{2}{\gamma-1}\right)^{\frac{\gamma+1}{\gamma-1}} \left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]\right\}^{1/2} + \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}} \left(\frac{2}{\gamma+1}\right)^{\frac{1}{\gamma-1}} \left\{ \left(\frac{\gamma-1}{\gamma+1}\right)\left[\frac{1}{1-\left(\frac{P_e}{P_c}\right)^{\gamma-1/\gamma}}\right]\right\}^{1/2} \quad (5.1\text{-}19)$$

A plot of this equation is shown in Figure 5.1-1. It contains a velocity thrust coefficient (C_{F_V}) and a pressure thrust coefficient (C_{F_P}), such that:

$$C_{F_i} = C_{F_V} + C_{F_P} \quad (5.1\text{-}20)$$

where:

$$C_{F_P} = \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}} \left(\frac{2}{\gamma+1}\right)^{\frac{1}{\gamma-1}} \left\{ \left(\frac{\gamma-1}{\gamma+1}\right)\left[\frac{1}{1-\left(\frac{P_e}{P_c}\right)^{\gamma-1/\gamma}}\right]\right\}^{1/2} \quad (5.1\text{-}21)$$

$$C_{F_V} = \gamma \left\{ \left(\frac{2}{\gamma-1}\right)\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}} \left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]\right\}^{1/2} \quad (5.1\text{-}22)$$

Specific impulse, I_{sp} , equals:

$$I_{sp_i} = \frac{F}{\dot{m}} \quad (5.1\text{-}23)$$

Combined with (5.1-11), (5.1-13), this becomes:

$$I_{sp_i} = \frac{F}{\dot{m}} = \frac{C_{F_i}}{\gamma} \left[\frac{\gamma K T_c}{g\left(\frac{2}{\gamma+1}\right)^{(\gamma+1)/(\gamma-1)}}\right]^{1/2} \quad (5.1\text{-}24)$$

Equation (5.1-24) can be further simplified to

$$I_{sp_i} = \frac{C_{F_i}}{W} \left[\frac{T_c}{m}\right]^{1/2} \quad (5.1\text{-}25)$$

by defining a ~~weight~~ flow parameter, W, as

$$W = \left[\frac{\gamma g \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}{K_c}\right]^{1/2} \quad (5.1\text{-}26)$$

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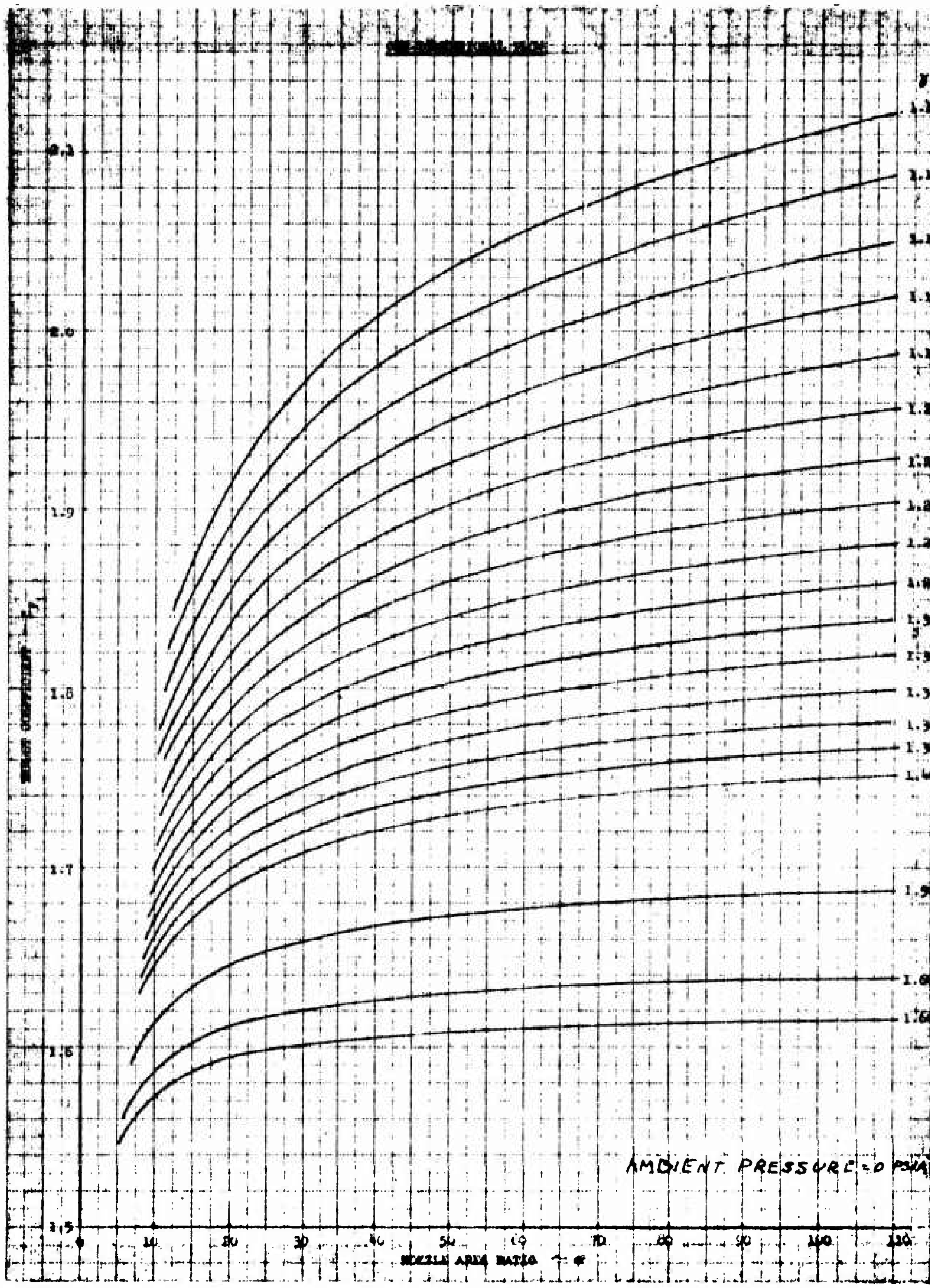

Figure 5.1-1 THRUST COEFFICIENT AS A FUNCTION OF GAMMA

so that

$$\dot{m} = W P_c A_t \left[\frac{m}{T_c} \right]^{1/2} \tag{5.1-27}$$

Equation 5.1-19 is shown graphically in Figure 5.1-1 for specific heat ratios of 1.1 to 1.66. Equation 5.1-26 is shown graphically in Figure 5.1-2 for specific heat ratios from 1.1 to 1.7. These equations assumes a constant specific heat ratio during the expansion process. In general, this is a valid assumption because the effects of temperature and pressure on specific heat ratio are small.

5.2 DELIVERED STEADY-STATE PERFORMANCE

Delivered performance differs from theoretical values because actual conditions vary from those assumed in the ideal case. Primary variations can be attributed to boundary layer effects, nozzle divergence angle effects, and non-homogenous, chemically changing propellant conditions during the flow process.

5.2.1 BOUNDARY LAYER EFFECTS

The effect of boundary layer growth on nozzle performance is two-fold:

a. The average gas exhaust velocity is reduced because of friction between the gas and the nozzle wall in the boundary layer.

b. The effective area ratio of the nozzle is reduced by growth of the boundary layer. The thickness of the boundary layer is characterized by the displacement thickness, δ^*.

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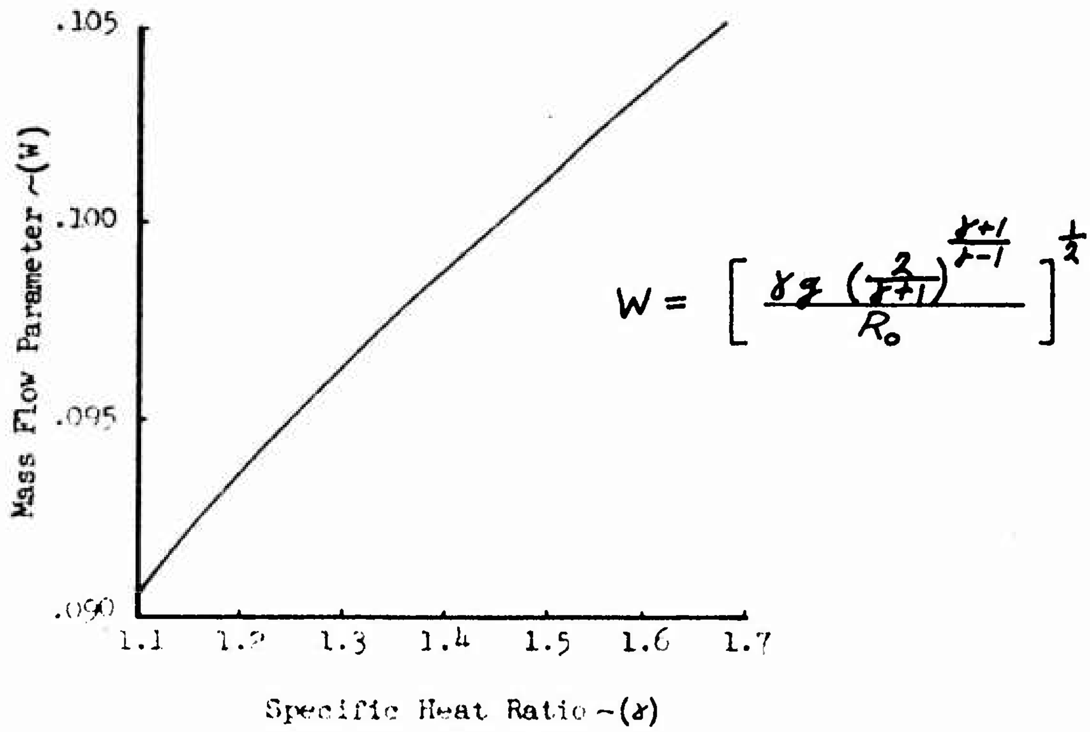


FIGURE 5.1-2 MASS FLOW PARAMETER

Nozzle flow with a boundary layer is schematically illustrated below:

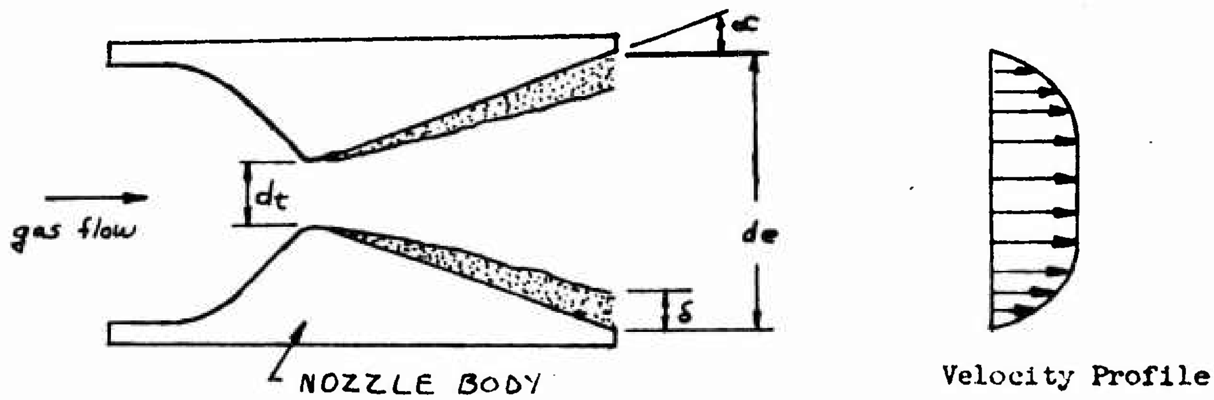


FIGURE 5.2-1 NOZZLE FLOW WITH A BOUNDARY LAYER

An estimate of nozzle performance can be obtained by calculating the boundary layer displacement thickness, and the total viscous drag force at the nozzle wall. For design purposes, the use of the correction factors C_V and C_D is most convenient. Values of C_V and C_D can be estimated from the experimental data summarized in Figures 5.2-2 and 5.2-3. The correction factors C_V and C_D are applied to the ideal performance factors in the following manner:

$$C_{F_{actual}} = \lambda\, C_{F_V} \cdot C_V \cdot C_D + C_{F_P} \qquad (5.2\text{-}1)$$

$$\dot{M}_{actual} = \dot{M}_i \cdot C_D \qquad (5.2\text{-}2)$$

The pressure thrust coefficient, C_{F_P}, is a small part of the total thrust coefficient, being less than 2% of the total thrust coefficient for area ratios above approximately 50:1, at $\gamma = 1.4$. Therefore, the velocity, divergence, and discharge coefficients can be applied to the total ideal thrust coefficient, C_{F_i}, without significant error. That is:

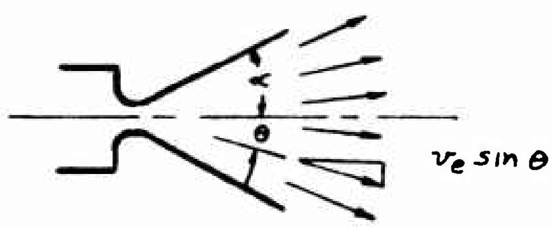
$$C_{F_{actual}} \cong \lambda\, C_{F_i} \cdot C_V \cdot C_D \qquad (5.2\text{-}3)$$

The difference between ideal and actual nozzle performance is primarily the result of boundary layer effects, so the velocity and discharge coefficients are correlated by the throat Reynold's number, as shown in Figures 5.2-2 and 5.2-3. The throat Reynold's number is calculated from:

$$Re_t = \frac{\dot{m}}{(\mu \pi)/(4 d_t)} \qquad (5.2\text{-}4)$$

5.2.2 DIVERGENCE ANGLE EFFECTS

The nozzle divergence correction is necessary because of the divergence of exit velocity vectors from the axial direction, represented below:



Although the velocity vectors have a magnitude equal to the ideal one-dimensional value v_{e_i}, the component (v_e sin θ) produces no useful axial thrust. The correction factor for nozzle divergence is designated (λ), and is evaluated as:

$$\lambda = 1/2\,(1 + \cos \alpha) \qquad (5.2\text{-}5)$$

where α is the nozzle divergence angle shown above. The divergence angle correction factor is applied to the velocity term in the thrust coefficient such that:

$$C_F = \lambda C_{F_V} + C_{F_P} \qquad (5.2\text{-}6)$$

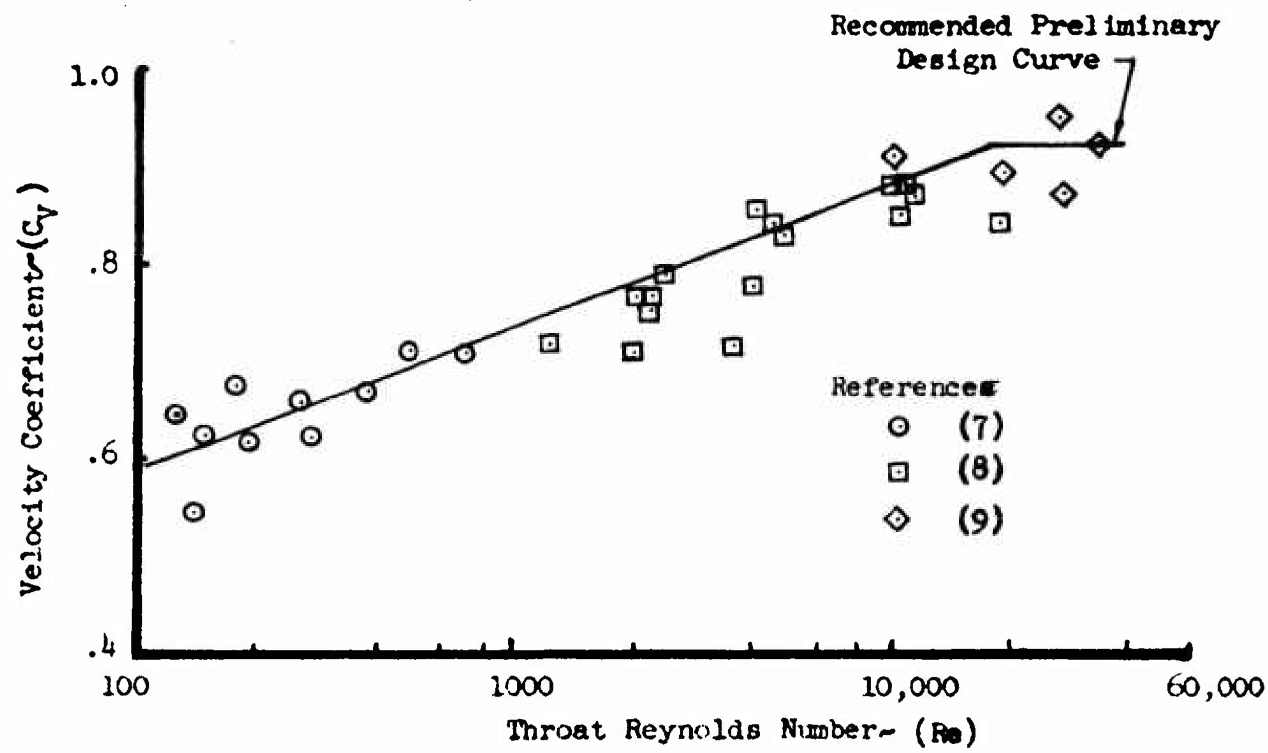


FIGURE 5.2-2 VELOCITY COEFFICIENT

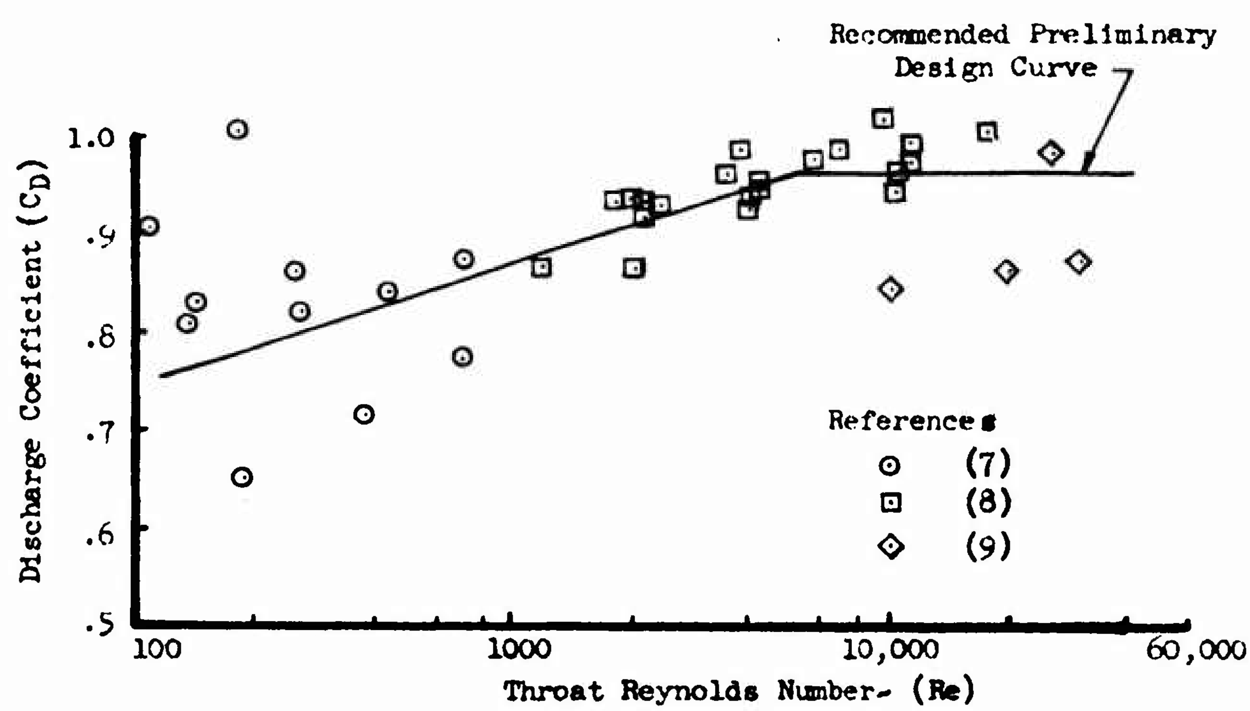


FIGURE 5.2-3 DISCHARGE COEFFICIENT

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Figure 5.2-4 shows the effect of nozzle divergence angle, α, on thrust coefficient as a function of nozzle expansion ratio, and specific heat ratio, and divergence angle of conical nozzles.

5.2.3 PROPELLANT AND GAS EFFECTS

In the real case, actual performance also differs from "ideal" values because the propellants are not really homogenous, compositionally invariant and in chemical equilibrium throughout the chamber and nozzle. Monopropellant engines are reactors designed to change propellant condition by catalysis. This process permits propellants to exist in various conditions within the chamber. Additionally, dissociation may also follow this process, as with neat hydrazine (N_2H_4). Bipropellant engines are designed to promote chemical reaction of fuel and oxidizer within the chamber. Obviously, compositional variations exist throughout the mixing and chemical reaction process which, when related to time, represents different locations within the engine. Fuel rich flow is usually provided to cool the chamber walls. Also, recombination can accompany these processes. Since these effects are related to engine design they will be discussed additionally under the particular engine involved.

Certain propellants contain water vapor as an exhaust product. Vapor condensation has been suggested as a source of performance variations from that calculated for "ideal" conditions. However, condensation effects on nozzle performance cannot now be predicted with confidence in view of the uncertainties involved in experimental efforts conducted so far. Further elaboration on this subject is available in References 7, 9 and 10. Until this effect is resolved, it is recommended that the magnitude of condensation effects be neglected in performance estimates.

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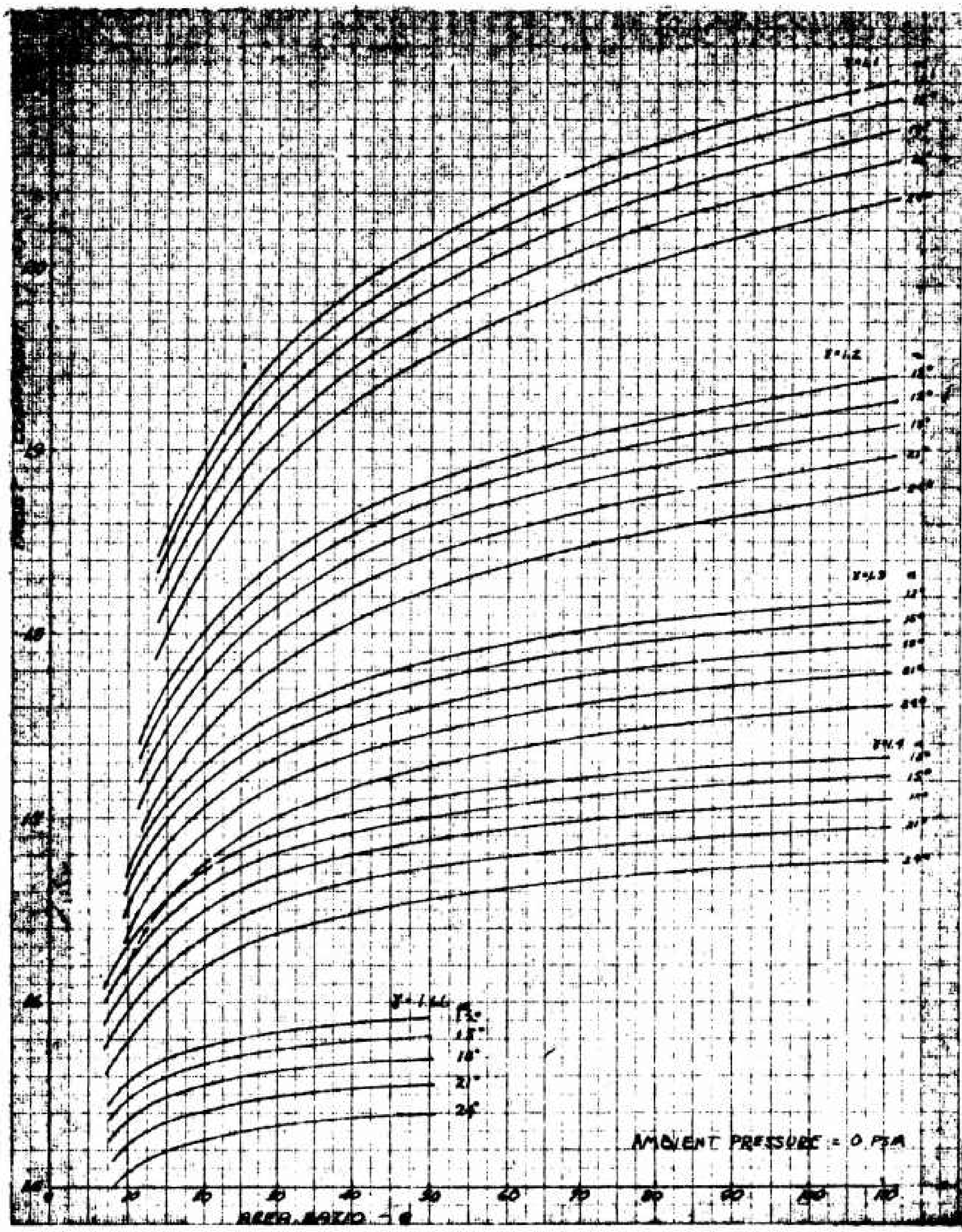

Figure 5.2-4 DIVERGENCE ANGLE EFFECTS ON THRUST COEFFICIENT

5.3 TRANSIENT PERFORMANCE

An analytical procedure is presented herein for determining transient performance, based on the analysis and experiments of Greer and Griep (Reference 9) and assuming

1) Constant density flow thru the valve orifice occurs
2) Choked flow at both the valve orifice and the thruster throat
3) Instantaneous valve opening at $\theta = 0$.

The following expressions for the pressure rise and decay transients are obtained:

$$\frac{P_c}{P_{ss}} = 1-(y_0+B)e^{-2\tau} + B(y_0+B)e^{-\tau} - B^2 \qquad (5.2\text{-}7)\ \text{(rise)}$$

$$\frac{P_c}{P_{ss}} = \left[1 - a_2(1-\gamma)\theta\right]^{\frac{2\gamma}{1-\gamma}} \qquad (5.2\text{-}8)\ \text{(decay)}$$

where

P_c = instantaneous chamber pressure

P_{ss} = steady-state, plenum chamber pressure

y_0 $= \left[1 - P_c/P_{ss}\right]^{1/2} = 1$ for vacuum operation (5.2-9)

B $= \frac{A_o}{A_t}\left(\frac{2}{\gamma}\right)^{1/2}\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{2(\gamma-1)}}$ (5.2-10)

τ = dimensionless time factor

$$= \theta\,\frac{A_t a^*}{2V_c}\left[\frac{2}{\gamma+1}\right]^{\frac{\gamma+1}{2(\gamma-1)}} \qquad (5.2\text{-}11)$$

a^* = sonic velocity

γ = specific heat ratio

θ = time

A_t = nozzle throat area

A_o = valve orifice flow area

V_c = volume of thruster plenum chamber

The thruster rise time, neglecting valve transient effects, is the time required to reach $P_c/P_{ss} = 1$, or,

$$\theta_r = -\frac{B}{a_1} \log \left[\frac{B}{y_0 + B} \right] \qquad (5.2\text{-}12)$$

The decay time required for the pressure to fall from P_{ss} to P_a, is

$$\theta_d = \left[\left(\frac{P_a}{P_{ss}} \right)^{\frac{1-\gamma}{2\gamma}} - 1 \right] \left(\frac{1}{a_2 (\gamma - 1)} \right) \qquad (5.2\text{-}13)$$

where

$$a_1 = \frac{A_o a^*}{2 V_c} \left(\frac{2}{\gamma} \right)^{1/\gamma} \qquad (5.2\text{-}14)$$

$$a_2 = \frac{A_t a^*}{2 V_c} \left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma+1}{2(\gamma-1)}} \qquad (5.2\text{-}15)$$

The preceding equations show that the thruster rise and decay times can be decreased by:

1) minimizing the plenum chamber volume, V_c
2) making the valve orifice area, A_o, large compared to the nozzle throat area, A_t.

Note that this analysis does not include the effects of a finite valve opening time, or the effects of electrical delays between application of an opening command and motion of the valve poppet. For well-designed thrusters, these delays are of the same magnitude as the thruster pneumatic delays. As a result, the transient response analysis must include the valve characteristics as well as the thrust characteristics.

6.0 THRUSTORS FOR LOW ΔV APPLICATIONS

6.1 COLD GAS THRUSTORS

6.1.1 GENERAL

"Cold" gas propulsion systems (Figure 6.1-1) are popular, state-of-the-art systems drawing from a large variety of space-qualified, "off-the-shelf" hardware. Such new component development as is necessary for particular applications involves minimum development time and expense. These systems are relatively simple, reliable, low cost and easy to develop items which still deliver adequate performance and very small, precise and repeatable impulse bits. The principal disadvantages of cold gas systems involve low specific impulse, hence high propellant weight, and relatively high storage system weight to store the propellant as a gas. Some weight reduction is possible by using propellants which are liquid at ambient conditions such as propane or ammonia.

Thrustors for cold gas systems are simple in design, consisting of a gas plenum chamber, nozzle, solenoid gas valve and an inlet line. Figure 6.1-2 shows a typical cold gas thrustor of the type used with satisfactory results throughout the Lunar Orbiter program. The principal variation to this design involves mounting the plenum and nozzle in some installations in such a way that the thrust vector is at an angle with respect to the gas supply vector to the plenum.

6.1.2 PRINCIPLE OF OPERATION

Cold gas propulsion systems rely upon gases stored at higher than environmental pressures to produce thrust upon expansion through a plenum-nozzle arrangement. The gaseous propellant is usually stored in high pressure (3000 - 4000 psia) gas bottles at ambient conditions. They are thus

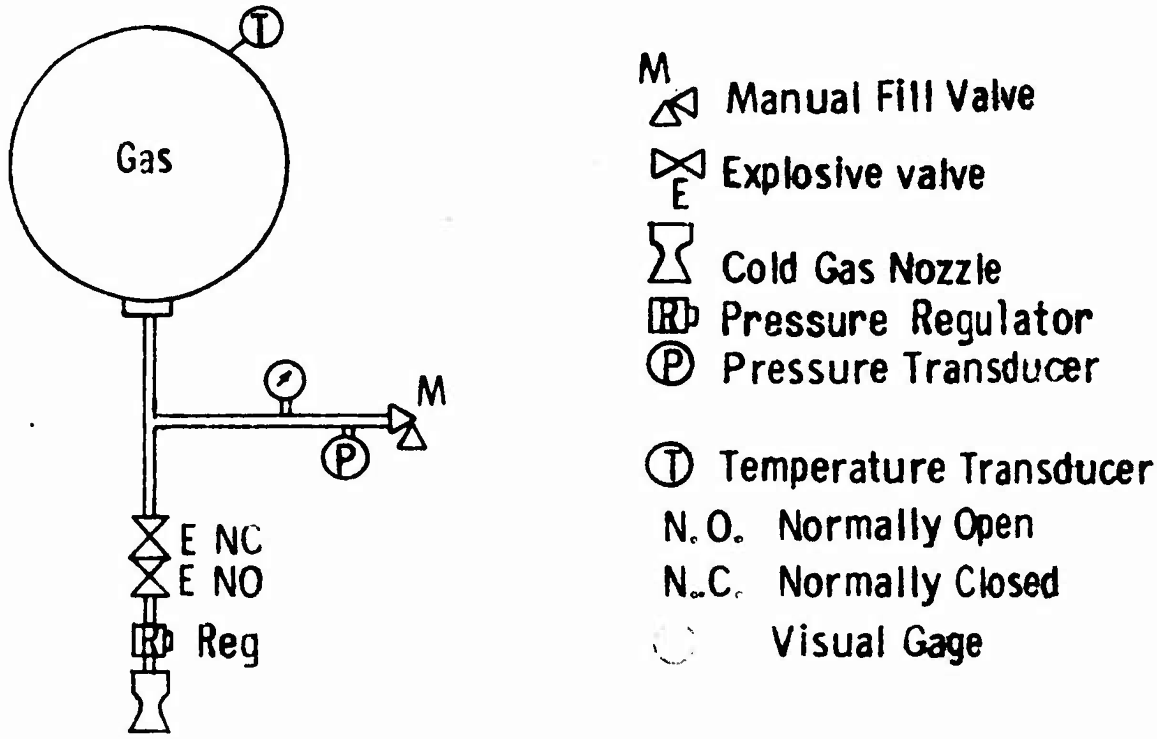

FIGURE 6.1-1. TYPICAL COLD GAS SYSTEM

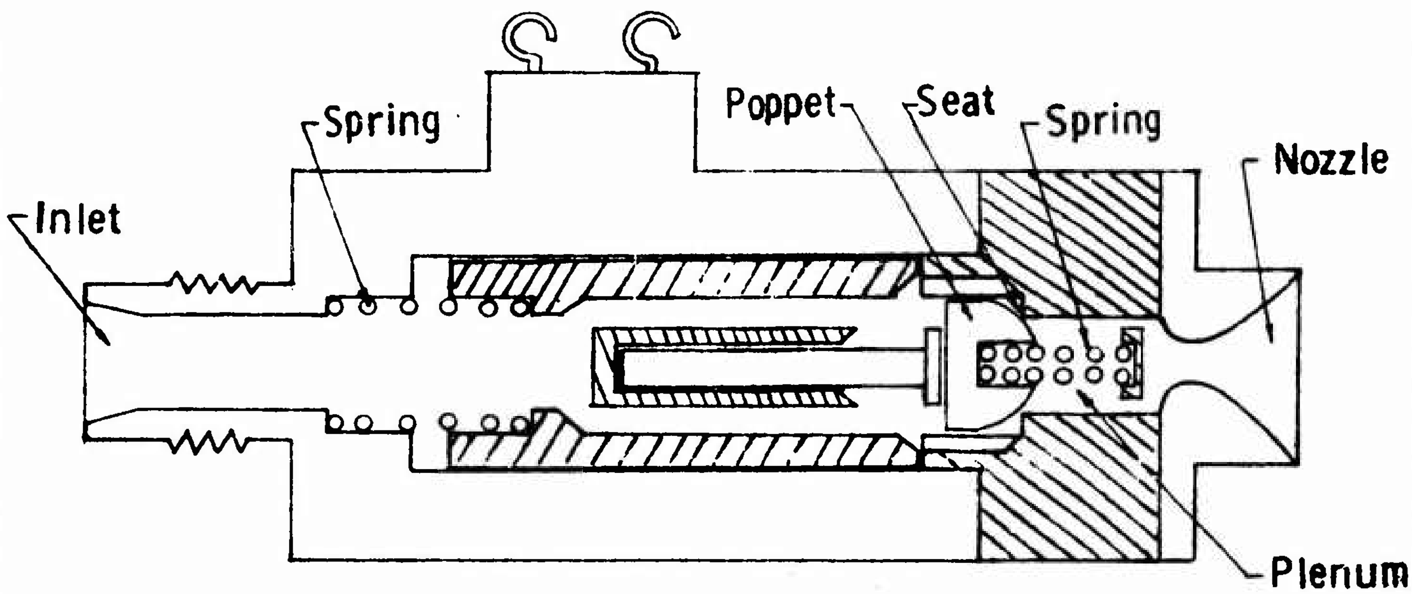

FIGURE 6.1-2. TYPICAL COLD GAS THRUSTOR

(Ref. 11)

referred to as "cold gas" systems, as opposed to "heated gas" systems in which thermal energy is added to the propellant prior to expansion.

In spacecraft, the gaseous propellant supply is generally isolated from the rest of the system, prior to operation, to allow servicing and checkout operations and to minimize propellant leakage. The system is activated by firing the isolation squib valves, providing propellant to the thrustor valve inlet.

This is usually done prior to launch in the pre-launch or countdown sequence, or after reaching orbit. Cold gas systems are used infrequently in boosters, and then without isolation valving of the high pressure gas circuit.

The thrustor valve controls gas flow, and consequently the impulse delivered by the engine. This valve is usually spring loaded so that failure of the power supply or solenoid coil causes the valve to fail in a closed position. In normal operation, an electrical command energizes the solenoid coil, moving the valve armature and lifting the valve poppet from the seat. This unports the valve exit, permitting gas to enter the plenum chamber. A smooth flow transition is provided to the nozzle entrance. The gas undergoes a controlled expansion in the nozzle, converting the random gas molecule velocity (measured by gas temperature) to an ordered exhaust velocity. The increase in mean gas velocity from essentially zero in the plenum chamber to the exhaust velocity at the nozzle exit produces a reaction force, or thrust. Additional thrust results from the pressure differential (nozzle exit pressure to ambient pressure) acting over the nozzle exit area.

6.1.3 COLD GAS THRUSTOR CONSTRUCTION

The cold gas thrustor is the simplest of small rocket engines, having only one propellant and requiring no catalyst. Figure 6.1-3 shows the construction of a typical unit involving a closely integrated valve and engine unit. Physically it consists of a nozzle, plenum chamber, and propellant control valve. The valve is predominant in cold gas thrustors, and especially so at lower thrust levels such as shown here.

Materials and construction methods are selected to satisfy the requirements for low cost, light weight, reliable systems having high cycle life and very low leakage characteristics. Shell and closure materials can be of high strength aluminum alloys, such as 2024, since thermal variations in the chamber are small. Attaching the nozzle assembly by means of mounting screws, as shown, makes it easy to change thrust level or nozzle expansion ratio by switching to nozzles with different throat and/or exit areas. This enables a single, basic unit to satisfy a range of thrust, expansion ratio, or geometry requirements, thereby reducing unit costs and development requirements.

The flow control unit consists of a hard, tungsten-carbide poppet and a stainless steel poppet seat. This assures a good gas seal for many operating cycles during prolonged exposure to the space environment. Internal and external leakage are reduced by providing teflon and silicon gaskets and by strongly loading the poppet seating spring.

6.1.4 COLD GAS PROPELLANTS

"Cold gas" thrustors used propellants stored in gaseous form at, or near, ambient conditions. They may also use gaseous propellants stored as liquids at reduced temperatures. They are referred to as "cold gas" thrustors primarily

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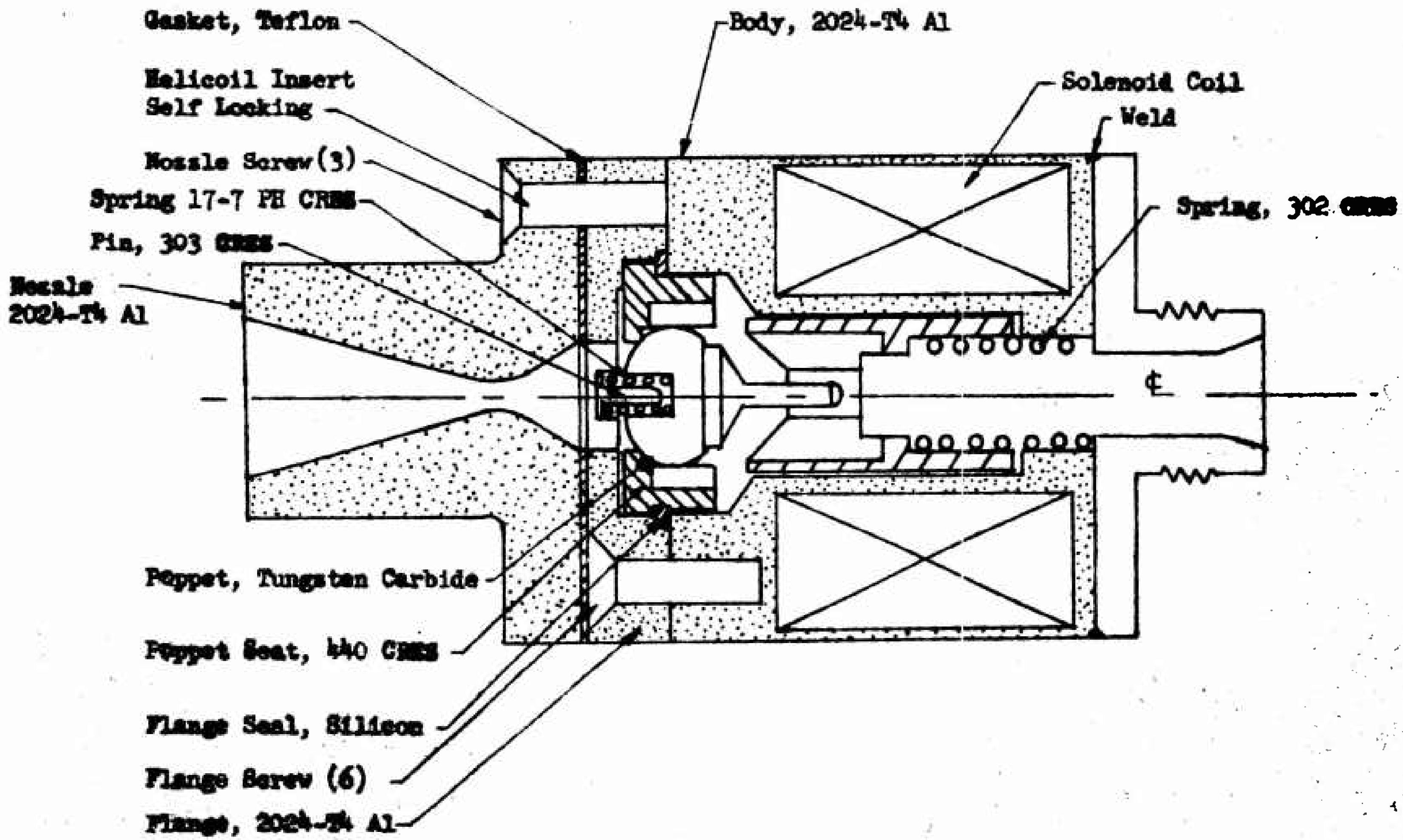


FIGURE 6.1-3 COLD GAS THRUSTOR - CONSTRUCTION DETAILS (Refs. 12 and 13)

to distinguish them from engines in which heating takes place by electrical, isotopic, or chemical means.

"Cold gas" thrustors have been operational for many years. During this period many different gases such as those in Table 6.1-1 have been suggested as candidate propellants. Primarily, they have consisted of single component gases, though some gas mixtures have also been considered. Generally, these gas mixtures occur as byproducts of other subsystems, such as for life support. Only single component gases are currently used, however, and among these, nitrogen applications predominate.

6.1.5 PERFORMANCE

Cold gas thrustor performance may be rather easily determined on an ideal basis by use of expressions 5.1-19 through 5.1-27. This involves an assumption that the specific heat ratio (γ) is constant during expansion. This is a valid assumption since pressure and temperature effects are small enough to be neglected with most propellants. Hydrogen gas is an exception in this case. These performance expressions also require assumptions for nozzle expansion ratio and chamber operating pressure and temperature. These latter two items (chamber pressure and temperature) do not remain constant when the system uses an unregulated blowdown mode. In fact, gas storage temperature is not constant during pressure regulated operation, either, due to gas expansion in the storage bottle.

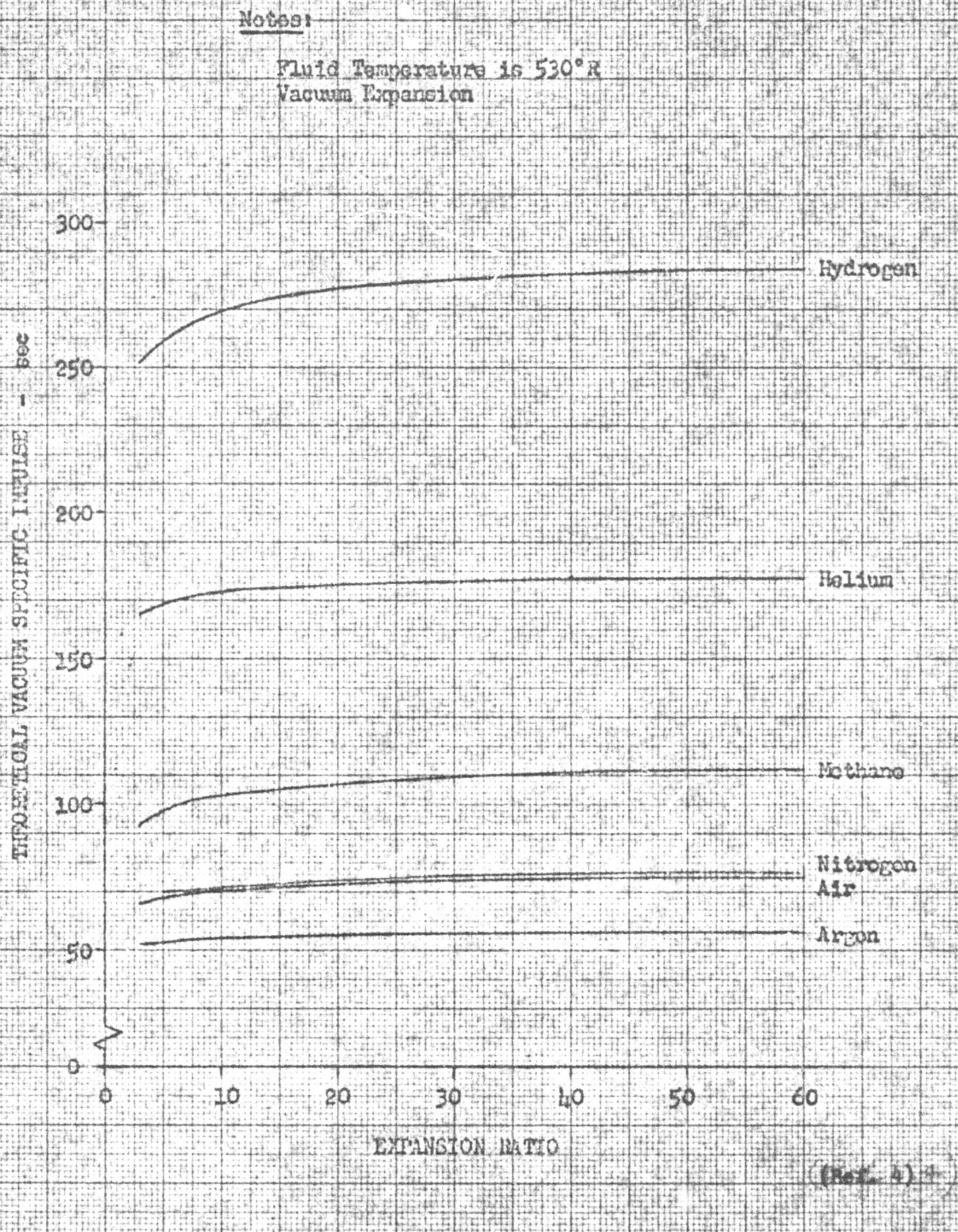
Theoretical performance of selected cold gas propellants is shown in Figure 6.1-4. However, in preliminary design exercises, steady state performance of cold gas propellants is not usually estimated from this data even if efficiency

TABLE 6.1-1

CHARACTERISTICS OF SINGLE COMPONENT COLD GAS PROPELLANTS

Name	Symbol	Molecular Weight	Normal Boiling Point (°F)	Specific Heat Ratio, (γ = Cp/Cv)	Max I_s Theo (@ 530°R)	Critical Properties Temp.	Critical Properties Press.
Acetylene	C_2H_2	36.04	-118.5 (sublimes)	to 26 @ 59°F	97	96.8°F	911 PSIA
Ammonia	NH_3	17.024	- 28	1.31 @ 59°F	112	270	1639
Argon	A	39.944	-302.55	1.67 @ 70°F	56	-188	705
Carbon Dioxide	CO_2	44.01	-109 (Sublimes)	1.304 @ 59°F	70	88	1072
Ethane	C_2H_6	30.07	-127	1.22 @ 59°F	97	90	717
Freon 11	CCl_3F	131.38	+75.4	1.135	56	386	612
Freon 12	CCl_2F_2	120.9	- 18	1.138	59	233	580
Freon 13	$CClF_3$	104.47		1.172 @ -22°F	58	839	561
Freon 14	CF_4	88.01		1.2	59	- 49.9	542
Freon 21	$CHCl_2F$	102.93	+ 48	1.12	68	353	750
Freon 22	$CHClF_2$	86.48	- 41	1.178 @ 250°F	62	205	716
Helium	He	4.003	-452.06	1.66 @ 70°F	179	-450	33.2
Hydrogen	H_2	2.016	-423	1.41 @ 70°F	295	-400	188
Methane	CH_4	16.04	-258.5	1.31 @ 59°F	116	-117	674
Nitrogen	N_2	28.016	-320.45	1.40 @ 70°F	80	-233	492

(Ref. 14)



CALC			REVISED	DATE	THEORETICAL PERFORMANCE COLD GAS FLUIDS	FIGURE 6.1-4
CHECK						
APR						
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factors are available. Rather, it is arbitrarily specified as 68 lb_f sec/lbm for nitrogen consistent with a +80°F to -40°F blowdown temperature range.

This procedure has developed from numerous designs, studies and applications conducted throughout the industry. It is also not uncommon to supplement this with a propellant loading based on safety factors between 2.0 and 3.0. In other words, industry experience with nitrogen systems in a variety of applications has produced a relatively conservative, but constant, value of 68 lb_f-sec/lb_m for use in preliminary subsystem design. Nitrogen, being easily the most popular "cold gas" propellant has also developed into somewhat of a standard in that performance of other gases may be estimated by comparison. For example, the "ideal" performance equation:

$$I_{s_{id}} = \frac{C_{F_i}}{W}\sqrt{\frac{T_c}{m}} \qquad (5.1\text{-}25)$$

can be further simplified to:

$$I_s \doteq K_G\sqrt{\frac{T_c}{m}} \qquad (6.1\text{-}1)$$

where: K_G = a constant for a particular gas at specified conditions

Hence, the specific impulse of any other gas, I_{s_x}, equals:

$$I_{s_x} \doteq I_{s_{N_2}}\left(\frac{K_x}{K_{N_2}}\right)\left[\frac{T_{c_x}\,m_{N_2}}{T_{c_{N_2}}\,m_x}\right]^{1/2} \qquad (6.1\text{-}2)$$

or,

$$I_{s_x} \doteq I_{s_{N_2}}\left(\frac{C_{F_x}}{C_{F_{N_2}}}\right)\left(\frac{W_{N_2}}{W_x}\right)\left[\frac{T_{c_x}\,m_{N_2}}{T_{c_{N_2}}\,m_x}\right] \qquad (6.1\text{-}3)$$

On this basis, helium yields a steady state specific impulse of 158 lb_f sec/lb_m at equivalent T_c values.

6.1.6 COLD GAS THRUSTOR DESIGN

Most cold gas thrustors are engine-valve arrangements combined in a single housing into which different size nozzles can be installed. Preliminary design of the cold gas thrust chamber and nozzle can be conducted in the following manner:

1) Assume engine thrust level, F
2) Determine propellant flow rate, $\dot{w}$, by:

$$\dot{w} = F/I_s \qquad (6.1\text{-}4)$$

where: I_s = 68 lb_f sec/lb_m for nitrogen

Calculate I_s for other propellants using equation 6.1-2

3) Determine C_F for the propellant from Figures 5.1-3, 5.2-5 or equations 5.1-19 through 5.1-22.
4) Assume chamber pressure equal to 40 psia for pulse operated engines or 150 psia for steady state operation. Structurally, any value less than 400-450 psia is usually possible. However, high chamber pressure needs a small nozzle throat which is sensitive to contamination and difficult to produce economically, within tolerances, below 0.01 inches. The 40 psia value is common for small attitude control engines. Steady state operation usually involves higher thrust levels permitting the higher chamber pressure suggested.
5) Calculate nozzle throat size by:

$$A_T = F/P_C\, C_F \qquad (6.1\text{-}5)$$

6) Determine chamber volume, V_C, by:

$$V_C = L^* \, (A_t) \qquad (6.1\text{-}6)$$

where: L*, chamber characteristic length is selected above a minimum of 10 inches to minimize inlet effects of gas flow into the chamber.

6.1.7 COLD GAS THRUSTOR GEOMETRY

The closely integrated valve-chamber design of most cold gas thrustors causes them to be dimensionally quite dependent on valve characteristics. These characteristics are quite dependent on the valve manufacturer's approach. The cold gas engine thrust chamber and nozzle, immediately adjacent to the valve pintle can be sized in a preliminary basis using the following procedure:

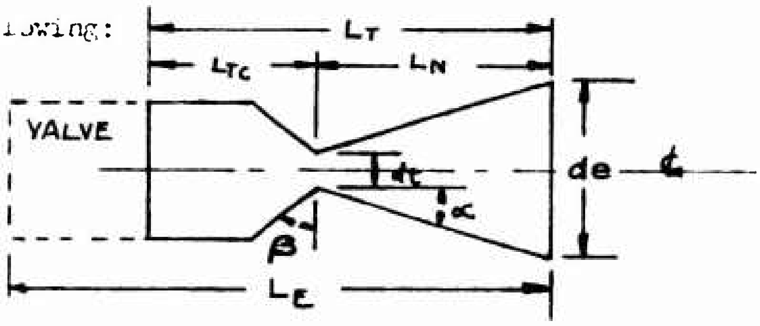
1) Assume, from 6.1.7, the following:

   Chamber pressure (P_C)

   Thrust (F)

   Expansion ratio (ϵ)

   Specific Impulse (I_s)



2) Assume the following values determined with the procedures described in 6.1.7:

   Propellant flow rate, $\dot{\omega}$

   Nozzle throat area, A_t

   Chamber volume, V_C

3) Assume the following based on common practice with cold gas thrustors:

   Nozzle divergence half angle, α, = 15° (conical)

   Nozzle convergence angle, β = 30°

4) Determine nozzle exit area, A_t, thrust diameter, d_t, and exit diameter, d_c:

$$A_E = A_t \epsilon \qquad (6.1\text{-}7)$$

$$d_t = \left[4A_t/\pi\right]^{1/2} \qquad (6.1\text{-}8)$$

$$d_e = \left[4A_t/\pi\right]^{1/2} = d_t\,(\epsilon)^{1/2} \qquad (6.1\text{-}9)$$

5) Calculate chamber length, L_{tC}, by:

$$L_{tC} = \frac{V_C + 0.398\, d_t^{\,3}}{3.14\, d_t^{\,2}} \qquad (6.1\text{-}10)$$

6) Determine nozzle length, L_N, by

$$L_N = \frac{d_t \left[(\epsilon)^{\cdot} - 1\right]}{0.530} \qquad (6.1\text{-}11)$$

7) Determine thrustor length, L_T, by

$$L_T = L_{tC} + L_N \qquad (6.1\text{-}12)$$

6.1.8 COLD GAS THRUSTOR WEIGHT

Thrustor weight is strongly affected by valve weight which in turn, depends on the valve manufacturer involved. Cold gas thrustors are usually basic units, sized for a range of thrust levels consistent with various nozzles and different propellant supply pressures. Hence, these units may be quite oversized on a weight basis at lower thrust ratings. That is, a lighter weight, more compact assembly might be developed if it were advisable from a weight and/or cost standpoint. However, cold gas thrustors are usually so small that they contribute little to total spacecraft weight. The total weight of cold gas thrust chamber, nozzle, and valve assembly may be estimated from the following expression:

$$W_{C.G.} = 0.2 + 0.15\, F \qquad (6.1\text{-}13)$$

This expression is reasonably valid within a thrust range of 0.05 to 50.0 lbs providing that an absolutely minimum weight package is not involved.

6.1.9 DEVELOPED COLD GAS THRUSTORS

Cold gas thrustors have been developed for use over a thrust range of 0.002 to 14.0 lbs. Primarily they have used nitrogen or helium though Freon, krypton, and argon have also seen limited use. A list of these cold gas thrustors is shown in Table 6.1-2 . This list implies a much greater number of thrustor designs than actually exist. Cold gas thrustors are quite easily applied throughout a wide thrust level by varying inlet pressure and nozzle fittings. Thus, a few basic thrustor/valve designs can easily cover a broad thrust range, providing structural design and response are adequate. Weight penalties associated with a large chamber pressure range are slight in the cold gas thrustor sizes commonly employed. Response is very good since the cold gas valve and thrust chamber are very closely coupled. Thus, Table 6.1-2 includes both basic thrustor designs and particular applications of these thrustors. in which operating characteristics are somewhat different. This is an advantageous feature of cold gas thrustors in that development expenses can be written off against numerous programs. The units are relatively inexpensive on a recurring cost basis, and little development time is required for most programs.

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TABLE 6.1-2 DEVELOPED COLD GAS THRUSTORS

THRUST (LBS)	PROGRAM	VENDOR	GAS	I_s (SEC)	P_{IN} (PSIA)	P_C (PSIA)	DIMENSIONS WITH VALVE (INCHES) WIDTH	LENGTH	WEIGHT (LBS) W/O VALVE	WITH VALVE	MIN. IMP BIT $I_{T_{MIN}}$ (LB-SEC)	POWER (WATTS)
.00207	OAO		N_2		(4 1) 7 0	5 0				0.32		2.0
.005	RANGER	STERER	N_2	76		15						
.02	MARINER ROLL	STERER	N_2	60	15							
.028	LUNAR ORBITER-ROLL	STERER	N_2	70	19					0.22		4.5
.03	SURVEYOR	STERER	N_2									
.04	MARINER PITCH, YAW	STERER	N_2	60	15							
.05	ABLESTAR ROLL		N_2	74								
.05	LUNAR ORBITER PITCH, YAW	STERER	N_2	70	19					.23		4.5
.05	OGO A, B, C	TRW	ARGON	56	50							6.5
.05	OGO-D	TRW	KRYPTON	36	50							6.5
0.05	OSO		N_2	66		30						
0.10	OSO		N_2	66		60						
0.104	OAO		N_2		94-70		1.90	5.0		.42		28
0.20	VASP	TRW	N_2	72								8
0.20	VELA	TRW	N_2	72	50							8
0.20	PIONEER	TRW	N_2	72	50						4×10^{-4}	8
0.5-0.2	NIMBUS D		FREON	45	55							
1.0-0.1	(WITH "C" NOZZLE)	JAMES, POND & CLARK	N_2, He	(N_2) 65	5-50		3.12	1.94	.13			5
1.0	PAYLOAD AEROBEE		He									
2.0	SYNCOM		N_2									
2.2	BURNER II	KIDDET JP&C	N_2	65		150-350	1.56	3.69				
2.5		STERER	N_2				1.97	3.08		.563		
2.5-0.25	(WITH "B" NOZZLE	JAMES POND & CLARK	N_2, He	(N_2) 65	5-50		3.12	1.94	.13			5
4.50	ABLESTAR		N_2	74								
4.0-0.4	(WITH "A" NOZZLE)	JAMES, POND & CLARK	N_2, He	(N_2) 65	5-50		3.12	1.94	.13			5
5.0		STERER					2.26	4.13		.563		
5.0	AEROBEE		He									
5.0-1.0	POLARIS STV	TRW		70	300							36
5.0-1.0	ABLESTAR			70	300							36
7.0	DELTA-ROLL		He	160	345							
10.0		STERER	N_2		120		2.78	5.06		1.67		12
10.-0.5	AGENA	STERER	N_2	68	104.5	100	3.0	5.75		2.0		
10.0		STERER	N_2				3.4	5.9				14.4
10.0	AEROBEE		He									
12		STERER	N_2									
13	DELTA		He	160	345							

(Refs. 4 and 14)

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6.2 HEATED GAS THRUSTERS

6.2.1 GENERAL

The disadvantages of cold gas thrusters related to their relatively low performance can be relieved by heating the gas. According to equation 5.1-25, specific impulse increases proportionally with the square root of gas temperature in the chamber, all other factors being equal. The currently popular cold gas systems are expected to be replaced in time with higher performance equipment which may include propellant heating provisions.

This section describes thrustors which are supplied with, and heat, a gas that is initially at ambient temperature. Thrustors supplied with a hot gas (e.g. from a main combustion chamber or gas generator) are omitted although portions of the performance data are applicable. Propellant gas heating is accomplished by using electrical resistance heaters (resistojets) or nuclear radioisotope decay (radioisojets).

6.2.2 PRINCIPLE OF OPERATION

A typical heated gas thrustor consists of an inlet tube, a propellant valve, a heater, a heat exchanger, and a converging-diverging nozzle. The configuration is illustrated schematically on Figure 6.2-1.

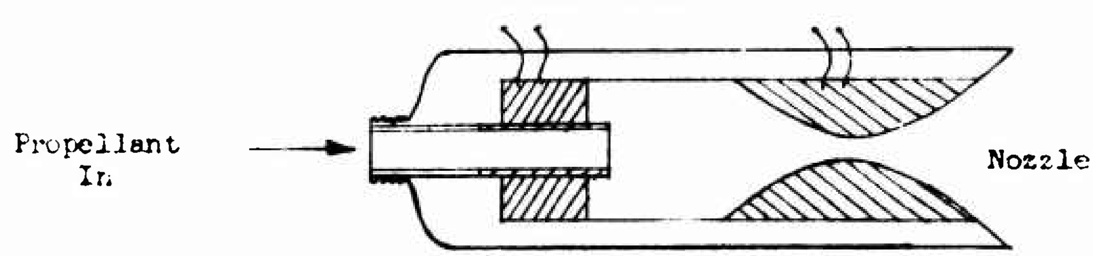


FIGURE 6.2-1 TYPICAL HEATED GAS THRUSTOR

This configuration has its own heater and heat exchanger unit, though several engines can be designed to use the same heater and heat exchanger.

Heated gas thrustors operate similarly to cold gas thrustors (6.1.2) except that the gas is heated prior to being exhausted through the nozzle. The heat exchanger is usually configured to raise gas temperature to some value between 1500°R and 4000°R.

A heated gas propulsion system can be designed identically to cold gas systems except for thruster differences. Thrustor design includes a high temperature heater and, possibly, thermal insulation. Propellant storage, regulation, distribution and control systems are similar to cold gas systems. Thus, in the event of heater element failure the system can be used as a cold gas system.

Heated gas thrustors are designed as either "thermal storage" or "fast-heat up" devices. Thermal storage thrustors are primarily suited to pulse-mode applications, whereas the fast-heat-up thrustor can be used in either pulse-mode or steady-state applications. Power, in the thermal storage thrustor, is supplied continuously by a heater element (nuclear or electric) and propellant flow is pulsed. The applied power is equal to the thrustor heat losses at the design operating temperature. The heat capacity of the thermal storage unit must be large enough for the heater element temperature to remain essentially constant during short propellant pulses. In the fast-heat-up thrustor, both propellant flow and power are pulsed. In contrast to the thermal storage thrustor, the heat capacity of the fast-heat-up device is minimized. As a result, the power input is equal to the thrustor heat losses plus the heat required to increase the temperature of the propellant.

The advantages and disadvantages of each of these concepts are summarized in Table 6.2-1

TABLE 6.2-1

COMPARISON OF HEATED GAS THRUSTOR CONCEPTS

	Thermal Storage (High heat capacity)	Fast Heat Up (Low heat capacity)
Advantages	No thermal cycling Minimum response time Constant power input Simple power supply	Low average power consumption Low thrustor weight constant Isp for all duty cycles
Disadvantages	High average power consumption High thrustor weight Isp decreases as duty cycle increases	Frequent thermal cycling Delay between command signal and impulse bit More complex power supply

Table 6.2-2 shows a list of applications of resistojet type heated gas thrustors.

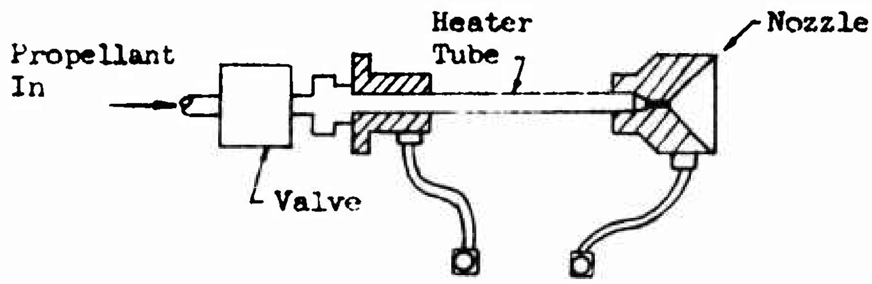
TABLE 6.2-2

RESISTOJET APPLICATIONS

Application	Thrust (lb)	Vendor
Vela II	0.042	TRW Systems
Advanced Vela	0.020	TRW Systems
ATS - 1	500×10^{-6}	AVCO
ATS - C	100×10^{-6}, 10×10^{-6}	AVCO
LES - 7		
DODGE-M	3×10^{-6}, 6×10^{-6} 200×10^{-6}	AVCO
R&D	0.020	G.E.

6.2.3 CONSTRUCTION OF HEATED GAS THRUSTORS

A simple heated gas thrustor (Figure 6.2-2) is basically a cold gas engine with a heater and heat exchanger unit. When several thrustors use a single heater unit, each thrustor has its own valve.



(Ref. 15)

LOW HEAT CAPACITY, FAST HEAT-UP HEATED GAS THRUSTOR
FIGURE 6.2-2

Propellant valves normally utilize cold-gas technology, such as elastomeric seals and valve seats, and therefore are thermally isolated from the high-temperature heat exchanger. Thermal isolation is accomplished by placing a propellant supply tube having a thermal resistance between the valve and the heat exchanger. When thrust is required, the valve is opened and propellant flows through the heat exchanger and the nozzle. The heat exchanger raises the propellant gas temperature from ambient temperature (typically about 500°R) to a temperature in the range of 1500°R to 4000°R.

The heat exchanger can be as simple as an electrically heated tube through which propellant gas flows, or it can use a relatively complex multiple-pass configuration.

Figure 6.2-2 shows a low heat capacity, fast-heat-up type of thrustor. The resistive heating element is the stainless steel propellant feed tube. This configuration is simple, small, and light, but it has relatively lower performance because the heat transfer process is not particularly efficient and because the heat losses during operation are quite high.

Figure 6.2-3 is a low heat capacity, steady-state type of thrustor, designed for high performance (above 800 seconds specific impulse) with hydrogen gas propellant. The use of regenerative cooling reduces the heat losses.

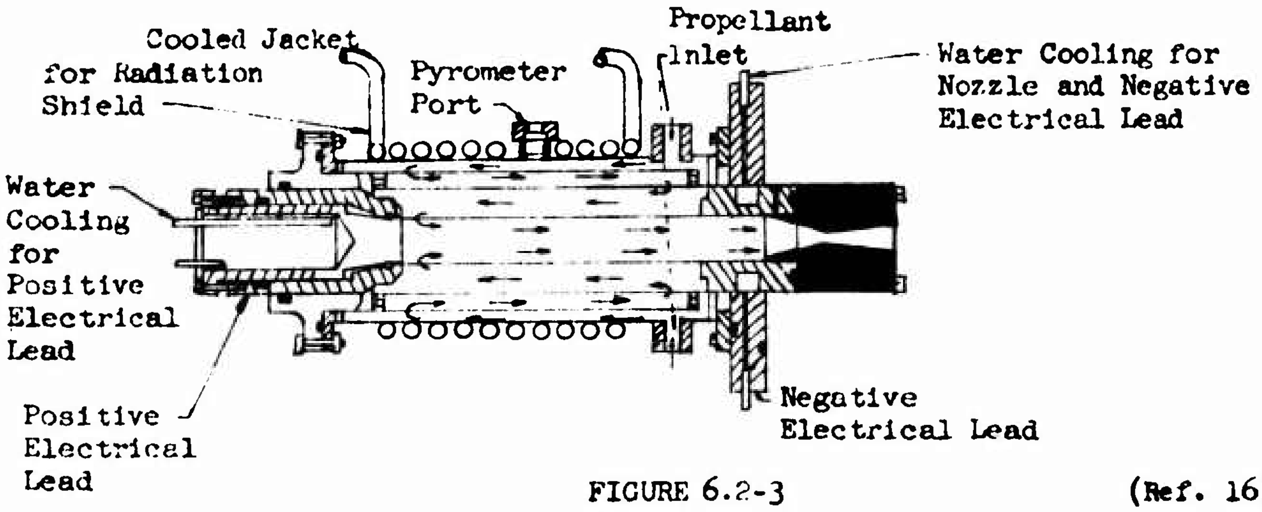
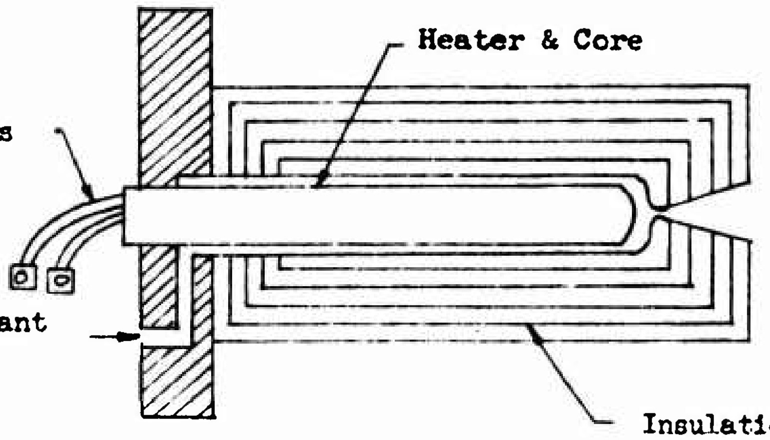


FIGURE 6.2-3 (Ref. 16)

WATER-COOLED TUBULER HEAT EXCHANGER THRUSTOR

Figure 6.2-4 shows a high heat capacity thrustor. The heater elements are mounted on a central core, which stores thermal energy when there is no propellant flow. When propellant is flowing, the central core is cooled as the propellant is heated. A radioisotope power core can be substituted for the resistance-heated core, if desired.



(Ref. 17)

FIGURE 6.2-4 HIGH HEAT CAPACITY HEATED GAS THRUSTOR

Insulation is used on thermal storage thrustors to reduce power requirements. It is not usually used with low heat capacity (fast heat-up) thrustors because (1) small thrustor heat loss is low, (2) low thermal capacity is required, and (3) larger thrustors (i.e., hydrogen) can be regeneratively cooled.

Power requirements are significantly influenced by the impulse per pulse, and the number of nozzles per heater/core element. The maximum impulse per impulse bit and the duty cycle determine the required energy storage capacity and therefore determine the weight and volume of the core structure to be insulated. Each nozzle represents a "radiation window" in the insulation, so power required is affected by the number of nozzles. In addition, each nozzle requires a separate propellant feed line, so that the conduction component of the heat loss is dependent on the number of thrust nozzles used.

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6.2.4 **PROPELLANTS FOR HEATED GAS THRUSTORS**

Propellants for heated gas thrustors include the single component gases nitrogen (N_2), helium (He), hydrogen (H_2), and ammonia (NH_3), as well as gas mixtures obtained from subliming solids such as ammonium sulfide $(NH_4)_2S$, ammonium carbonate $NH_4CO_2NH_4$ and ammonium hydrosulfide, NH_4HS. The VELA satellites used heated nitrogen (N_2) with 2% argon by volume.

Current emphasis is on the use of ammonia (NH_3) for smaller systems because of its high storage density (low storage volume) in liquid form. The advanced technology satellite (ATS-1) uses ammonia with a fast heat-up thrustor. Advanced VELA is currently planned for ammonia, in conjunction with a thermal storage type thrustor.

Previous emphasis on using hydrogen in high power systems ($\geq$1KW) was based on the expectation that its potentially high specific impulse ($\approx$ 850 lb_fsec) would offset the tankage penalties associated with its low density. This interest has recently abated in the absence of immediate missions and practical power supplies.

Characteristics of propellants seriously considered for use with heated gas thrustors are described in Table 6.1-1, since they are generally the same as those considered for cold gas systems.

6.2.5 **HEATED GAS THRUSTOR PERFORMANCE**

Estimates of steady state performance of heated gas thrustors can be made for preliminary design studies using the propellant performance data and procedures covered in 6.1.5 for cold gas thrustors. Basically, a broad range of performance is possible if enough power is available for raising gas temperature. Performance for nitrogen thrustors can thus be estimated by using:

$$I_{sp_{N_2}} = 3.32\sqrt{T_c} \tag{6.2-1}$$

where: T_c = gas temperature in thrust chamber (°R)

The performance of other gases can be estimated by modifying this expression to accommodate the difference in molecular weight, providing other processes such as dissociation are not involved. For other gases, equation 6.2-1 becomes:

$$I_{sp_x} \cong 17.58\left[\frac{T_{c_x}}{\overline{m}_x}\right]^{1/2} \tag{6.2-2}$$

where: $\overline{m}_x$ = molecular weight

The principal differences from cold-gas thrustor performance estimation concern dissociation effects with certain propellants such as ammonia. The effects of dissociation on performance evaluation are two-fold:

a. Propellant chemical composition may change.

b. The energy required to increase propellant temperature is significantly affected by the degree of propellant dissociation.

Figure 6.2-5 shows the effect of temperature and pressure on ammonia as it is progressively dissociated to nitrogen and hydrogen. The effect of dissociation on energy can also be inferred from this figure. It is thus shown, that after dissociation commences, large amounts of energy are absorbed by the dissociation process rather than by increases in gas temperature. For example, Figure 6.2-6 illustrates the effect of dissociation on specific impulse for two different thrustors designs having different surface-to-volume ratios

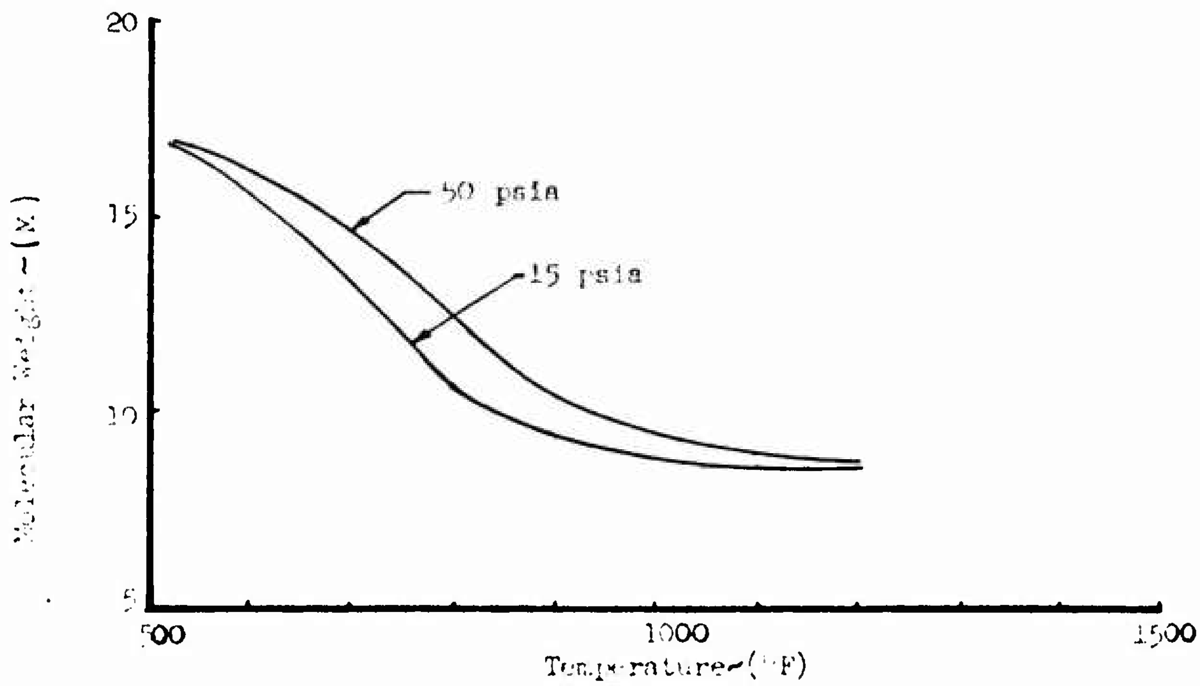

FIGURE 6.2-5 MOLECULAR WEIGHT OF THE EQUILIBRIUM $2NH_3 \rightleftharpoons N_2 + 3H_2$ SYSTEM

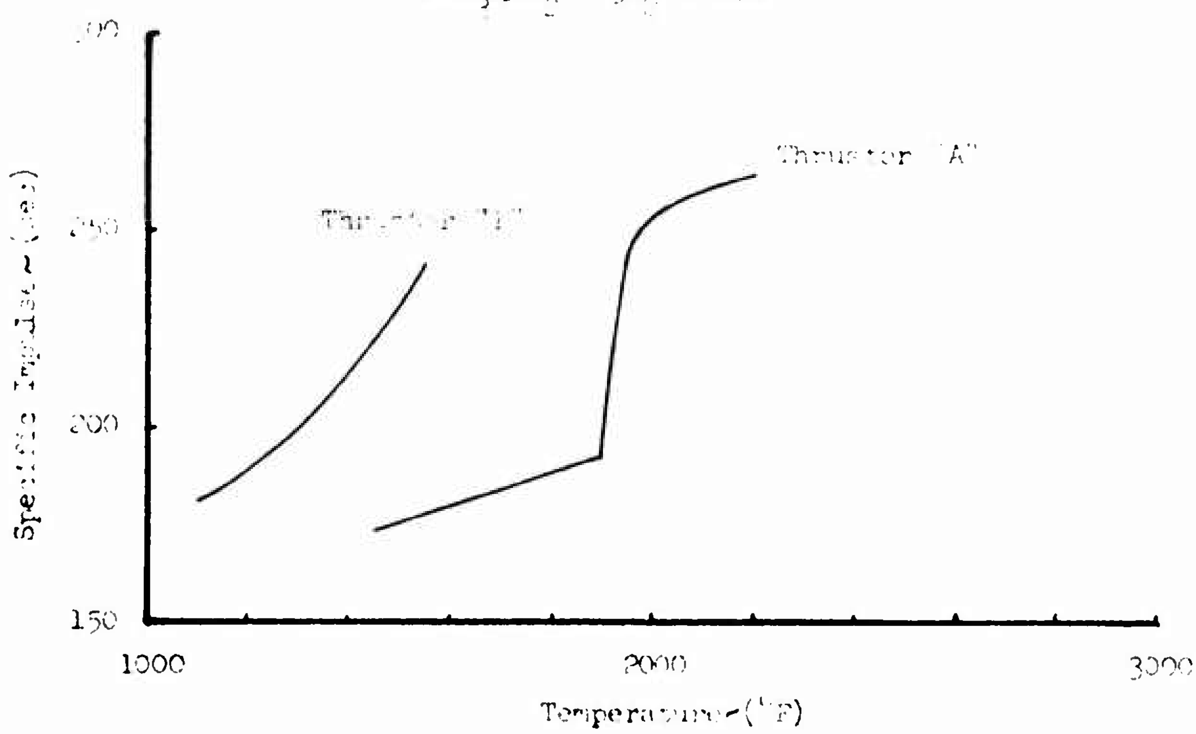

FIGURE 6.2-6 DISSOCIATION EFFECT ON PERFORMANCE OF HEATED AMMONIA THRUSTORS

(Refs. 15 and 18)

in the flow passages. The larger relative surface area means that more of the ammonia flow is exposed to the catalytic action of the propellant flow tubes, and hence more complete dissociation is obtained.

In the low heat capacity thrustors, the principal efficiency factors used to measure performance are the frozen flow efficiency, η_F, and the overall engine efficiency, η_e. The frozen flow efficiency is the ratio of available thermal power to the total power put into the propellant. The term "frozen flow" is used since propellant composition is essentially frozen at stagnation conditions. This occurs at low thrust levels since short nozzles produce propellant residence times measured in microseconds (Reference 15). Frozen flow efficiency is evaluated by:

$$\eta_F = \frac{\dot{m}\,(H_o - H_F)}{\dot{m}\,H_o} \qquad (6.2\text{-}3)$$

where:

H_o = stagnation enthalpy of propellant

H_F = enthalpy of dissociation or ionization

$= 41526\,\alpha\,E_i/\mathcal{M}$ (Btu/lb)

α = degree of dissociation ($0 \leq \alpha \leq 1$)

E_i = dissociation potential (volts) of the propellant molecule

This definition of frozen flow efficiency assumes that propellant enthalpy at the nozzle exit, H_e, is much less than the stagnation enthalpy, H_o, and so can be neglected. Since dissociation lowers frozen flow efficiency, thrustor operating conditions are selected to minimize dissociation when good performance is desired.

Overall engine efficiency, η_e, is the ratio of the jet power to input power or:

$$\eta_e = \eta_F \eta_P \eta_n \qquad (6.2\text{-}4)$$

where: η_F = frozen flow efficiency

η_P = heater efficiency

η_n = nozzle efficiency = c_v

Heater efficiency is difficult to define analytically since it depends on the efficiency of insulation (if any), the method of cooling the thrustor (radiation or regenerative) and the efficiency of the heat transfer process between propellant and heater. Nozzle efficiency, η_n, is simply the nozzle velocity coefficient, c_v, defined in 5.2-1.

Overall engine efficiency is calculated from the basic definition,

$$\eta_e = \frac{P_j}{P_i} = \frac{\text{Jet Power}}{\text{Input Power}} \qquad (6.2\text{-}5)$$

Jet power can be calculated from measurements of thrust and Isp by

$$P_j = .0218 \text{ Isp } F \qquad (6.2\text{-}6)$$

Input power is electrical power plus effective power of the incoming propellant stream,

$$P_i = P_{electrical} + \dot{m} H_i \qquad (6.2\text{-}7)$$

The slope of the curve on Figure 6.2-7 is an indication of the engine efficiency (the lower the slope, the higher the efficiency). The curve also shows that more power per unit of thrust is required when the specific impulse is increased.

Power required as a function of operating temperature is shown on Figure 6.2-8 for a thermal storage type of electrical thrustor.

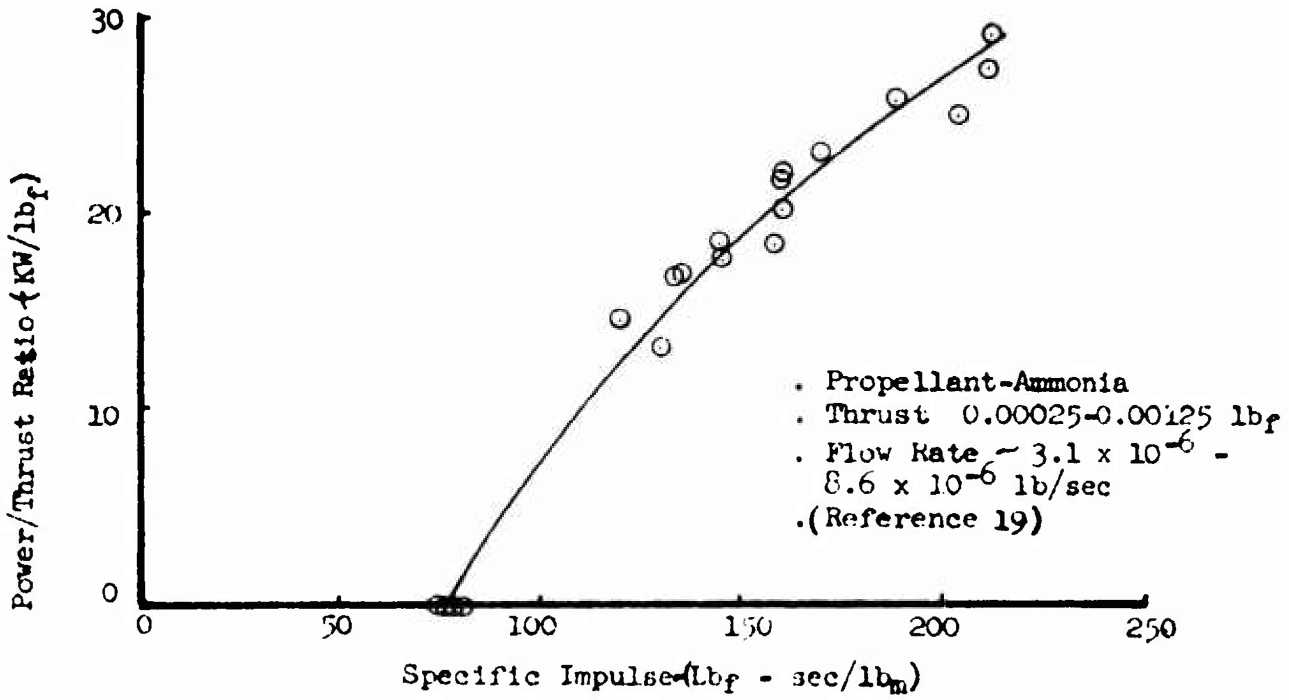

FIGURE 6.2-7 RESISTOJET POWER REQUIREMENTS, FAST HEAT-UP THRUSTOR

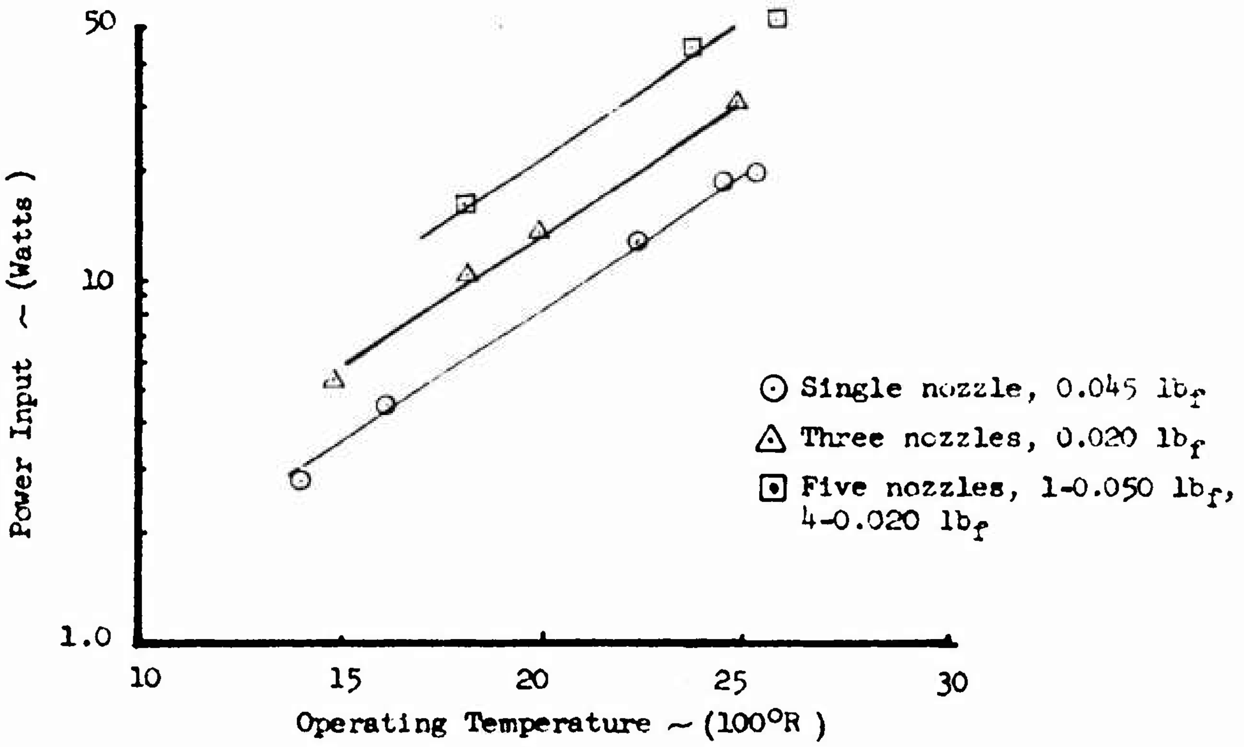

FIGURE 6.2-8 RESISTOJET POWER REQUIREMENTS, THERMAL STORAGE THRUSTOR

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For high heat capacity thrustors, the preceding efficiency factors, η_F, η_p, η_n do not adequately characterize the thrustor performance. For example, maximizing the frozen flow efficiency means minimizing the degree of propellant dissociation. This occurs since additional electrical energy is required for propellant dissociation which is not then available during the expansion process. However, for high heat capacity thrustors, propellant dissociation does not require an increase in electrical power, so that the lower molecular weight of the dissociation propellant results in a performance increase with no increase in power requirements. As a result, a design goal for high heat capacity thrustors involves maximizing propellant dissociation at operating temperature and pressure.

6.2.6 DESIGN OF HEATED GAS THRUSTORS

Heated gas thrustors are basically cold gas thrustors which utilize gases at temperatures significantly above ambient conditions. The heater unit may be contained within the thrustor, or remotely located. Consequently, the chamber design is similar to that required for cold gas thrustors except as it relates to heater, heat exchanger, and electrical provisions. Consequently, the following preliminary design procedure, similar to that suggested for cold gas thrustors, is presented:

1) Assume engine thrust level, F
2) Calculate propellant specific impulse, I_s, from:

$$Isp = 17.58\left[\frac{T_{c_x}}{\overline{m}_x}\right]^{\frac{1}{2}} \qquad (6.2\text{-}2)$$

where: $\overline{m}_x$ = molecular weight

3) Determine propellant flow rate by:

$$\dot{w} = F/Isp \qquad (5.1\text{-}23)$$

4) Determine C_F from Figures 5.1-3, 5.2-5, or equations 5.1-19 through 5.1-22.

5) Assume chamber pressure. Current practice involves chamber pressures between 1 to 30 psia, and usually between 7 to 20 psia. The real limiting condition is again that of manufacturing very small nozzle throats to close tolerances.

6) Calculate nozzle throat area by:

$$A_T = \frac{F}{P_c\ C_F\ \eta} \qquad (6.1\text{-}5)$$

7) Determine free chamber volume, V_c, by:

$$V_c = L^* (A_t) \qquad (6.1\text{-}6)$$

where: L^* - characteristic length is equal to a typical value somewhat greater than 10.

6.2.7 GEOMETRY

Figures 6.2-2 through 6.2-4 demonstrate that the physical size of heated gas thrustor units considerably exceeds that of the basic chamber nozzle and valve. This additional space is occupied by the heater and heat exchanger unit which comprise most of the thrustor. Since few actual thrustor designs exist, insufficient dimensional information is currently available to develop general expressions for use in preliminary design.

6.2.8 HEATED GAS THRUSTOR WEIGHT

Heated gas thrustors are being considered in spacecraft preliminary design exercises with increasing frequency. However, thrustor information suitable for preliminary design exercises has yet to be made available. Hence, this information must be inferred from the few designs extent. The weight of heated gas thrustors is strongly affected by thermal parameters, in addition to the usual engine design parameters of chamber pressure, specific impulse, thrust, and nozzle expansion ratio. Engine weight is particularly susceptible to duty cycle. The following expression is included for use in estimating the weight of heated gas thrustors. It has fair correlation to the few designs currently known. However, it is based on considerably fewer designs than was used to develop similar expressions for monopropellant and bipropellant engines. Nevertheless, it does appear to offer a reasonable solution to preliminary weight estimates for heated gas thrustors. Since these devices operate at very low thrust levels, and are thus quite small, the tolerances involved are probably not significant on a spacecraft weight basis. This expression is:

$$W_E = K_H \left[F I_s P \right]^{\frac{1}{2}} \left[0.33 + (P)^{1/3} \right] \qquad (6.2\text{-}8)$$

where:

P = Engine total input power (watts)

K_H = A constant equal to 0.50 for all thermal storage engines, and for fast-heat-up engines above 0.0025 lbs thrust. Fast-heat-up engines below this value will have a K_H value defined by:

$$K_H = \frac{0.05}{\left[F \right]^{\frac{1}{2}}}$$

6.2.9 DEVELOPED HEATED GAS THRUSTORS

Heated gas thrustors have been developed from 10^{-6} to 0.4 lb thrust using nitrogen or hydrogen propellants heated electrically or by isotope radiation. Flight applications have so far been limited to an upper thrust level of approximately 0.04 pounds. Table 6.2-3 lists known details of these engines.

TABLE 6.2-3 DEVELOPED HEATED GAS THRUSTORS

THRUST PER NOZZLE	PROGRAM CONTRACT OR AGENCY	VENDOR	MODEL	NUMBER NOZZLES/ THRUSTOR	HEATUP TIM (MIN)	PROPELLANT	CHAMBER TEMP T_c (°F)	I_s (SEC)	P_c (PSIA)	P_{in} (PSIA)	AE/AT	DIMENSIONS (INCHES) WIDTH	LENGTH	WEIGHT (LBS) W/O VALVE	WITH VALVE	ELECTRICAL POWER (WATTS) VALVE	HEATER	MINIMUM IMPULSE BIT I_{Min} (LB SEC)	LIFE (SEC)
(10)^{6}		AVCO			FHU	NH_3													
3(10)^{6}	DODGE M	GE			FHU	NH_3													
6(10)^{6}	DODGE M	GE			FHU	H_2		650											
(10)$^{-4}$		AVCO			FHU	NH_3													
2(10)^{4}	DODGE M	GE				NH_3													
5(10)$^{-4}$	ATS I	AVCO		1	FHU	NH_3													
6(10)^{4} (10)^{6}		AVCO			FHU	NH_3													
5(10)$^{-3}$	ATS B	TRW			FHU	NH_3		150									10		
0.1, 0.005	USAF	TRW	DART	4		NH_3	1500-2000	240 280	15		50	5.3++	6.0++	(CLASSIFIED)		5	155 ISOTOPE (PU238)	10^{-4}	
0.1, 0.005	AEC	TRW	HRT	MULTI		H_2	1500 2000	485-585			50			4.5			(PU238)	10^{-4}	
0.015	TEST	GE		5		NH_3													
0.015	TEST	GE				H_2	4500	260											
0.02	IR&D	TRW				NH_3													
0.02	IR&D	TRW	ACSKS	4	30	NH_3	1550 1750	240-260	15		50	5.0	9.7	0.40	0.50		14	4(10)^{4}	
0.02	ADV VELA	TRW	ADV. VELA	3	2	N_2+ O2A	1250-1425	132	30	50	67++	1.4++	2.9	0.50			17 30*	4(10)^{4}	
0.02		GE			TS	NH_3													
0.02	NASA-GODDARD	GE	TSK-2000-1P	1	TS	NH_3	1840 2100	228-266	22.5			2.0	5.0		1.75		30 37		
0.02	NASA GODDARD	GE				NH_3	1180-1450	150-230									60 ISOTOPE (PM147)		
0.042	VELA III	TRW	VELA 3	1	5	N_2+ O2A	1000-1200	123*	15		100	2.3++	5.1++	0.65			92**		SVC 4800
0.045	AFFDL, WPAFB	GE	XR204-4	1		NH_3	2000	230	30					0.96					
.05-.01		GE			TS	H_2													
.05 .01		GE			TS	NH_3													
.05-.015		GE			TS	NH_3													
.05 .0[illegible]	NASA GODDARD	GE		4(.021), 1(.05)	TS	NH_3	1378 2085												
.07		TRW				N_2												4(10)$^{-4}$	
0.10	NASA	MARQT				NH_3	4080	360								740			
0.10		MARQT.				H_2	4080	740								240			
0.12	NAS3 4101	AVCO				H_2	4400	838	34.7			3.4	6.2			3000			
0.25	POODLE	TRW		1		H_2	3600	710*				7.4	17.0	45		5000 ISOTOPE (PO 210)			
0.30		AVCO	6-A					150	14							10			
0.36		AVCO	7-A					150	14							10			

(Refs. 2, 14, 20 and 21)

6.3 MONOPROPELLANT THRUSTORS

6.3.1 GENERAL

Liquid monopropellant rocket engines produce thrust by catalytically decomposing certain chemically active propellants called monopropellants in a reactor and directing the exhaust through a plenum-nozzle arrangement. Monopropellant engines are used in Syncom, Scout, ATS, Apollo, Centaur, Titan III, Burner II and many other programs.

Early monopropellant engines used either ethylene oxide or low percentage (60 - 80%) hydrogen peroxide as propellants. Since then, 90% hydrogen peroxide has come into common use, because it has higher performance and greater density. Future generations of monopropellant engines will probably employ either 98% hydrogen peroxide (H_2O_2) or anhydrous hydrazine (N_2H_4). The 98% hydrogen peroxide thrustors are being developed as growth versions of hardware currently used with 90% peroxide. Overall gains from these modifications can be as much as 10% in performance and 1% in propellant density which is enough to justify the necessary development. Hydrazine (N_2H_4) monopropellant engines are currently becoming more popular. In their first real space applications, Ranger and Mariner, the hydrazine engines required an "oxidizer slug" to initiate the decomposition process. Subsequently, "spontaneous" catalysts have been developed which are able to initiate decomposition at ambient conditions, and sustain it. Predominant among these is the Shell 405 (Shell Development Company) spontaneous catalyst which has provided the impetus for further use of hydrazine systems. Hydrazine engines have a performance advantage over peroxide engine, with typical steady state I_s values of 235 seconds for hydrazine as opposed to

160 seconds for 90% peroxide and possibly 180 seconds for 98% peroxide. However, peroxide is about 40% more dense than hydrazine so volume requirements of 90% peroxide systems and hydrazine systems are essentially comparable. The 98% peroxide systems have about a 10% volume advantage over hydrazine systems.

Peroxide systems are used where cost and availability is important and where more performance is wanted than can be obtained with cold gas systems. Peroxide systems are somewhat storage limited, however, in that peroxide is so active that storage for long periods should be either in high pressure containers or in vessels constructed of materials which are particularly nonreactive with peroxide. Either option raises flight system weight, tending to offset the performance advantage over cold gas and the cost advantage over hydrazine. Fewer compatibility problems exist with 98% peroxide. However, it decomposes at higher temperatures than does 90% peroxide, exceeding the structural limits of the standard catalyst. Hence, a high temperature spontaneous catalyst must be developed for use in 98% peroxide engines.

Both spontaneous and nonspontaneous catalysts are in use with hydrazine engines, so hydrazine engines are available for either pulse mode or steady-stage operation. Hydrazine has superior space storage characteristics, and is thus generally considered for missions requiring high performance, several operating cycles and extended space storage capability.

Growth versions of hydrazine systems are based primarily on mixing hydrazine (N_2H_4) with water to depress its freezing point and with hydrazine nitrate ($N_2H_5NO_3$) to increase performance. The latter mixture substantially increases decomposition temperature to the extent that it exceeds the structural capability of 405 spontaneous catalyst. Current research is directed toward developing a spontaneous catalyst-binder combination suitable for use with hydrazine-hydrazine nitrate mixtures. Materials compatibility is also being researched for this propellant though it is not emphasized to the extent that the catalyst research is.

Proposals are occasionally made to use other monopropellants having significantly greater performance and better density characteristics. Generally, these monopropellants have considerable development work remaining. It is doubtful at this time whether the advanced monopropellants will be used to any extent before they are replaced by high energy bipropellants, resistojets and electrical thrustors. Hence, it may be that the very high energy monopropellant systems will attain only limited use.

6.3.2 OPERATING PRINCIPLE - MONOPROPELLANT ENGINES

Monopropellant engines resemble other rocket engines except as their operation relates to the propellant decomposition process. Figure 6.3-1 is a schematic of a typical pressure regulated monopropellant propulsion system. In this system the propellant is supplied at constant pressure to the inlet of the engine valve. Figure 6.3-2 shows a typical monopropellant thrustor, consisting of a valve assembly, injector, plenum chamber, catalyst bed and nozzle assembly. The engine valve controls monopropellant flow to the injector and into the catalyst bed where it undergoes a decomposition reaction. The products of decomposition, dissociation and possibly any additional recombination which take place in the plenum chamber then exit through the nozzle, producing thrust.

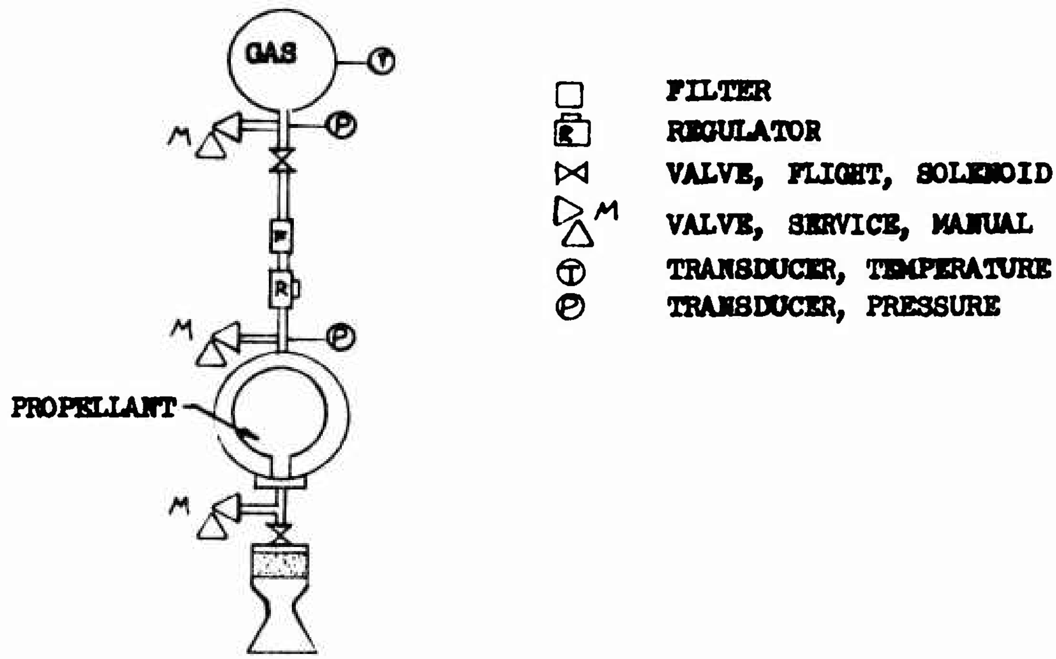

FIGURE 6.3-1 TYPICAL MONOPROPELLANT PROPULSION SYSTEM

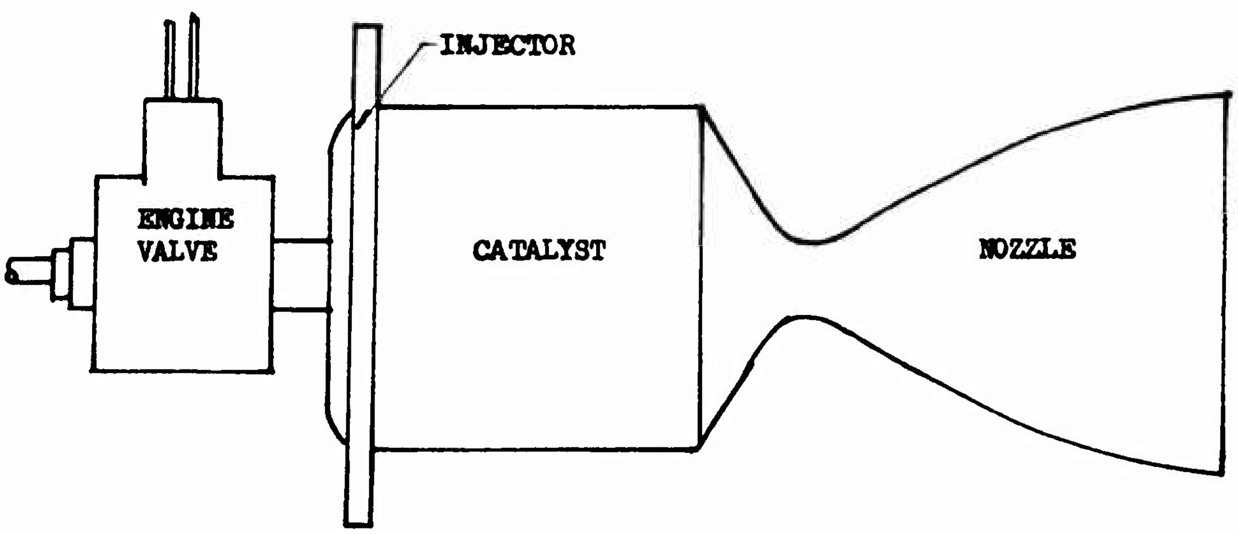

FIGURE 6.3-2 TYPICAL MONOPROPELLANT THRUSTOR

6.3.2.1 HYDROGEN PEROXIDE ENGINE OPERATION

In monopropellant rocket engines, hydrogen peroxide is decomposed by passing it through a bed of stacked, silver alloy or silver-plated nickel screens in the thrust chamber. The decomposition process yields superheated water vapor and oxygen (O_2) gas. Decomposition gas temperature varies as a function of peroxide inlet temperature and thrust chamber pressure as shown in Figure 6.3-3.

6.3.2.2 HYDRAZINE ENGINE OPERATION

Hydrazine, and hydrazine mixtures with water or hydrazinium nitrate, are decomposed by passing them through a bed of granules or small cylinders of a catalytic agent. Commonly the catalyst bed is located in the thrust chamber. Designs have been proposed in which the hydrazine is decomposed in a remotely located gas generator, the exhaust stored in a plenum chamber and thence directed to the thrustor, as in Figure 6.3-4. The hydrazine decomposition process follows these consecutive reactions:

$$3N_2H_4 \xrightarrow{cat} 4\ NH_3 + N_2 + 144{,}300\ \text{Btu}$$

$$4\ NH_3 \xrightarrow{cat} 2\ N_2 + 6H_2 - 79{,}200\ \text{Btu}$$

(where N_2H_4 and NH_3 are expressed in lb-mol)

Initially, hydrazine decomposition is exothermic, producing ammonia, nitrogen and heat. Ammonia dissociation, an endothermic process, follows, producing hydrogen and additional nitrogen and absorbing heat. Assuming that the initial reaction proceeds to completion, these reactions can be combined into a single equation related to the fraction of ammonia dissociation, X, as follows:

$$3N_2H_4 \xrightarrow{cat} 4\ (1-X)\ NH_3 + 6XH_2 + (2X + 1)\ N_2 + \left[144{,}300-X(79{,}200)\right]\ \text{Btu} \quad (6.3\text{-}1)$$

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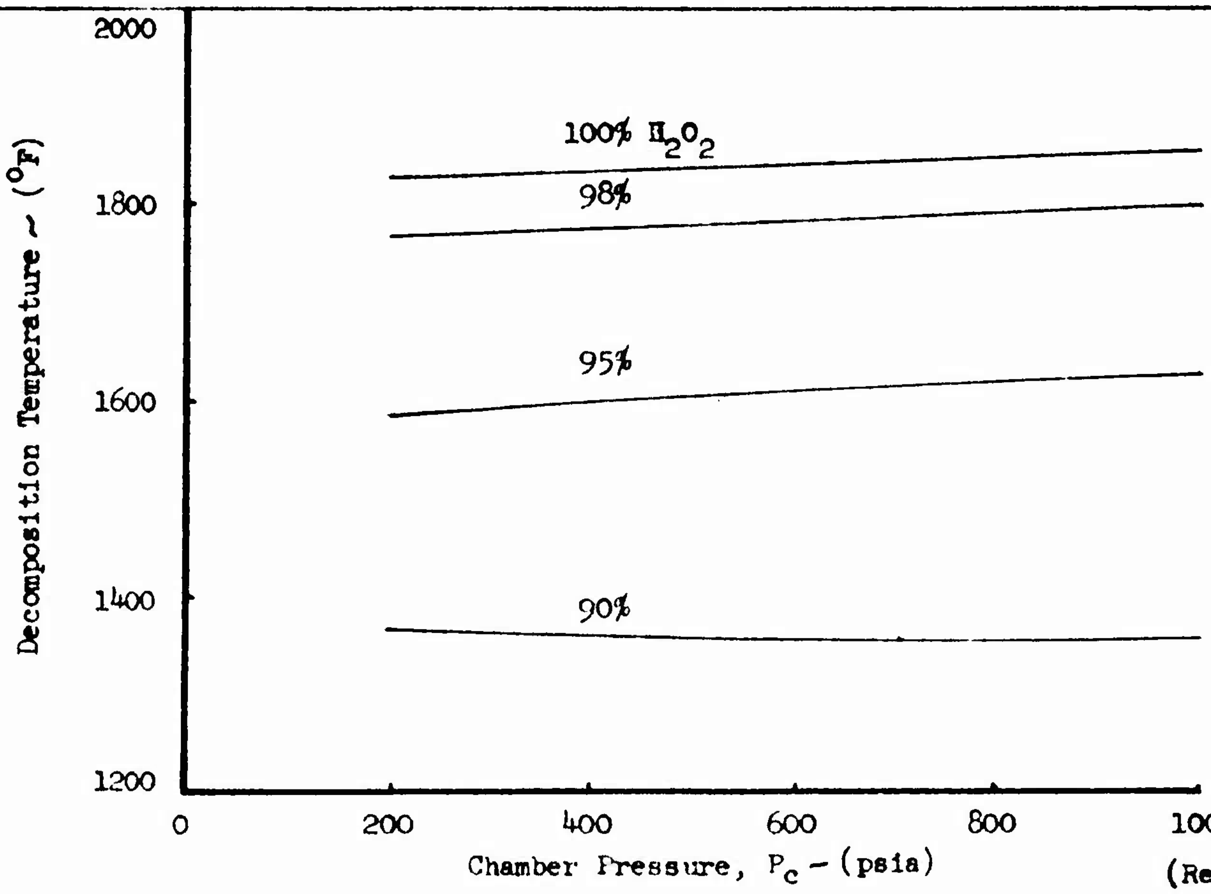

(Ref. 22)

FIGURE 6.3-3 HYDROGEN PEROXIDE DECOMPOSITION TEMPERATURE

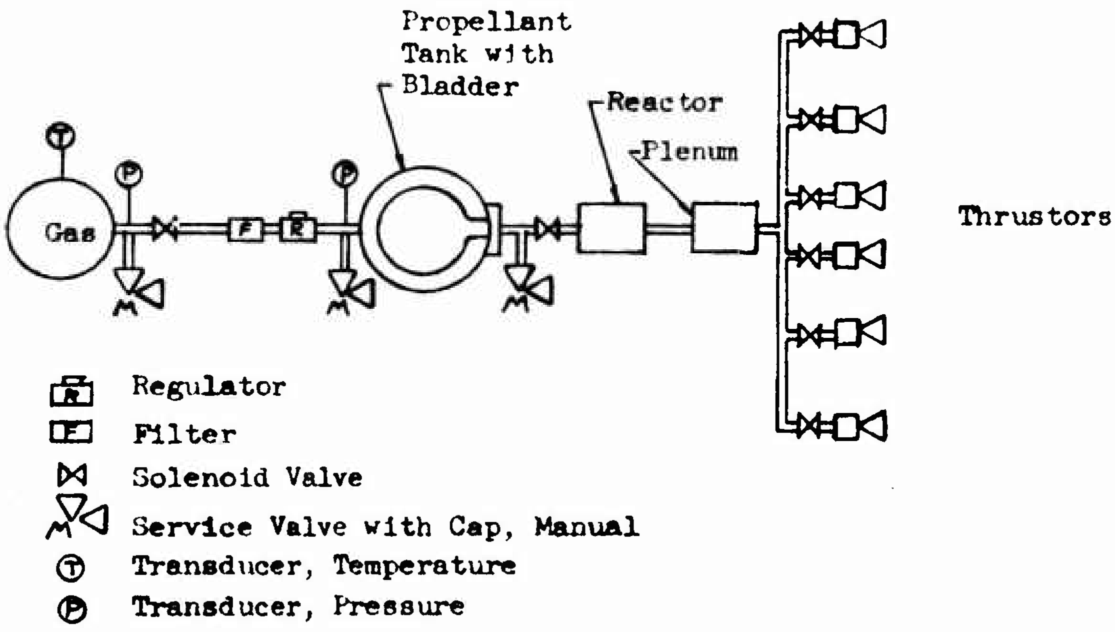

FIGURE 6.3-4 HYDRAZINE PLENUM SYSTEM

Inspection of this relationship shows that heat release, hence adiabatic temperature, is highest with no dissociation and decreases with increasing dissociation. This situation is shown in Figure 6.3-5. The molecular weight of exhaust gases and the chamber temperature also are shown to decrease with increasing ammonia dissociation.

The designer can control decomposition and dissociation to a certain extent by influencing flow variables and engine geometry. These factors are also affected by the type of catalyst and how it is used. The "spontaneous" nature of the Shell 405 catalyst (i.e., it needs no other means of ignition) makes it a leading candidate in hydrazine applications either by itself or in combination with other catalysts. Propellant decomposition is controlled by the exposure it has to the catalyst. Thus, important catalyst design parameters are area, catalyst bed length and ratio of propellant flow rate to catalyst bed cross-sectional area. Ammonia dissociation is also influenced by these factors, especially with the Shell 405 catalyst. A specific catalyst bed length is required for 100% hydrazine decomposition under given conditions. However, ammonia dissociation will also commence in the catalyst bed to the extent that there can be 30% or more dissociation by the time that 100% hydrazine decomposition has occurred. This means that reaction temperatures of 2200°F or less can be expected which is convenient in that it permits relatively simple radiation cooled engine designs with currently available materials.

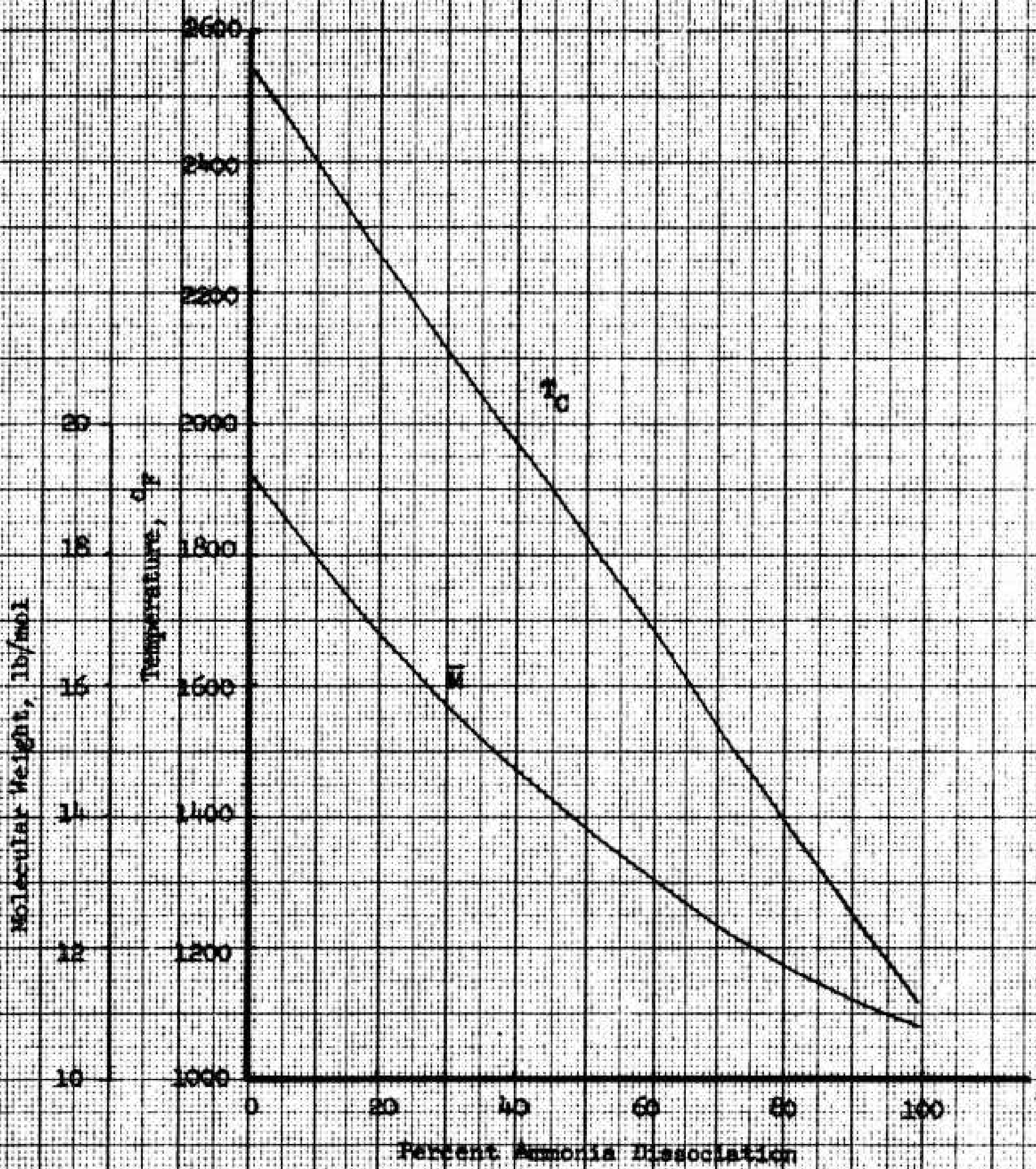

Figure 6.3-5 HYDRAZINE DECOMPOSITION GAS TEMPERATURE AND MOLECULAR WEIGHT

(Ref. 1)

6.3.3 MONOPROPELLANT THRUSTOR CONSTRUCTION

The general design features of hydrogen peroxide or hydrazine monopropellant thrustors are quite similar. Downstream from the propellant flow control valve is an inlet and manifold for distributing propellant to the injector. Adjacent to the injector is a "catalyst bed" comprised of materials which sustain propellant decomposition by catalysis. It is desirable that the catalyst also initiate the catalytic process through other devices are sometimes necessary for ignition. These methods either raise catalyst bed temperature to a level which will promote decomposition or a brief hypergolic reaction is initiated. This is done with electric or isotope heaters or by introducing small amounts of oxidizer to react hypergolically with the propellant. The catalyst bed is retained by structural baffles through which exhaust passes into the convergent-divergent nozzle section.

The specific design features of hydrogen peroxide and hydrazine monopropellant thrustors are sufficiently different to be discussed individually.

6.3.3.1 HYDROGEN PEROXIDE THRUSTOR CONSTRUCTION

A monopropellant hydrogen peroxide thrustor consists of an injector, thrust chamber, catalyst bed with catalyst, nozzle and, commonly, the propellant valve. The thrust chamber with catalyst are frequently referred to as the reactor. The catalyst bed consists of a metal catalyst in screen form, stacked in such a manner as to promote even flow distribution across the catalyst area.

A typical hydrogen peroxide thrustor is shown in Figure 6.3-6. This thrustor uses a "staged" catalyst bed design in which different materials, construction or flow direction are involved in different phases of the decomposition process. Initially, propellant is admitted to the chamber by the propellant flow control valve, often referred to as the "engine valve." Propellant is distributed

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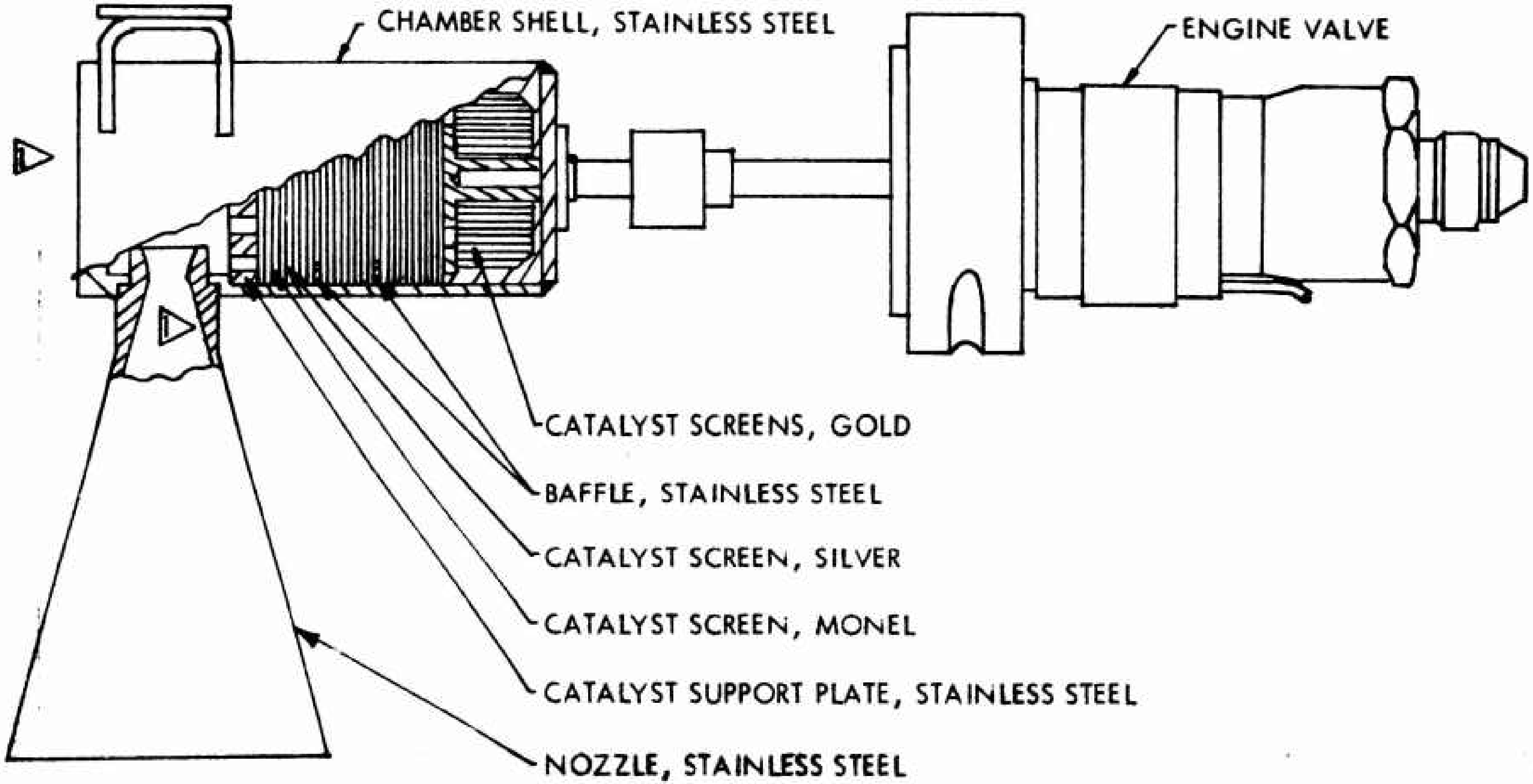

1 Thruster can be designed for nozzle installation at either location

FIGURE 6.3-6 TYPICAL HYDROGEN PEROXIDE THRUSTOR

(Ref. 23)

radially from a central injector into a gold catalyst screen to preheat the engine and propellant. The gas flow is then directed through the main catalyst bed composed primarily of silver catalyst screens. Final catalyst screens are made of monel. Anti-channel baffles are used to prevent bypassing the catalyst bed at start-up. Retainer screens are installed in such a way as to prevent local "hot spots" which can damage the catalyst bed. These screens are held in place by a baffled support plate.

Structural materials for the chamber walls, "head" and "tail" ends, support plate, and nozzle sections are generally of 321 or 347 stainless steel used with welded construction. The 304L or 316L stainless steels can also be used. The heavier thrustors use 347 stainless steel due to its superior strength at elevated temperatures.

6.3.3.2 HYDRAZINE THRUSTOR CONSTRUCTION

A monopropellant hydrazine thrustor consists of an injector, thrust chamber, catalyst bed with catalyst, and nozzle. The thrust chamber and catalyst bed are called the reactor. The catalyst bed consists of a catalyst in granular form which is held in place by various screens and supports.

A typical hydrazine thrustor is shown in Figure 6.3.7. This thrustor uses a "staged" ("layered") catalyst bed design in which different types and sizes of catalytic material are arranged in the bed in layers to support a particular phase of the decomposition process. Propellant is admitted to the chamber by the engine valve to and through the injector into the catalyst bed. Injector designs for hydrazine engines vary by manufacturer such that baffled plate or showerhead-type injectors (Figure 6.3-7) are used by some while others use perforated probe or coil units which penetrate into the catalyst bed.

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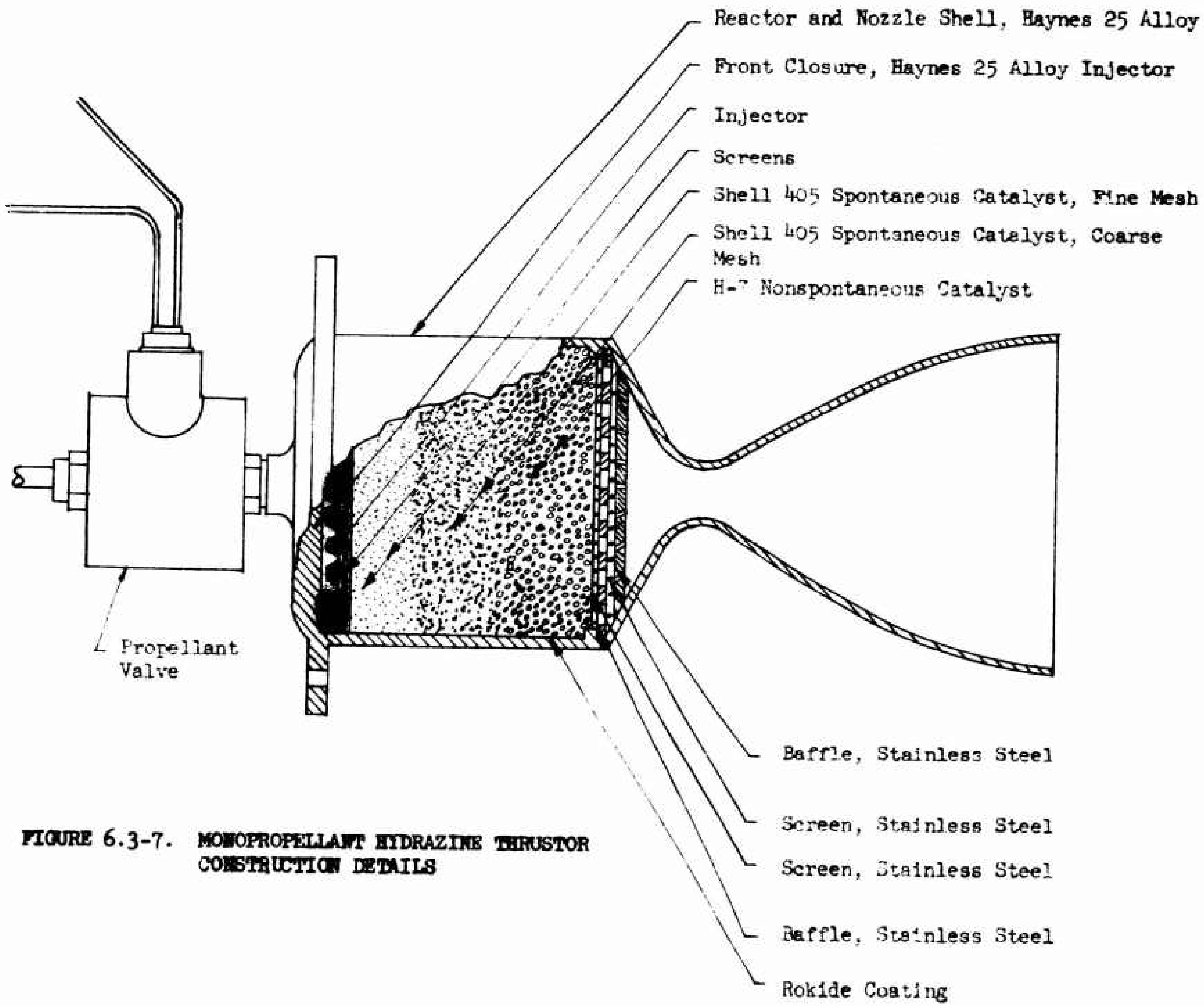


FIGURE 6.3-7. MONOPROPELLANT HYDRAZINE THRUSTOR CONSTRUCTION DETAILS

This thrustor design is based on the use of a "spontaneous" catalyst, such as Shell 405, which spontaneously initiates the decomposition process on contact, under ambient temperature conditions. Older catalysts, such as the H-7 used in Ranger and Mariner hydrazine engines used small quantities ("slugs") of nitrogen tetroxide (N_2O_4) reacting hypergolically for start-up. Other designs at that time involved electrical or isotopic heaters in the catalyst bed to raise bed temperature to a level at which it would initiate the decomposition process. The advent of spontaneous catalysts, especially the Shell 405 catalyst, has largely replaced these approaches.

The actual composition of the Shell 405 catalyst remains a classified item. Basically, it consists of a high surface area ceramic binder impregnated with metals which are catalytically very active. This arrangement is satisfactory for chamber temperatures of up to 1800°F, but the binder becomes structurally insufficient much beyond this point. Research has been, and is being, conducted to develop higher temperature binders for use with the higher performance hydrazine (N_2H_4)/hydrazinium nitrate ($N_2H_5NO_3$) mixtures which decompose at higher temperatures.

The Shell 405 catalyst originally cost about $1500 per pound, and the price has not changed significantly. It is available on a standard basis in 1/8 by 1/8 inch or 1/8 by 1/16 inch cylindrical pellets. It is also marketed in fines crushed from basic pellets in 10-30 mm granules and microspheres from 0-5 to several millimeters in diameter. However, the 1/8 inch cylindrical pellet is still the standard form. The 1/8 by 1/8 inch pellet is the maximum pellet size currently available.

The catalyst is arranged by layers in the reactor such that the finest, highest surface area particles are adjacent to the injector to promote smooth ignition and operation. These particles may be lightly fused in the first engine firing which promotes catalyst bed strength and lengthens catalyst life since very small particles can be easily dislodged from the engine. Sometimes the Shell 405 catalyst is used adjacent to the injector and backed up by the conventional H-7 catalyst to save cost and extend life. In this case, the Shell catalyst is used for the spontaneous ignition feature and the H-7 catalyst is used for extended life requirements.

Chamber construction is usually of thin-wall Haynes 25 alloy for the chamber walls, front and aft closures, and nozzle assembly. Sometimes, in small engines, these parts are made of 347 or similar stainless steels. Interior chamber walls are coated with Rokide or similar substances. Catalyst screens and support plates are made from 310 or similar stainless steels. Injectors are usually tubes or plates of aluminum alloys carefully selected and designed to survive the thermal conditions accompanying and following operation.

Engine design generally permits welding the aft closure-nozzle section to the chamber wall and inserting the catalyst support plate, catalyst, and upper screen. The forward closure, injector plates and/or tubes are installed in place, and this assembly mated with the upper chamber wall and welded at this point. A design preference has involved placing the engine mounting lugs at the forward closure weld for a thermal relief path after engine operation. However, the mounting ring has also been located upstream of the injector head in the vicinity of the engine valve. Location of the engine valve and inlet manifold is sometimes extended farther upstream of the catalyst face for thermal standoff.

6.3.4 MONOPROPELLANTS

Numerous compounds, or mixtures of compounds, have been suggested for use as monopropellants. However, many have been discarded in view of the stability and handling requirements of the operational situation. The most frequently mentioned monopropellants include hydrogen peroxide, hydrazine, hydrazine-hydrazinium nitrate, ethylene oxide, nitromethane and tetranitromethane. Of these, hydrogen peroxide and hydrazine are the only monopropellants to receive extensive use. The characteristics of these monopropellants, and their variations, will be separately discussed in the following sections.

6.3.4.1 HYDROGEN PEROXIDE (H_2O_2)

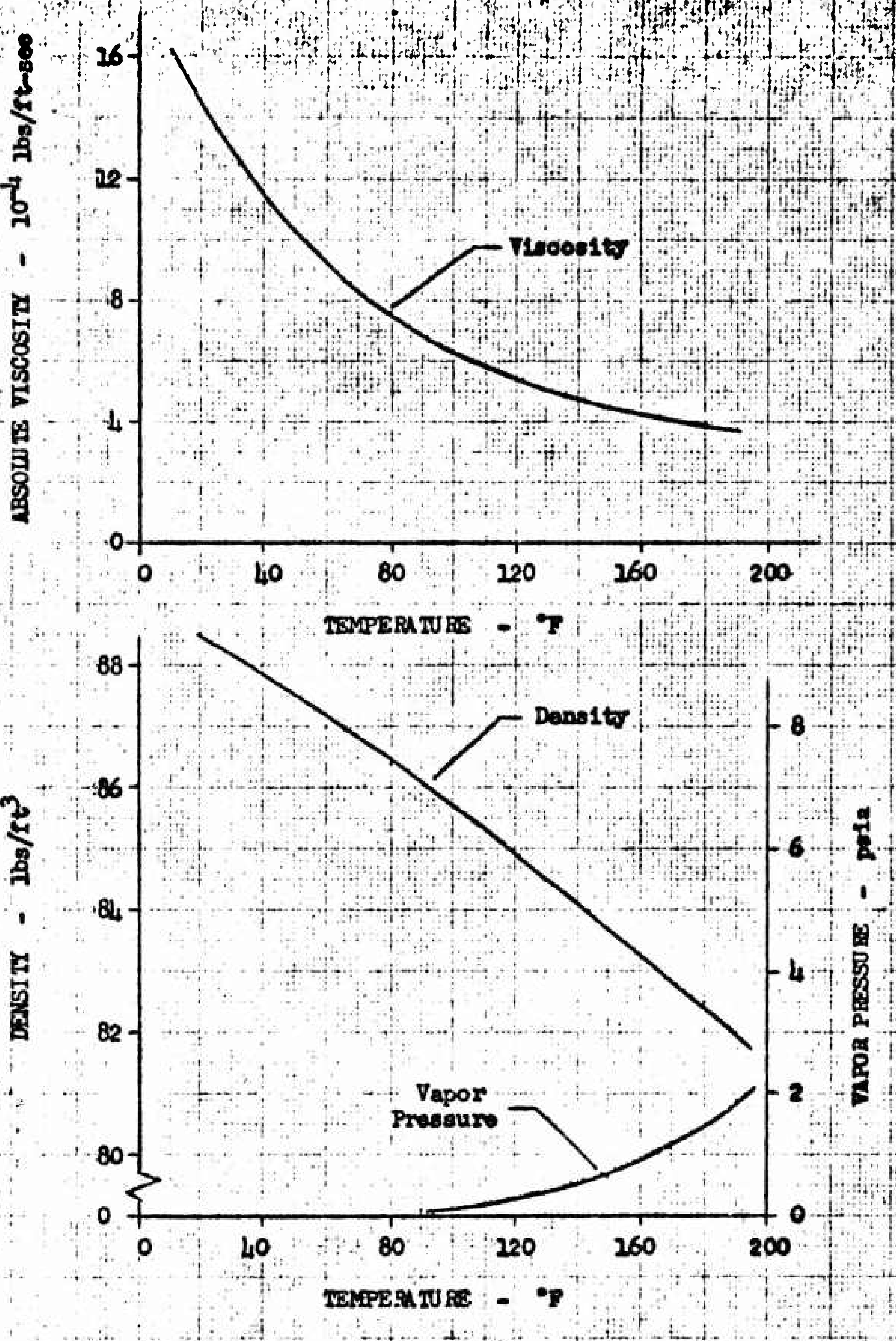
Hydrogen peroxide has been used to drive turbines in both rocket engines and APU's and as a monopropellant in attitude and velocity control systems. Its performance is directly related to the percent of peroxide in the peroxide-water solution. Currently, 90% peroxide is used most frequently though increasing emphasis is being placed on 98%. Characteristics of both varieties are shown in Table 6.3-1. Peroxide solutions are insensitive to the initiation and propagation of detonations. Although the decomposition products are oxidizing, the flame temperature is low enough to preclude significant materials problems. Decomposition is usually initiated by a suitable catalyst such as permanganate salts, or activated silver screens. Storage and handling of peroxide is complicated by the fact that it reacts to some extent with almost every substance. Peroxide systems must be kept quite clean to prevent accidental catalysis. Thus it is necessary to clean and chemically treat materials that will be exposed to the liquid. Even so, there will be a slight concentration loss (0.5-1.0% per year) during long storage periods. Absolute viscosity, density and vapor pressure of 90% hydrogen peroxide are shown in Figure 6.3-8 as a function of temperature. Density is shown in Figure 6.3-9 as a function of peroxide concentration.

TABLE 6.3-1

PROPERTIES OF HYDROGEN PEROXIDE

		Concentration by Weight		
		90%	98%	100%
Average Molecular Weight		31.241	33.422	
Normal Boiling Point	°R	746.2	759.2	
Normal Freezing Point	°R	471.3	487.5	
Density (@ 537°R)	g/cc	1.383	1.431	
Vapor Pressure (@ 537°R)	PSI	.0735	.0426	
Critical Pressure	PSI			3144.98
Critical Temperature	°R			1318
Dielectric Constant		77	77	
Electrical Conductivity (@ 537°R)	$ohm^{-1}cm^{-1}$	$1,9(10^{-6})$	$0.8(10^{-6})$	
Fire Point				
Flash Point				
Heat Capacity, Liquid (492-533°R)	Btu/lb°F			58.0
Heat of Decomposition (@ 537°R)	Btu/lb SOLID	1108	1215	
Heat of Formation, Liquid (@ 537°R)	Btu/lb			2369
Heat of Fusion	Btu/lb SOLID			158.1
Heat of Dilution to Infinite (@ 537°R)	"	-31.2	-41.0	
Heat of Vaporization, Total, SOLID	Btu/lb	700.3	662.0	
Viscosity (@537°R)	Centipoise	1.153	1.155	

(Refs. 22 and 24)



(Ref. 4)

CALC			REVISED	DATE	PHYSICAL PROPERTIES 90% HYDROGEN PEROXIDE	FIGURE 6.3-8
CHECK						
APR						
APR					THE BOEING COMPANY	PAGE 94

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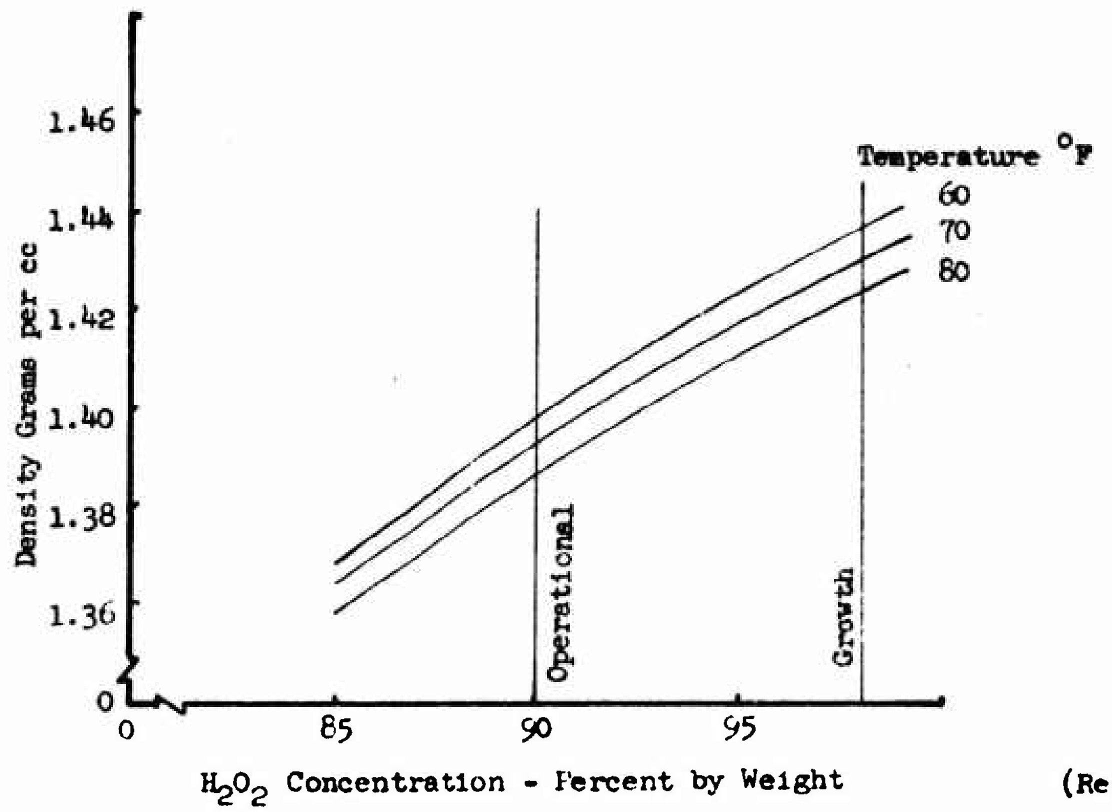


(Ref. 25)

FIGURE 6.3-9 DENSITY OF HYDROGEN PEROXIDE

6.3.4.2 HYDRAZINE (N_2H_4)

Hydrazine has been used as a turbine-drive gas, as well as a monopropellant in attitude and velocity control systems. It is a clear, colorless, hygroscopic, toxic, flammable, caustic liquid and a strong reducing agent. "Neat" or anhydrous hydrazine used in rocket engines is controlled by MIL Spec MIL-P-26536B. Physical properties of hydrazine are listed in Table 6.3-2. Density, specific heat, vapor pressure and viscosity of hydrazine are shown in Figure 6.3-10 as a function of temperature.

Hydrazine is compatible with most stainless steels, aluminum, tantalum, titanium, Haynes 25 alloy, teflon, polyethylene, glass, butyl rubber, and ethylene propylene.

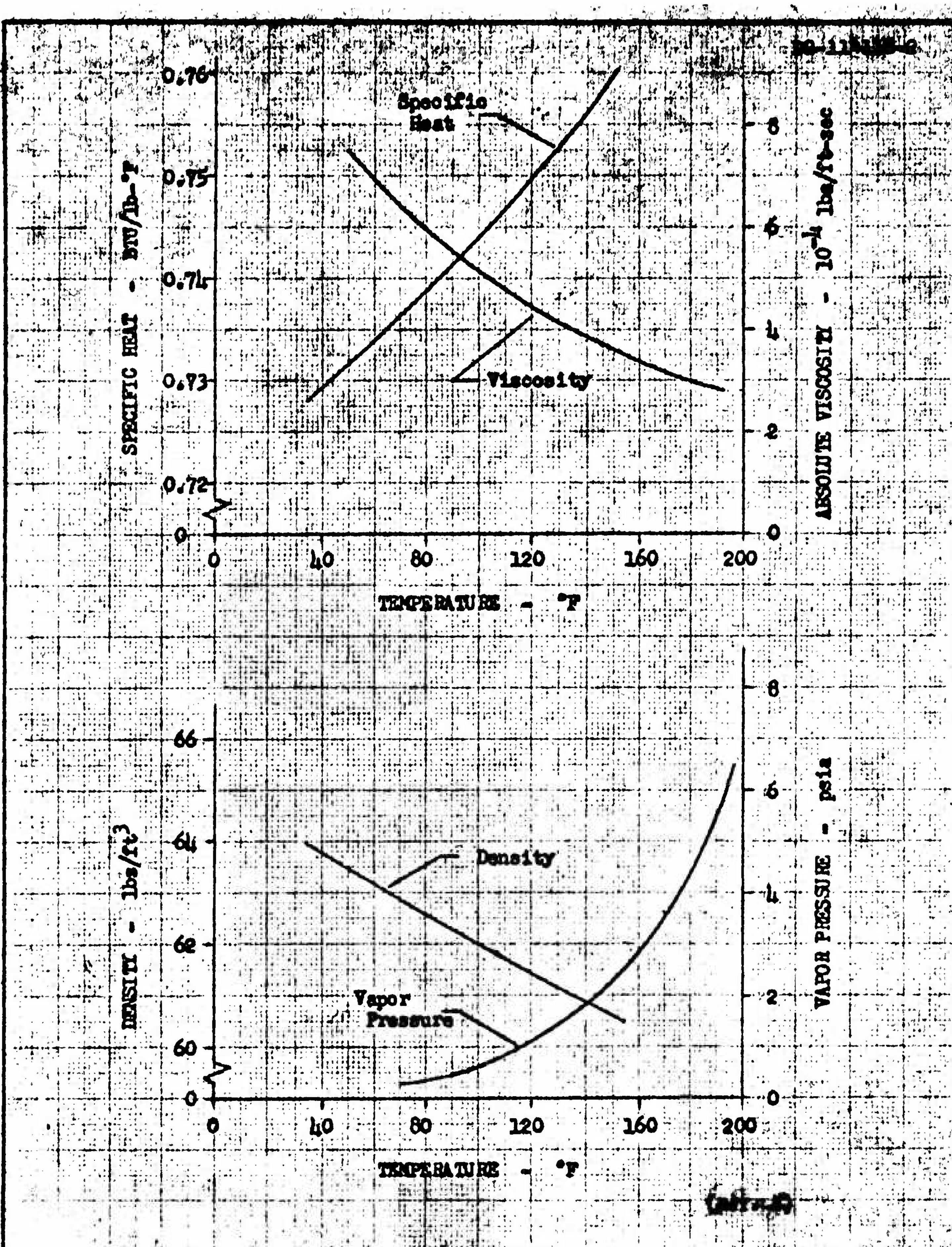
Considerable research effort has been expended on reducing the relatively high freezing point of hydrazine (35.6°F) by adding water, ammonia, or hydrazine nitrate. Simple binary or tertiary solutions of these compounds form low freezing point eutectics. These mixes also change other characteristics of the propellants, including density and performance. Figure 6.3-11 shows how hydrazine freezing point changes with water addition. The density variation of solutions of hydrazine, hydrazine nitrate, and water are shown in Figure 6.3-12 as a function of temperature. The physical properties of hydrazine-ammonia solutions are shown in Figure 6.3-13 as a function of ammonia content.

TABLE 6.3-2

PROPERTIES OF HYDRAZINE

Molecular Formula	N_2H_4
Molecular Weight	32.05
Normal Boiling Point	696.3°R
Normal Freezing Point	495.6°R
Density (@ 528°R)	1.0083 g/cc
Vapor Pressure (@ 537°R)	0.28 psia
Critical Pressure	2132 psia
Critical Temperature	1176°R
Dielectric Constant (@ 537°R)	51.7
Electrical Conductivity (@ 537°R)	2.3-2.8(10^{-6}) ohm^{-1}
Fire Point (Tag Open Cup)	585.6°R
Flash Point (Tag Open Cup)	585.6°R
Heat Capacity (liquid) (@ 537°R)	0.737 Btu/lb°R
Heat of Combustion (to N_2 + $2H_2O$ liq)(@ 537°R)	-8,359 Btu/lb
Heat of Formation, Liquid (@ 537°R)	676 Btu/lb
Heat of Fusion (@ 495.5°R)	170 Btu/lb
Heat of Solution, Liquid (@ 537°R)	-219.6 Btu/lb
Heat of Vaporization (@ 696.3°R)	540 Btu/lb
Viscosity (@ 537°R)	0.90 centipoises

(Refs. 24 and 26)



CALC			REVISED	DATE	PHYSICAL PROPERTIES HYDRAZINE	FIGURE 6.3-10
CHECK						
APR						
APR					THE BOEING COMPANY	PAGE 98

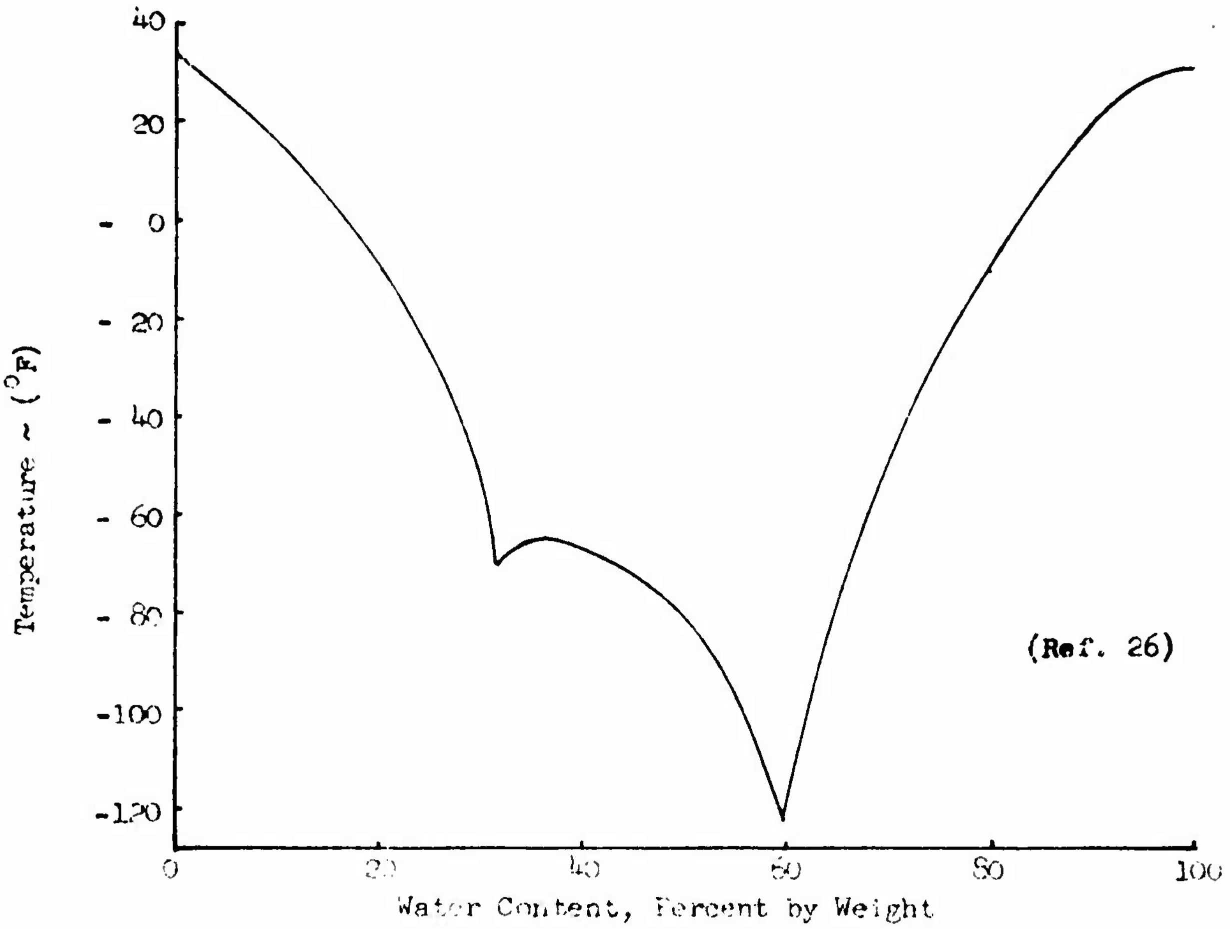


FIGURE 6.3-11 FREEZING POINT OF HYDRAZINE AND WATER SOLUTIONS

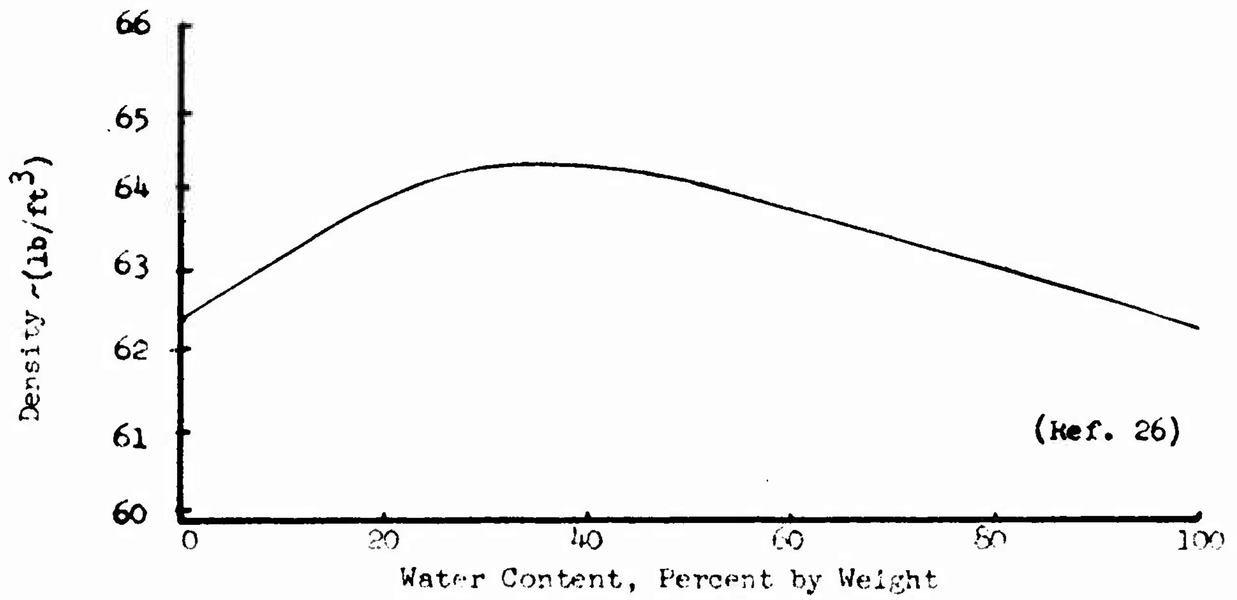


FIGURE 6.3-12 SPECIFIC GRAVITY OF HYDRAZINE AND WATER SOLUTIONS

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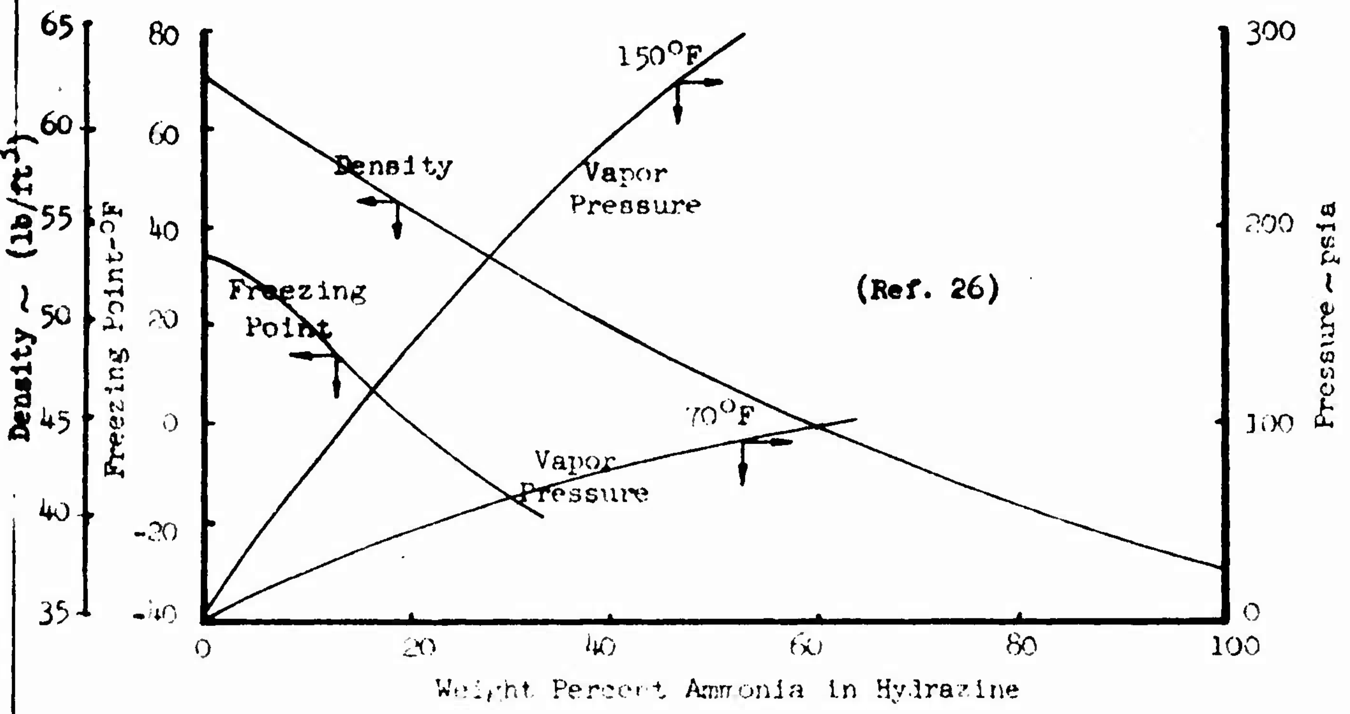

FIGURE 6.3-13 PHYSICAL PROPERTIES OF HYDRAZINE AND AMMONIA SOLUTIONS

6.3.5 MONOPROPELLANT PERFORMANCE

The performance of hydrogen peroxide and of hydrazine are compared in a general fashion in Figure 6.3-14 as a function of duty cycle and pulse width. The superior performance of hydrazine is apparent, especially as operation approaches steady state conditions. More detailed performance information for these propellants is provided in the following sections.

6.3.5.1 PERFORMANCE - HYDROGEN PEROXIDE

Hydrogen peroxide decomposes exothermally to superheated oxygen gas and water vapor. Figure 6.3-3 shows the decomposition gas temperature of hydrogen peroxide, in various concentrations, as a function of chamber or reactor pressure.

The theoretical specific impulse of hydrogen peroxide is shown in Figure 6.3-15 as a function of peroxide concentration. Propellant gamma is related to peroxide concentration in Figure 6.3-16. The performance improvement possible with 98% peroxide, including the increased specific impulse, and increased propellant density for equal volume applications, is approximately 13%.

Delivered specific impulse under steady state conditions with 90% hydrogen peroxide is shown in Figure 6.3-17 as a function of nozzle expansion ratio. In pulsing operation, performance can be expected to follow that shown in Figure 6.3-18 in which delivered specific impulse is related to pulse length and the spacing between pulses. Figure 6.3-19 shows how several different engines using 90% hydrogen peroxide perform in pulsed operation.

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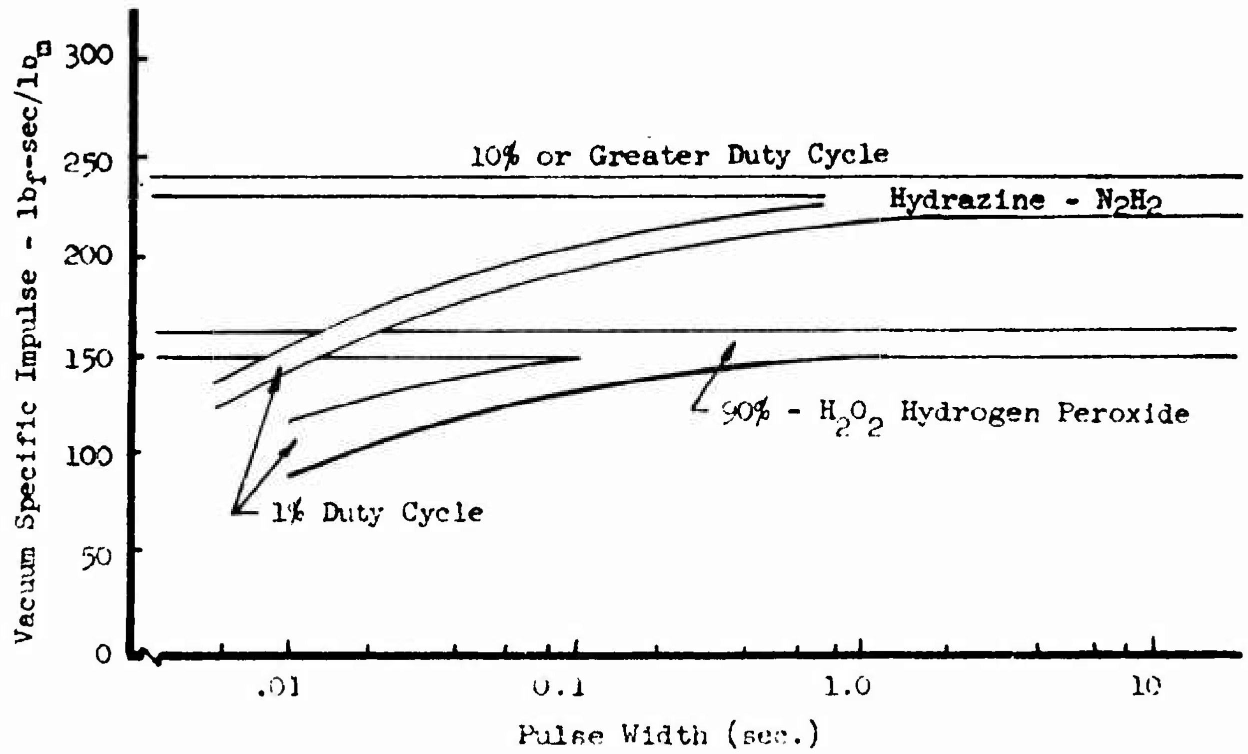
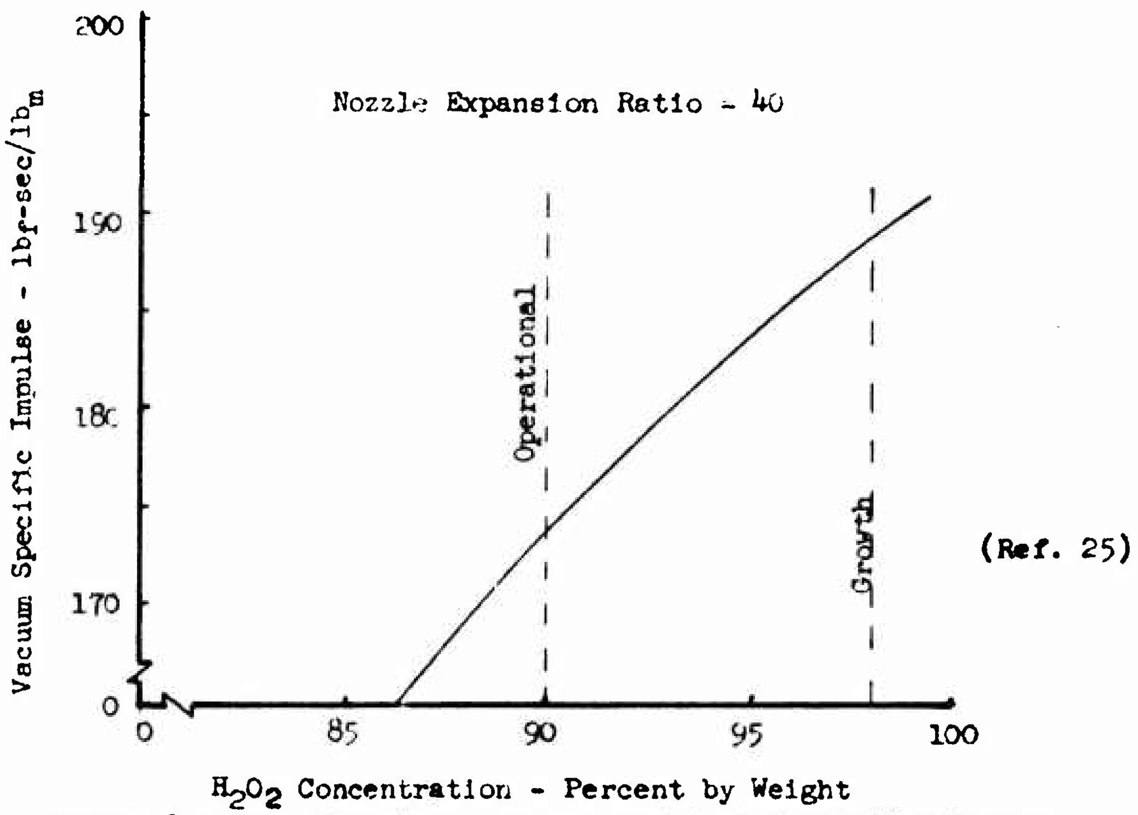


FIGURE 6.3-14. PERFORMANCE OF COMMON MONOPROPELLANTS



(Ref. 25)

FIGURE 6.3-15. THEORETICAL PERFORMANCE OF HYDROGEN PEROXIDE

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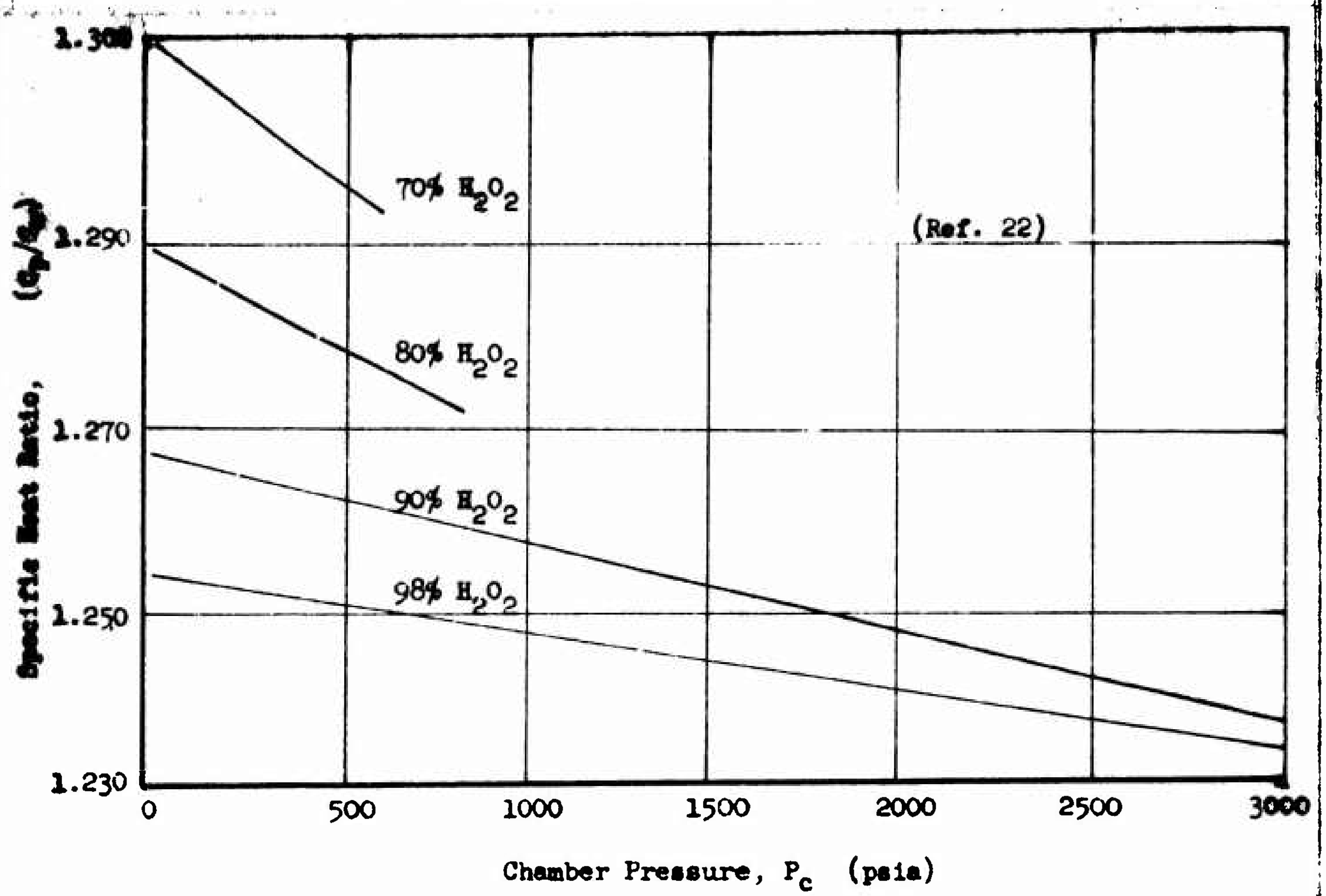

Figure 6.3-16 SPECIFIC HEAT RATIO OF HYDROGEN PEROXIDE

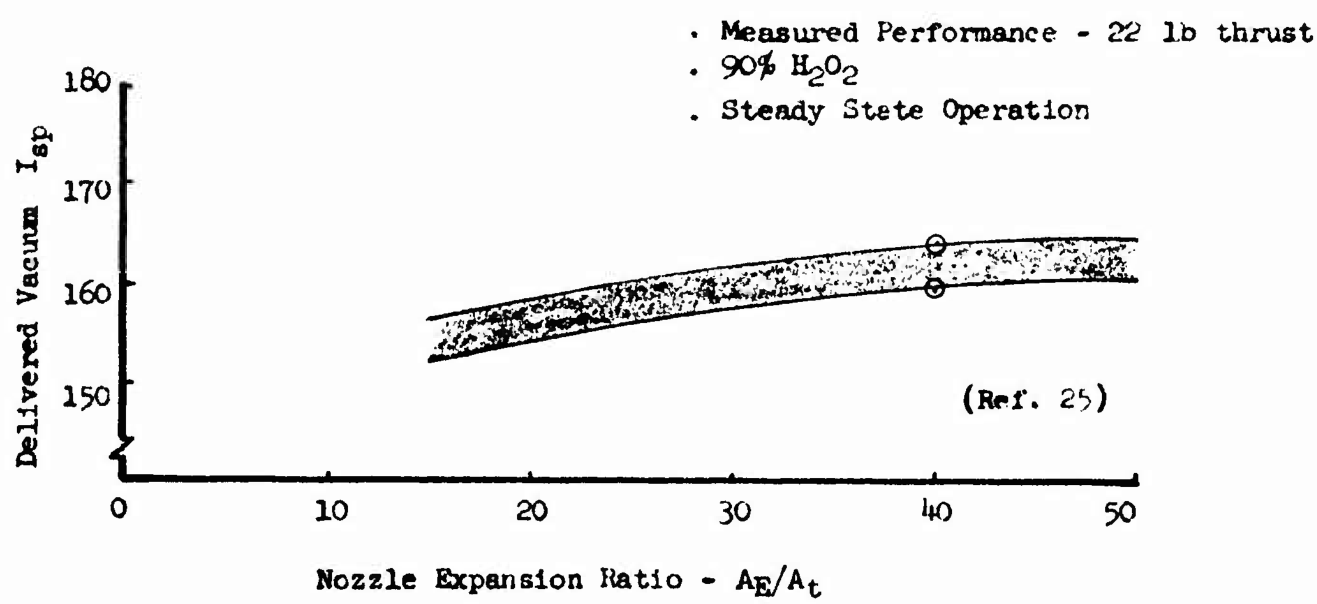

FIGURE 6.3-17 HYDROGEN PEROXIDE DELIVERED PERFORMANCE

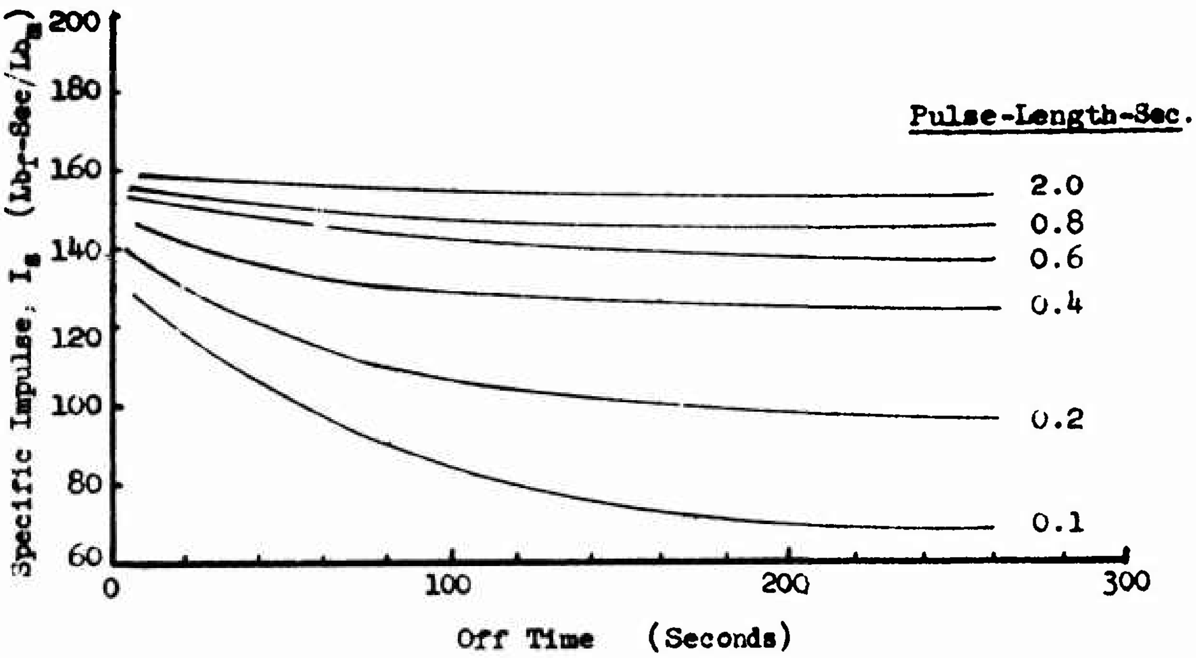

FIGURE 6.3-18 90 PERCENT HYDROGEN PEROXIDE PERFORMANCE IN PULSED OPERATION (Ref. 4)

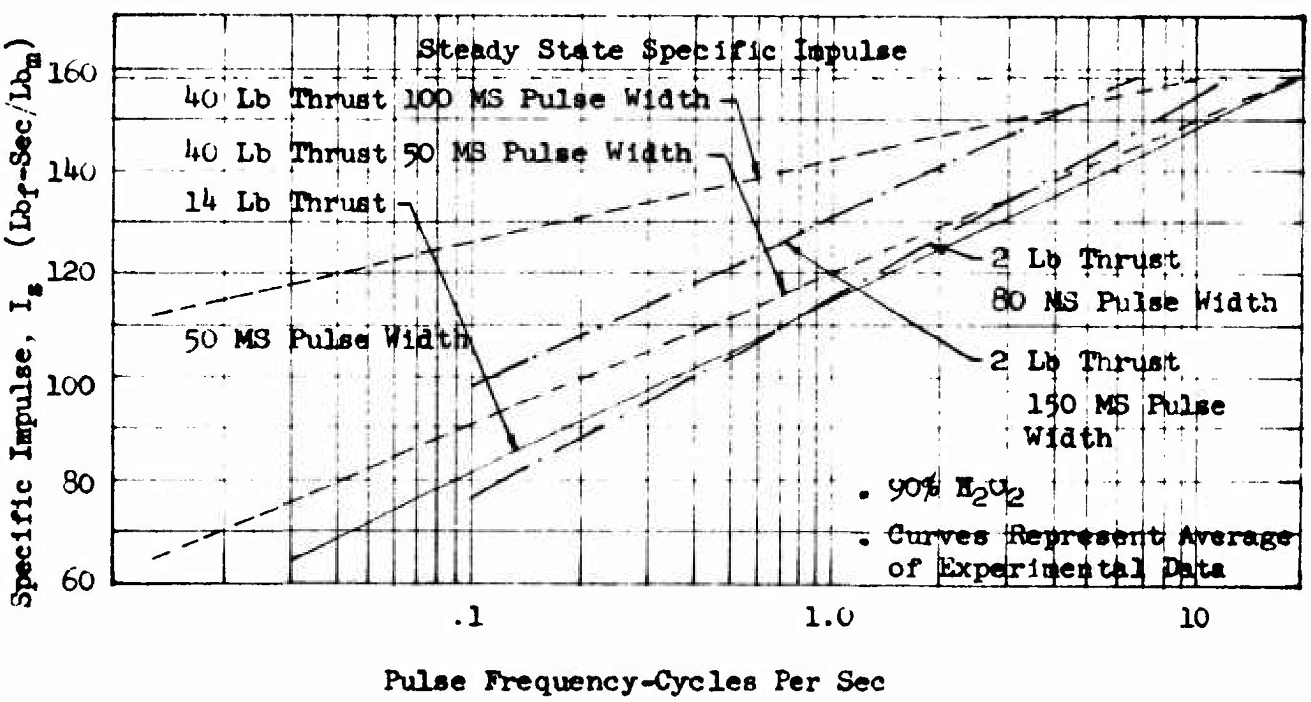

FIGURE 6.3-19 HYDROGEN PEROXIDE PERFORMANCE IN PULSED OPERATION - DEMONSTRATED (Ref. 2)

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6.3.5.2 **PERFORMANCE - HYDRAZINE**

Hydrazine decomposes exothermally to nitrogen gas and ammonia which partially dissociates to nitrogen and hydrogen. This decomposition reaction and dissociation process are related in expression 6.3-1 to the parameter, X, representing the fraction of ammonia dissociation. Figure 6.3-20 shows the performance of 100% hydrazine (N_2H_4) in terms of specific impulse, characteristic velocity (C*), chamber temperature, and exhaust product mole weight and composition.

The performance actually delivered by monopropellant engines is affected primarily by nozzle expansion ratio, engine thrust level, and whether the engine was designed for steady-state or pulsing operation. Figure 6.3-21 shows delivered specific impulse as a function of thrust level for many different hydrazine engine designs. In preliminary design exercises, delivered specific impulse (vacuum, steady state) is generally assumed as 230-235 lbf-sec/lbm with engines larger than one pound thrust. Performance of very small engines may be assumed as low as 210 to 215 lbf-sec/lbm. These values are based on "hot bed" results. Cold bed performance is substantially lower, extending possibly to the cold bed temperature limit of approximately 118 lbf-sec/lbm for a 60°F "cold" catalyst bed.

Performance of a typical hydrazine engine under pulsed operation is shown in Figure 6.3-22 as a function of pulse width and the spacing between pulses.

Minimum impulse bit capability of hydrazine engines is strongly affected by the engine-valve relationship, and by reactor design "learning curve" effects. Minimum reproducible impulse bits (lb-sec) equal to 0.005 times the nominal thrust rating have recently been demonstrated in hydrazine engines from 2 to 50 lbs of thrust. Minimum reproducible impulse bit values ($I_{T_{min}}$) are shown in Figure 6.3-23 for existing hydrazine engine designs.

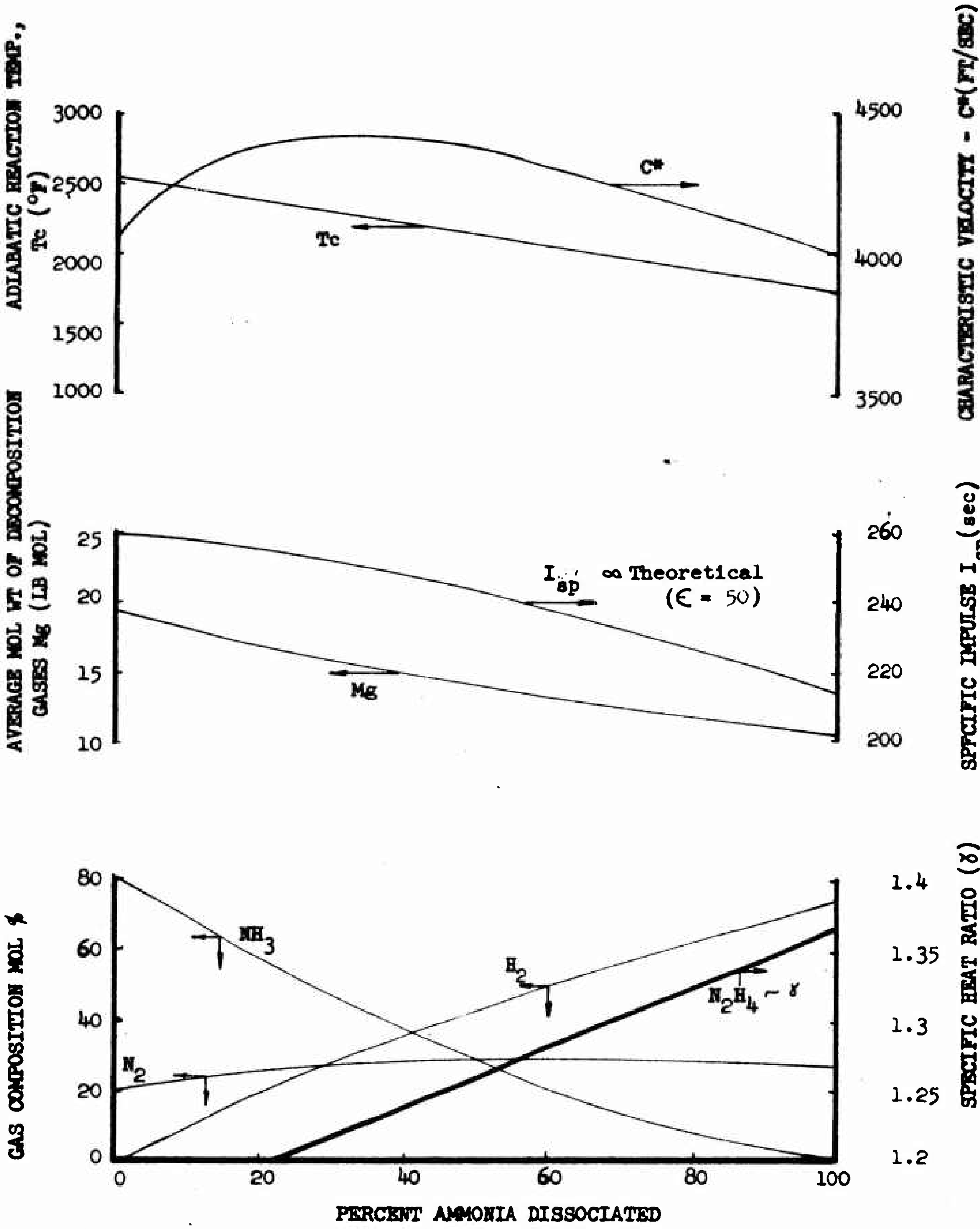

FIGURE 6.3-20 HYDRAZINE PERFORMANCE CURVES (100% N_2H_4) (Ref. 26)

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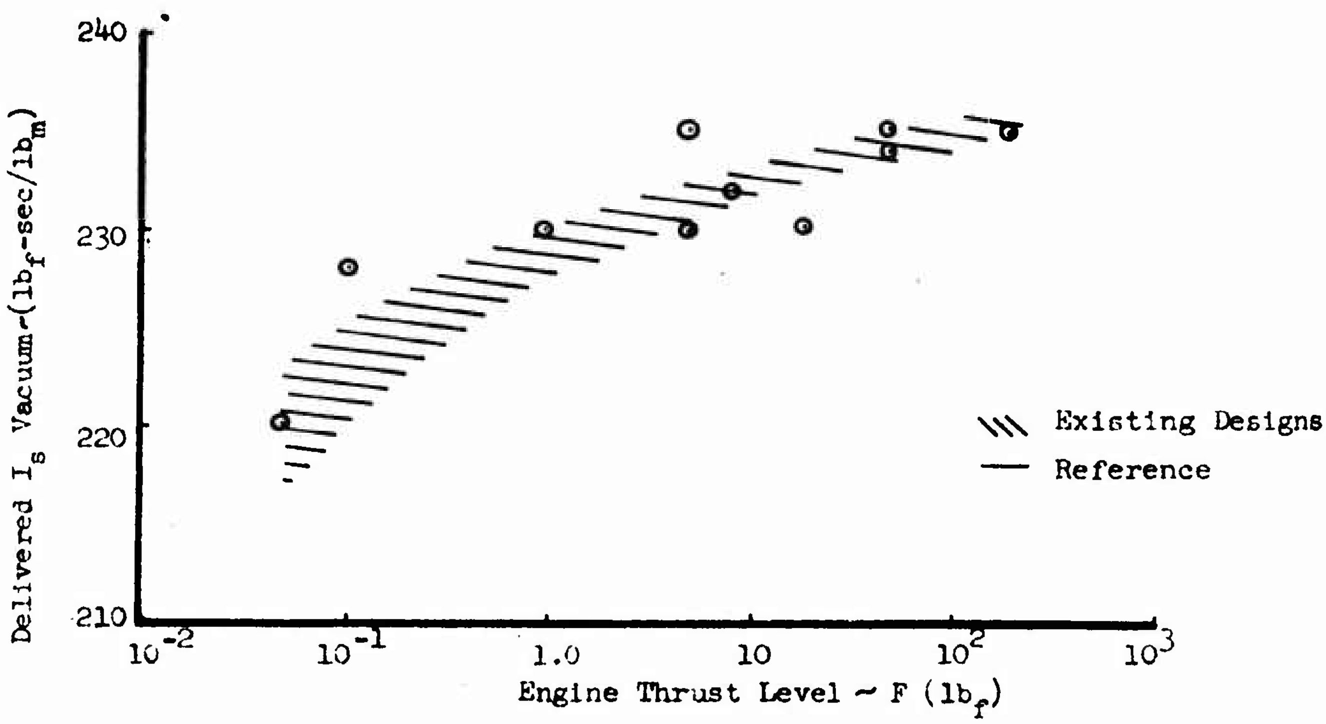

6.3-21 STEADY STATE PERFORMANCE DELIVERED BY HYDRAZINE MONOPROPELLANT ENGINES

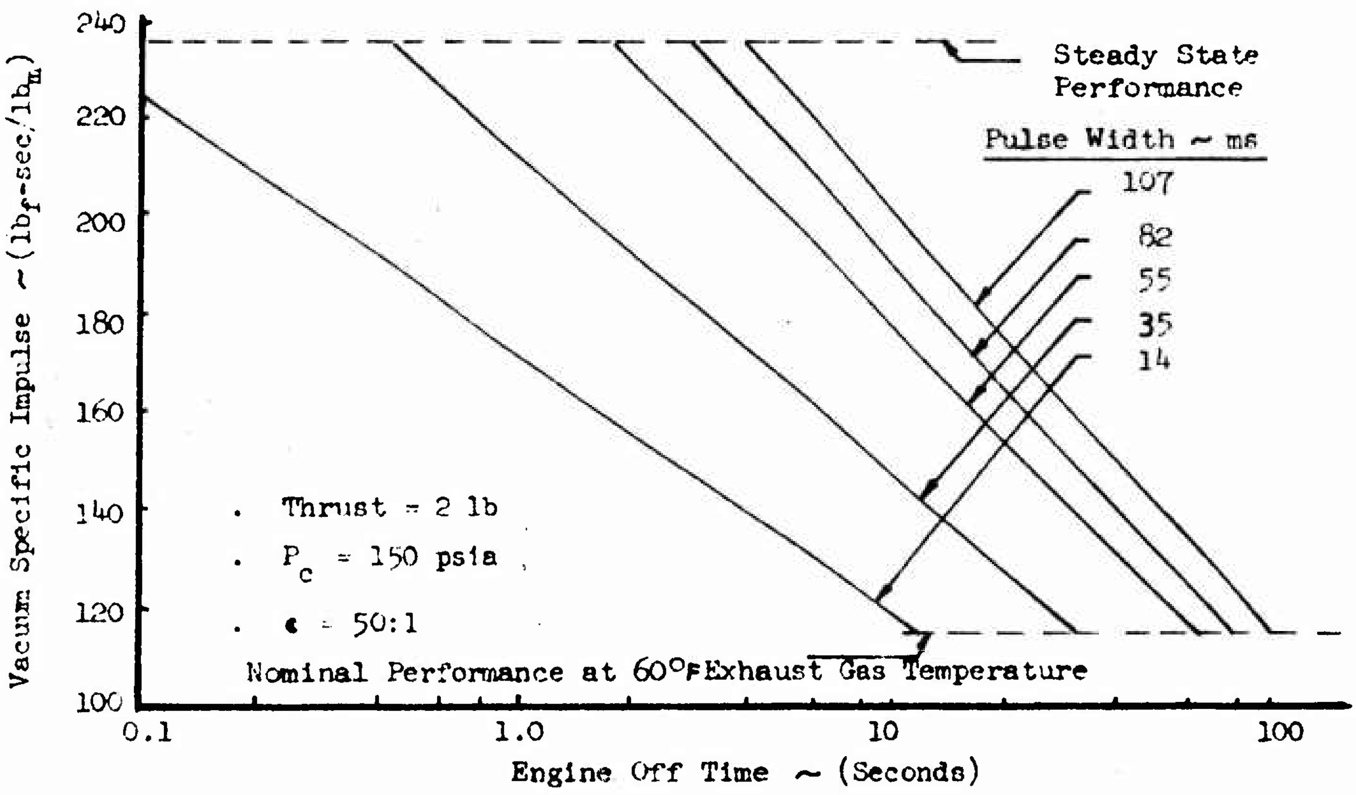

FIGURE 6.3-22 HYDRAZINE MONOPROPELLANT ENGINE PULSE PERFORMANCE CAPABILITY (Ref. 26)

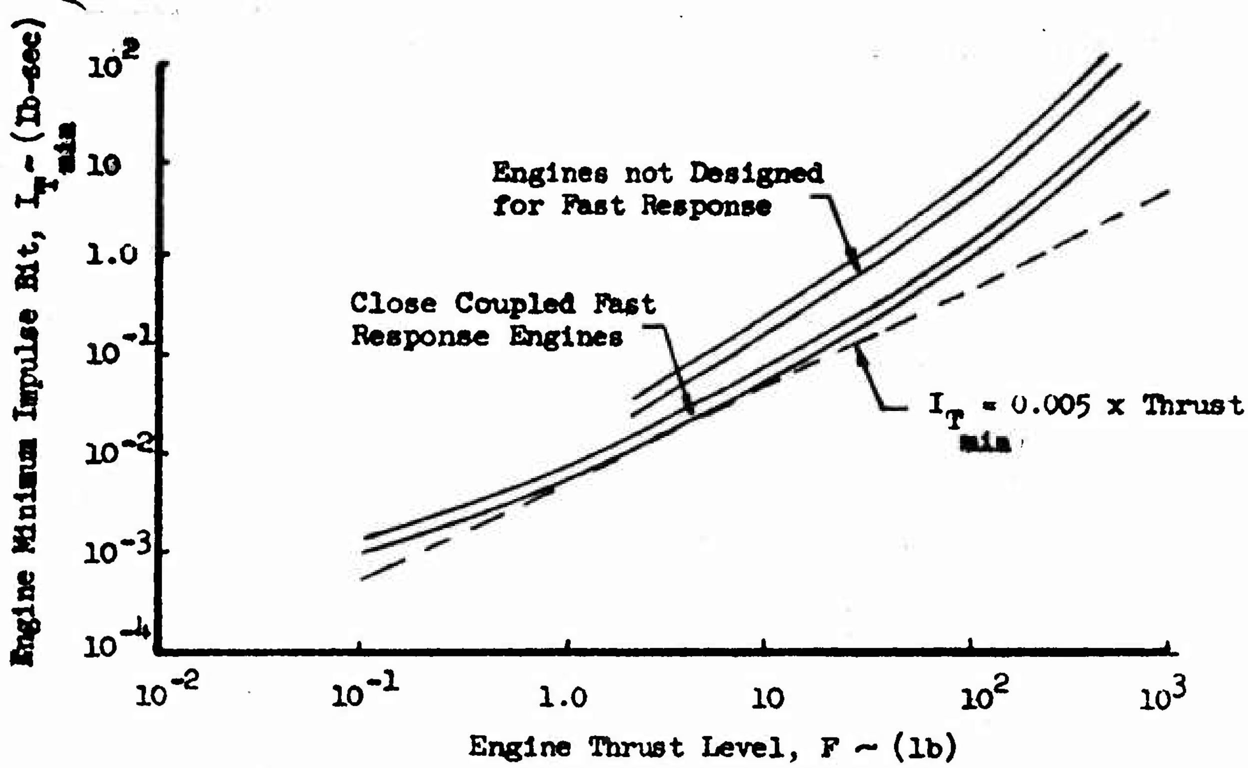

FIGURE 6.3-23 MINIMUM IMPULSE BIT CAPABILITY OF HYDRAZINE ENGINES

Caution should be exercised in selecting design values of $I_{T_{min}}$, since fast response systems can be substantially more expensive. This greater expense is due to more involved development programs, and selective engine delivery. Values of $I_{T_{min}}$ may be increased considerably where mutliple engine valves are used for reliability purposes.

Impulse tolerance is usually specified in conjunction with minimum impulse requirements. These operations are always time related, and are thus sensitive to timing errors. As a general rule, engine minimum impulse tolerances may be estimated as $\pm$ 10% of the minimum impulse bit, and are commonly stated that way. Some improvement is possible though, when necessary. Mariner II (Mariner R) which was accelerometer controlled was capable of $\pm$ 5% tolerance on the minimum impulse bit. Where possible, it is suggested that impulse tolerance values shown in Figure 6.3-24 be used for design purposes.

Hydrazine performance can be significantly improved by mixing it with hydrazinium nitrate ($N_2H_5NO_3$). Unfortunately, reaction temperatures of the mixture (2500°F) exceed the capability of the spontaneous catalyst binder, although they are within the capability of H-7 catalyst. Hydrazinium nitrate is also quite shock sensitive. Performance, decomposition temperature, shock sensitivity and propellant freezing point can be improved, however, by mixing the hydrazine and hydrazinium nitrate with water. Figure 6.3-25 is a ternary diagram of the mixture showing freezing point, specific impulse, and shock sensitivity as a function of the amount of each substance involved. Fair stability has been found with mixtures containing up to 15% hydrazinium nitrate, but shock sensitivity increases rapidly beyond this point.

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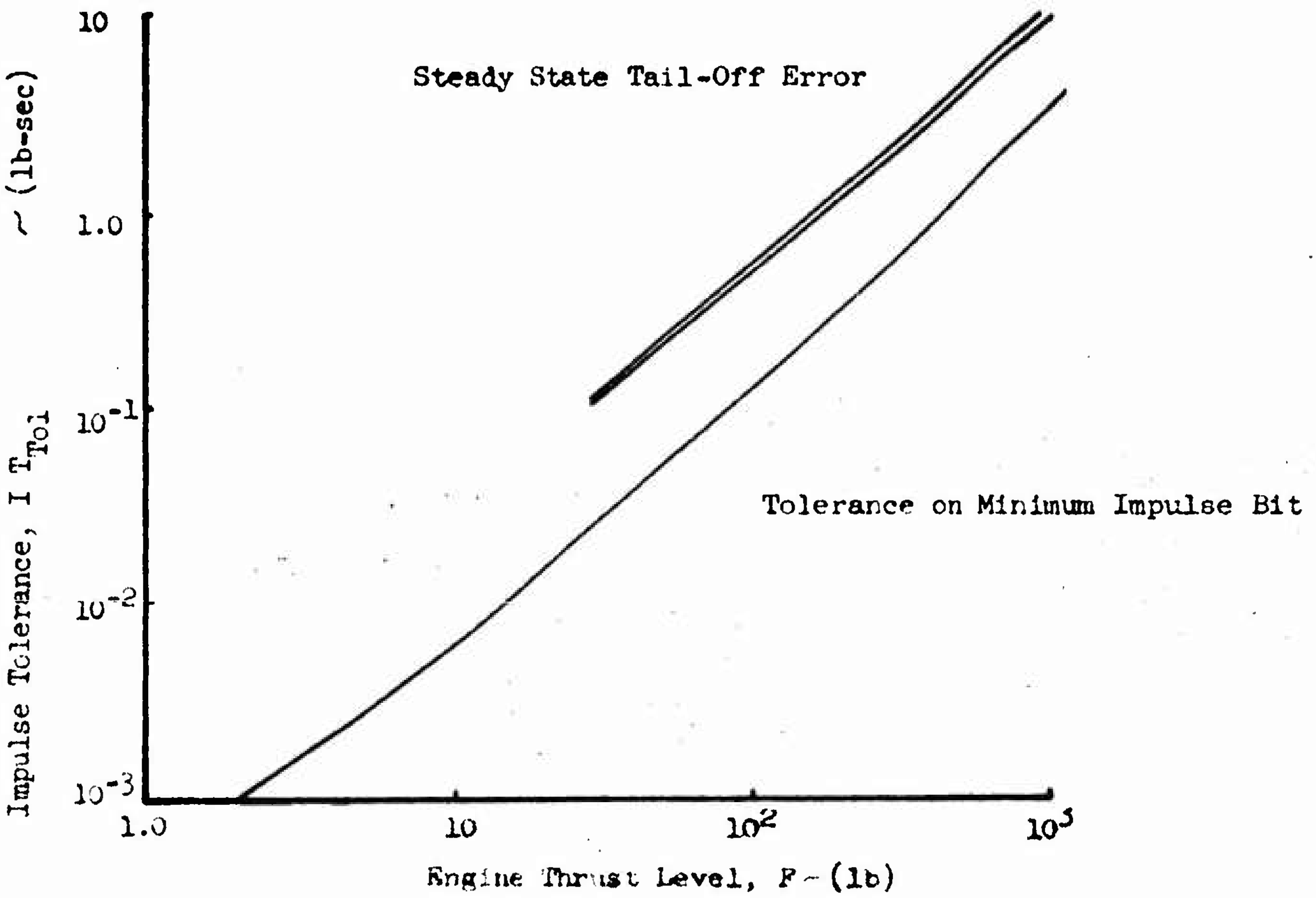
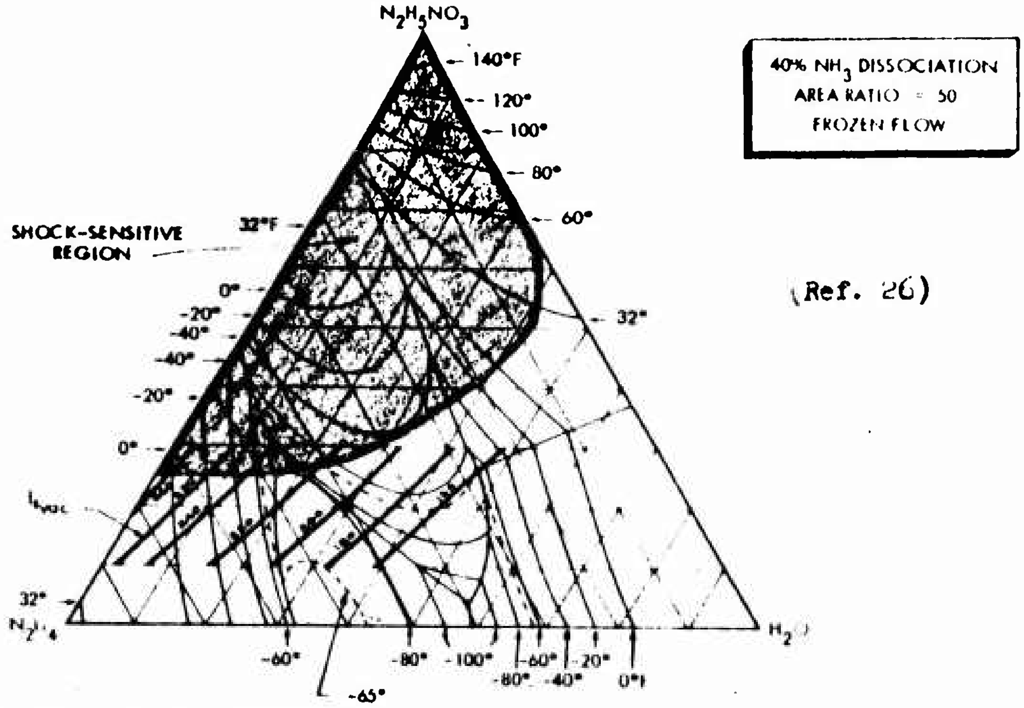


FIGURE 6.3-24 IMPULSE TOLERANCE FOR HYDRAZINE ENGINES



(Ref. 26)

FIGURE 6.3-25 HYDRAZINE-HYDRAZINE NITRATE - WATER TERNARY DIAGRAM SHOWING FREEZING POINT AND VACUUM SPECIFIC IMPULSE

6.3.6 MONOPROPELLANT THRUSTOR DESIGN

The design of monopropellant thrustors using hydrogen peroxide and those using hydrazine is sufficiently different to merit separate discussion. Their design is strongly affected by the particular catalyst used, which is quite different in the two systems. Variations in design, to accommodate more advanced propellants, such as 98% hydrogen peroxide, or the hydrazine-hydrazinium nitrate mixtures, can be treated on a preliminary basis as a simple change to basic design parameters. However, these changes actually involve significant materials changes due to the higher temperatures involved, hence should not be treated as currently operational approaches.

6.3.6.1 THRUSTOR DESIGN - HYDROGEN PEROXIDE ENGINES

Hydrogen peroxide thrustors can be configured, for preliminary design exercises, by using the procedure outlined below. Primarily, this concerns setting performance levels, defining catalyst configuration, and sizing the engine as shown in 6.3.8.1.

1) Assume engine thrust level, F, and duty cycle
2) Assume chamber pressure, P_c, and nozzle expansion ratio (ϵ)
3) Determine specific heat ratio, γ, for the propellant using Figure 6.3-16
4) Determine thrust coefficient, C_F, for the engine using Figures 5.1-1 and 5.2-5.
5) Estimate propellant specific impulse, I_s, by means of Figures 6.3-17 and 6.3-18 considering duty cycle effects
6) Determine propellant flow rate, $\dot{\omega}$, by

$$\dot{\omega} = \frac{F}{I_s} \tag{6.3-2}$$

7) Determine catalyst frontal area, A_c, by

$$A_c = K_c \dot{\omega} \tag{6.3-3}$$

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where:

K_c = a constant relating catalyst frontal area to propellant flow rate. Peroxide thrustors with pre-heat sections use a K_c value of 4 $in^2/(lb/sec)$. If no pre-heat section is used, assume $K_c = 3$.

8) Assume catalyst pack length, L_b;

$$L_b = 2.0 \text{ inches} \qquad (6.3\text{-}4)$$

9) Define catalyst pack pressure drop, ΔP_{c_p}, by use of Figure 6.3-26.

10) Modify thrustor design, or catalyst pressure drop when changing engine operating conditions, such as when operating at different pressure levels or when throttling, by:

$$(\Delta P_{cp})_2 = (\Delta P_{cp})\left[\frac{K_{c_1}}{K_{c_2}}\right]^{1.9}\left[\frac{1.84 \log_e (L_{c_2}) + 49.6}{1.84 \log_e (L_{c_1}) + 49.6}\right]\left[\frac{P_{c_1}}{P_{c_2}}\right]^{0.92} \qquad (6.3\text{-}5)$$

6.3.6.2 THRUSTOR DESIGN-HYDRAZINE THRUSTORS

Monopropellant hydrazine decomposes at relatively low temperatures. Figure 6.3-5 relates hydrazine decomposition temperature and molecular weight to the amount of ammonia dissociation involved. It is generally desirable to minimize ammonia dissociation to maximize performance within the temperature limitations of practical materials. However, minimum residence time requirements for complete hydrazine decomposition also result in approximately 30 percent ammonia dissociation. The decomposition temperature of hydrazine under these conditions allows the use of such materials as Haynes 25 alloy in conjunction with radiation cooled or radiation/heat sink engine designs, and the Shell 405 catalyst.

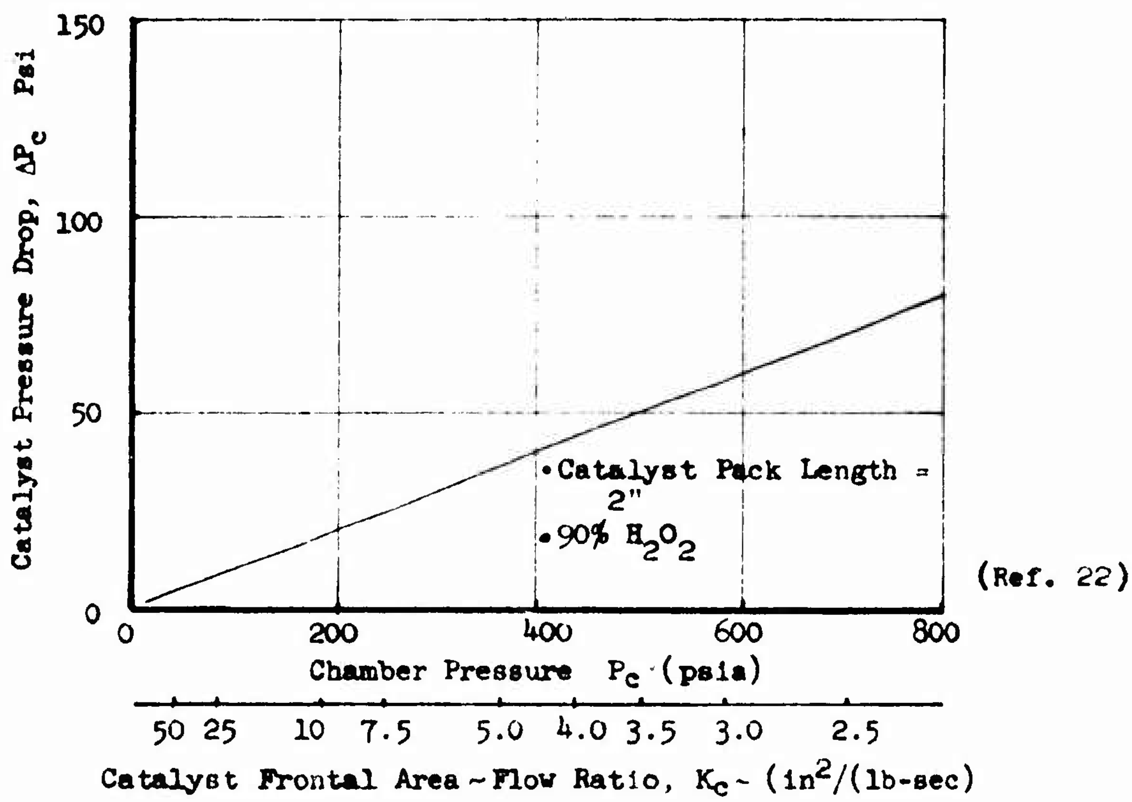

Figure 6.3-26 HYDROGEN PEROXIDE ENGINE CATALYST BED PRESSURE DROP

Hydrazine thrustors of this type can be defined in a preliminary fashion by use of the following iterative procedure (Reference 26).

1) Define engine thrust level, F. In velocity control engines this is done by evaluating limits to maneuver duration, acceleration, gravitational environment, control authority, engine location, and duty cycle. Maximum thrust limits are set by maximum vehicle acceleration limits for structural or control purposes, single pulse minimum maneuver velocity limits and engine system size and weight. Minimum thrust level limits are defined by maximum maneuver time limits involving engine life, performance penalties associated with finite burn-time effects, an. thermal and power limits involved with being in the maneuver position.

   Thrust level selection for reaction control purposes involves defining upper and lower thrust limits associated with disturbance torques, minimum impulse bit, response rate, and engine location for limit cycle operation and for all attitude positioning maneuvers.

2) Assume chamber pressure and nozzle expansion ratio and establish delivered specific impulse using Figures 6.3-21 and 6.3-22.

3) Determine propellant flow rate, $\dot{w}$, by:

$$\dot{w} = F/I_s \qquad \text{(lb/sec)} \qquad (6.3\text{-}6)$$

4) Assume an initial value of reactor bed loading, G (propellant flow per square inch of catalyst cross-sectional area), which can be subsequently iterated. Bed loading values can be selected between:

$$G = 0.03 \text{ to } 0.045 \text{ (lb/sec) per in}^2$$

5) Determine chamber diameter:

$$D_{TC} = \left[\frac{4\,\dot{w}}{\pi\, G}\right]^{\frac{1}{2}} \qquad (6.3\text{-}7)$$

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6) Determine catalyst size at lower end of catalyst bed. It is recommended that different size catalyst pellet particles be used at various levels in the catalyst bed to promote smooth and responsive decomposition. This is the so-called "layered" catalyst consisting of discrete layers of catalyst of different sizes. For the Shell 405 spontaneous catalyst, a 0.2 to 0.3 inch deep layer of 24-30 mesh catalyst located at the upstream end of the bed is usually sufficient. The lower portion of the bed can then be made up of the 1/8 x 1/8 or 1/8 x 1/16 inch particles. The 1/8 x 1/8 size catalyst is currently the largest size available from Shell. Catalyst particle diameter (dp) at the lower end of the bed is:

$$dp \leq D_{Tc}/8 \qquad (6.3\text{-}8)$$

7) Evaluate the major catalyst bed parameters of bed porosity, ϵ_B, and specific surface area, A_s. Figure 6.3-27 shows catalyst bed porosity, ϵ_B, as a function of bed diameter for granular and cylindrical pellets of Shell 405 catalyst. Figure 6.3-28 shows catalyst specific surface area as a function of bed diameter for these same catalyst pellets. These figures were calculated from measured data by the following relationships

$$\epsilon_B = 1 - \left[\frac{\rho_b}{\rho_p}\right] \qquad (6.3\text{-}9)$$

where: ρ_b = catalyst bed density (lb/in^3)

ρ_p = catalyst particle density (lb/in^3)

Catalyst specific surface area, A_s, equals:

$$A_s = \frac{6\left[1 - \epsilon_B\right]}{\phi_s \, d_p} \qquad (6.3\text{-}10)$$

where: ϕ_s = catalyst sphericity

Catalyst sphericity is a measure of exposed surface area on the catalyst

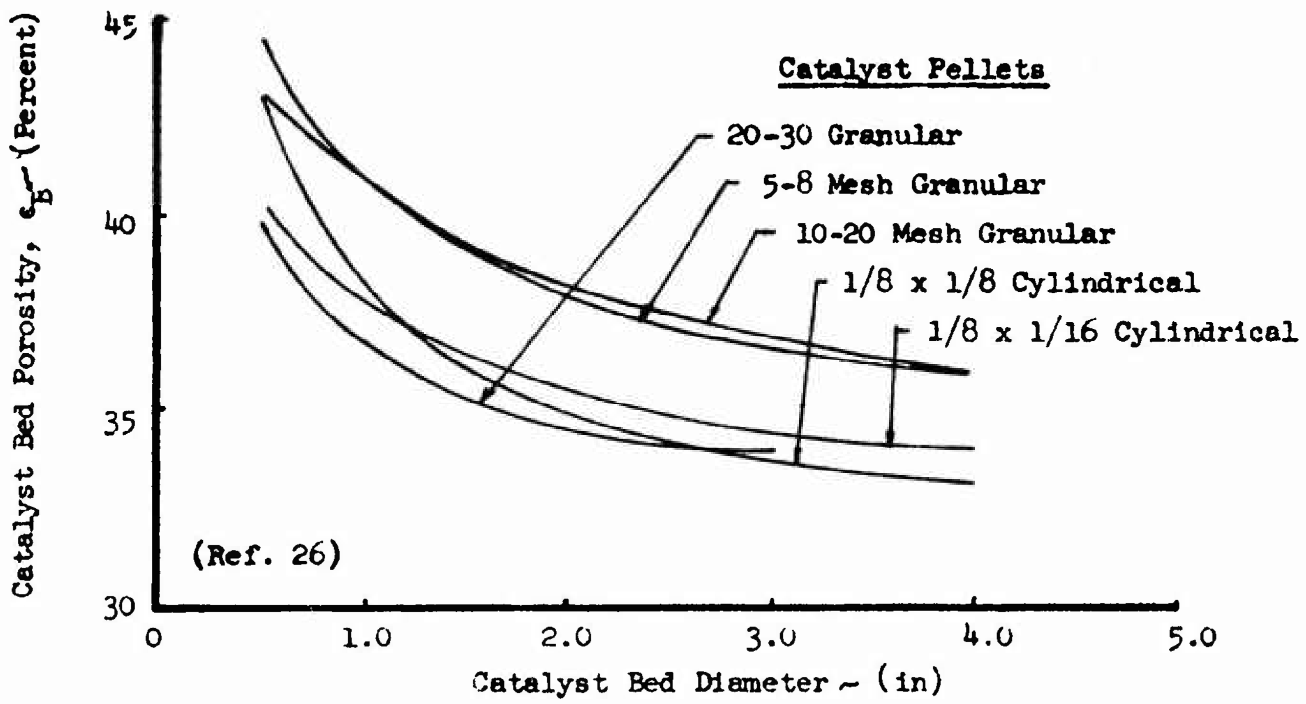


FIGURE 6.3-27 CATALYST BED POROSITY - HYDRAZINE ENGINES

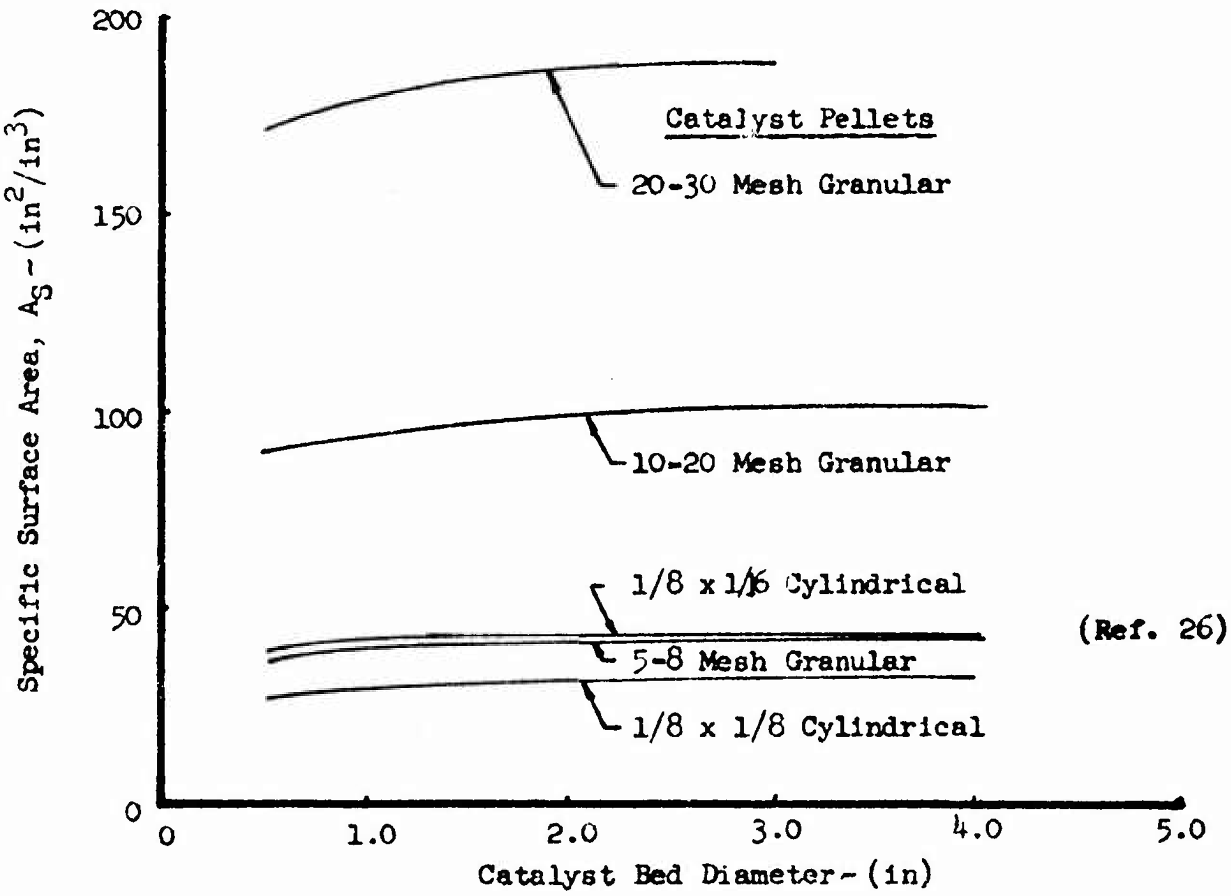


FIGURE 6.3-28 CATALYST BED SURFACE AREA - HYDRAZINE ENGINES

pellet. It is expressed as the ratio of exposed surface area of a sphere to that of the cylinder, where both have equal volume. Thus, for cylindrical pellets:

$$\phi_s = \frac{\left[1.5\, D_c^2 L_c\right]^{2/3}}{0.5\, D_c^2 + D_c L_c} \qquad (6.3\text{-}11)$$

where: $A_o/A_c = \phi_s$

$V_o = V_c$

D_c = cylinder diameter (in)

L_c = cylinder length (in)

A_o = Exposed surface area, sphere (in^2)

A_c = Exposed surface area, cylinder (in^2)

V_o = Volume, sphere (in^3)

V_c = Volume, cylinder (in^3)

8) Determine catalyst bed length, L_B:

$$L_B = 0.2 + \left[145\, G^{0.554} \big/ P_c^{0.306}\, A_s^{0.3}\right] \qquad (6.3\text{-}12)$$

where: P_c = average chamber pressure, psia

The specific surface area, A_s, is for the lower portion of the bed. This equation represents minimum bed length for stable reactor operation, defined as less than $\pm$ 3% P_c oscillation peak-to-peak. This results in NH_3 dissociation of approximately 55%.

9) Determine Reynolds number through each layer of the catalyst bed:

$$R_e = 5.41 \left[G/A_s\right] (10^5) \qquad (6.3\text{-}13)$$

(10) Determine catalyst bed pressure drop for each particle size layer in the catalyst bed. Total bed pressure drop, which is the total for all the layers, is commonly limited to 30-40 psid.

$$\Delta P_{CH} = \frac{1260\, A_s^{1.2}\, G^{1.8}\, L_B}{\epsilon_B\,(1.7\,\bar{P}_c)} \quad \text{where: } (100 < N_r < 600) \tag{6.3-14}$$

or

$$\Delta P_{CH} = \frac{3900\, A_s G^2 L_B}{\epsilon_B (1.7\bar{P}_c)} \quad \text{where: } (600 < N_r < 3000) \tag{6.3-15}$$

$$(\Delta P_{CH})_T = \sum_{n=1}^{n=N} (\Delta P_{CH})_n \tag{6.3-16}$$

11) Iterate bed loading (step 4) until $(\Delta P_c)_T$ = 30 to 40 psid.

Sometimes the reactor is designed as a gas generator to provide gases to remotely located thrusters, to pressurize propellant tanks, or to drive turbines. In these applications, the hydrazine decomposition reaction is usually conducted for greater ammonia dissociation, lowering gas temperatures and exhaust gas molecular weight. In these applications flow rate ($\dot{\omega}$), chamber pressure (P_c) and ammonia dissociation (X) are either specified or easily established. The catalyst bed is then designed by the above procedure, except that catalyst bed length is calculated from

$$L_B = \frac{7.83}{[10^5]}\, G^{0.29} \left[\frac{d_p^{.32}}{\epsilon_B}\right] \left[\frac{RT_c}{\bar{m}}\right] \log_e \left[\frac{0.5}{1-X}\right] \tag{6.3-17}$$

where: dp = particle size in lower portion of catalyst bed

Injector design is a very involved subject which is strongly affected by design preferences of the engine vendors. Generally injector details are of little consequence in preliminary design exercises for spacecraft. Occasionally, however, it becomes important to configure the injector in a preliminary fashion. The following procedure for injector sizing is thus provided (Ref. 26) together with the caution that it be replaced at the earliest convenience by more exact data from the engine vendor:

1) Assuming a showerhead type injector, determine the necessary number of injector orifices, N_T, from catalyst bed dimensions by:

$$N_T = 6A_c \tag{6.3-18}$$

where: A_c = catalyst bed cross sectional area (in^2)

2) Determine injector pressure drop, ΔP_i, by:

$$\Delta P_i = K_1 P_c \tag{6.3-19}$$

where: K_1 = a constant from 0.10 to 0.20

3) Set the injector to catalyst bed spacing at zero, and determine the orifice hole spacing by

$$S_o = \frac{\pi n D_c}{N_T + (\pi n)(n+1)} \tag{6.3-20}$$

where: S_o = spacing between orifice hole centers

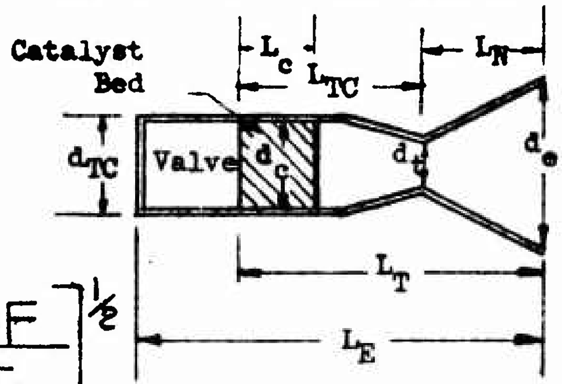
n = number of injector orifice rows

6.3.7 MONOPROPELLANT THRUSTOR GEOMETRY

The basically different design approaches developed in 6.3.6, and extended here to cover thrustor geometry, require individual coverage for hydrogen peroxide thrustors and for hydrazine thrustors.

6.3.7.1 SIZE-HYDROGEN PEROXIDE THRUSTERS

Thrust chamber size may be developed from the procedures described in 6.1.7.1, in conjunction with the following operations referenced to Figure 6.3-27 ;



1) Determine catalyst pack diameter, d_c, by:

$$d_c = \left[\frac{4A_c}{\pi}\right]^{1/2} = \left[\frac{4K_c\dot{\omega}}{\pi}\right]^{1/2} = \left[\frac{4K_cF}{\pi I_s}\right]^{1/2} \qquad (6.3\text{-}21)$$

2) Define thrust chamber exterior diameter, d_{tc}, by:

$$d_{tc} \simeq 1.02\, d_c \qquad (6.3\text{-}22)$$

3) Define nozzle throat diameter, d_t, by

$$d_t = \left[\frac{4F}{\pi P_c C_F}\right]^{1/2} \qquad (6.3\text{-}23)$$

4) Define engine major diameter, at the nozzle exit plane, d_e, by:

$$d_e = d_t\left[\epsilon\right]^{1/2} \qquad (6.3\text{-}24)$$

5) Determine thrust chamber length, L_{TC}, by:

$$L_{TC} \simeq L_c + 0.07\left[F\right]^{1/2} \qquad (6.3\text{-}25)$$

6) Determine nozzle length, L_N, by

$$L_N = \frac{d_t\left[(\epsilon)^{1/2} - 1\right]}{0.536} \qquad (6.3\text{-}26)$$

7) Determine thrustor length, L_T, by

$$L_T = L_{TC} + L_N \qquad (6.3\text{-}27)$$

8) To define total engine assembly length, a value must be assumed for valve size, and displacement from the head end of the thrustor. This factor tends to follow a relatively constant value of 4.0 inches in many installations. Thus, in numerous hydrogen peroxide engine and valve installations, engine assembly length, L_E, equals

$$L_E \simeq 4 + L_T \qquad (6.3\text{-}28)$$

It should be cautioned, however, that substantially smaller arrangements are possible, providing the engine assembly is specifically configured to a size constraint. In dimensionally critical applications, it is suggested that specific valve designs be used in conjunction with equation 6.3-27.

6.3.7.2 SIZE-MONOPROPELLANT HYDRAZINE THRUSTORS

Thrust chamber size may be determined with expressions developed from the design procedures previously described in 6.3.6.2. Thrust chamber exterior diameter, D_{TC}, equals approximately:

$$D_{TC} \simeq \sqrt{\frac{4\dot{\omega}_P}{\pi G}} = \sqrt{\frac{4F}{\pi G I_s}} \qquad (6.3\text{-}29)$$

Steady-state specific impulse ($I_{s_{ss}}$) for the probable range of applications may range from 230-234 sec. and reactor bed loading, G, from 0.03 to 0.045 lb/sec in^2. Thus, this expression can be reduced to:

$$D_{TC} = K_D [F]^{1/2} \tag{6.3-30}$$

where K_D = a factor between 0.35 and 0.43.

At low chamber pressures or small nozzle expansion ratios, a point is reached when chamber diameter exceeds nozzle exit plane diameter. This point occurs in monopropellant hydrazine thrustors when the reactor bed loading, G, equals the propellant flow rate per unit area at the exit plane ($\dot{\omega}p/A_E$). Thus, when $\dot{\omega}_p/A_E$ is larger than G, chamber diameter exceeds nozzle exit plane diameter, and maximum thrustor diameter, D_M equals:

$$D_M = 1.128 \left[\frac{F}{GI_s}\right]^{1/2} \quad \text{when: } \left(\frac{\dot{\omega}}{A_E} > G\right) \tag{6.3-31}$$

When $\dot{\omega}p/A_E$ is smaller than G, nozzle exit diameter predominates and maximum thrustor diameter, D_M, equals:

$$D_M = 1.128 \left[\frac{F\epsilon}{P_c C_F}\right]^{1/2} \quad \text{when: } \left(\frac{\dot{\omega}}{A_E} \leq G\right) \tag{6.3-32}$$

Total thrustor length is comprised of the length of the installed valve assembly, the reactor and plenum, and the nozzle assembly. It is particularly sensitive to reactor length in low thrust engines which, in turn, is a strong function of the particular manufacturer's design approach. Valve dimensions are also important in small engines, and especially so because they come in discrete sizes and are rarely repackaged for particular engines. Total thrustor length can be approximated with the following expression developed from numerous hydrazine thrustor designs:

$$L_E = 1 + \frac{C_1}{5.06}\left[F\right]^{1/2}\left[(\epsilon)^{1/2} - 1\right] \qquad (6.3\text{-}33)$$

where L_E = thrustor total length, inches

C_1 = nozzle factor, equals 1.0 for conical nozzles and 0.80 for 80% bell nozzles.

6.3.8 MONOPROPELLANT THRUSTOR WEIGHT

Hydrogen peroxide and hydrazine thrustors are different enough in design and construction to merit separate discussion relating to weight definition.

6.3.8.1 WEIGHT - HYDROGEN PEROXIDE THRUSTORS

The weight of hydrogen peroxide thrustors is affected by operating chamber pressure, nozzle expansion ratio and particular catalyst bed design. In most installations, existing thrustor designs are modified with basically simple changes such as adding additional catalyst screens, enlarging injector holes or screwing in new nozzle sections. Consequently, the hypothetical minimum weight hydrogen peroxide thrustor rarely is used. In preliminary design exercises, it is desirable to use weight estimates which reflect hardware situations which can be realistically expected. Thus, the following expression, derived from actual hydrogen peroxide thrustor design, is suggested for use in preliminary design situations:

$$W_E = .5 + .025\,F \qquad (6.3\text{-}34)$$

6.3.8.2 WEIGHT-MONOPROPELLANT HYDRAZINE THRUSTORS

The weight of monopropellant hydrazine thrustors is affected by the many design choices available in terms of chamber pressure, expansion ratio, bed loading, chamber geometry, cooling provisions, catalyst, and the particular valve assembly to be used. Weight estimates can be made for preliminary design purposes using the following expression derived from many existing engine designs. Engine assembly weight, W_E, including valve, injector, chamber, catalyst, and nozzle equals approximately:

$$W_E = .3 + .05\,F \qquad (6.3\text{-}35)$$

6.3.9 DEVELOPED MONOPROPELLANT THRUSTORS

Monopropellant thrustors have been developed for use over a thrust range of 0.002 to 1200 pounds. Hydrogen peroxide thrustors extend from 1 to 1200 pounds within this range, though space applications have not exceeded 630 pounds. Table 6.3-3 is a list of monopropellant hydrogen peroxide engines which have been developed. Developed hydrazine engines span a thrust range of 0.002 to 300 pounds though flight applications have so far been limited to 0.02 to 50 pounds. Table 6.3-4 lists monopropellant engines developed for spacecraft applications and for research purposes.

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TABLE 6.3-3

DEVELOPED HYDROGEN PEROXIDE THRUSTORS

THRUST (LBS)	PROGRAM	VENDOR	I_S (SEC)	P_C (PSIA)	ϵ	NOZZLE/CHAMBER ANGLE	DIMENSIONS (INCHES) W/O VALVE WIDTH	DIMENSIONS (INCHES) W/O VALVE LENGTH	DIMENSIONS (INCHES) WITH VALVE WIDTH	DIMENSIONS (INCHES) WITH VALVE LENGTH	WEIGHT (LBS) W/O VALVE	WEIGHT (LBS) WITH VALVE
1.0	PN 874504	Kidde	163		30	0°	1.40	2.60	1.40	6.5		0.52
1.0	ADV.SYNCOM	Bell	150	100	40	0°	0.50	2.50	1.25	6.5	0.23	0.65
1.0	MERCURY-ACS	Bell	156	276	15	0°	2.08	6.62	2.80	8.3	0.21	0.70
1.5	CENTAUR	Bell	155	198	15	90°	1.625	4.00	1.75	6.62	0.34	1.50
2.0	CENTAUR	Bell	155	198	15	0°	1.625	4.00	1.62	6.50	0.32	1.48
2.0	ASSET	Kidde	150		18							1.10
2.0	SATAR	Kidde	150		18	90°	0.75	3.00	1.75	6.775		1.10
2.0	SYNCOM COMSAT	Kidde	156		17.2	0°	0.75	3.855	1.75	8.03		0.58
2.0	SCOUT SLV-1A	Kidde	150		22	90°	2.0	2.5	3.0	5.0		1.17
2.5	MMU	Bell	156	300	40	0°			1.90	7.6	0.25	0.7
2.7	SYNCOM	Kidde	156		17.2	0°						0.58
3.0	CENTAUR	Bell	155	196	15	90°	1.625	4.0	1.75	6.62	0.35	1.51
3.0	SATOR	Kidde	150		17	90°	0.75	3.0	1.75	6.775		1.00
5.0	ADV.SYNCOM	Bell	156	100	40	90°			1.00	8.45	0.57	0.92
5.0	ASSET	Kidde	150	150	11.3	90°	1.44	3.5	2.56	6.375		1.00
5.0	ATS	Kidde	165		40	0°	1.0	5.75	2.50	9.47	0.80	
5.0	ATS	Kidde	165		40	90°	0.625	2.75	1.0	6.875	0.60	
5.0	MMU	Kidde	165		40	0° or 90°	1.0	4.0	2.0	9.37		1.62

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TABLE 6.3-3 (Cont.)

THRUST (LBS)	PROGRAM	VENDOR	I_S (SEC)	P_C (PSIA)	ϵ	NOZZLE/ CHAMBER ANGLE	DIMENSIONS (INCHES) W/O VALVE WIDTH	DIMENSIONS (INCHES) W/O VALVE LENGTH	DIMENSIONS (INCHES) WITH VALVE WIDTH	DIMENSIONS (INCHES) WITH VALVE LENGTH	WEIGHT (LBS) W/O VALVE	WEIGHT (LBS) WITH VALVE
5.0	MMU	Kidde	165		40	0°	0.75	4.09	2.5	8.12		1.12
6.3	MERCURY	Bell	157	270	15	90°	2.4	7.085	2.625	8.7	.56	1.0
14	MERCURY	Bell	156	300	15	90°	3.1	9.2			0.80	1.8
14	SCOUT SLV-1A	Kidde	150		17	90°	3.25	3.5	5.0	6.9		1.98
15	ASSET	Kidde	150		12	90°	2.2	4.0	2.8	8.15		1.75
19	X-20	Bell	156	300	15	0°	2.15	4.25	3.35	7.50	0.59	1.18
22	Burner II	Kidde	163	203	40	90	4.2	3.8	4.2	8.8	1.1	1.55
24	MERCURY	Bell	157	250	15	90°	2.17	7.5	0.17	12.2	0.85	1.93
24	SCOUT SLV-1A	Kidde	142		98							
34	X-20	Bell	156	300	15	90°	3.35	8.35	4.4	8.70	0.90	1.49
35	SATAR	Kidde	150		17.7	90°	3.375	5.4	3.5	6.14		2.8
40	ASSET	Kidde	150		12.6	90°	3.45	4.44	3.45	9.14		2.8
42	X20-ACS	Bell	156	300	15	0°	3.35	9.5	3.35	10.0	0.91	1.5
43	X15-ACS	Bell	159	250	15	90°	2.25	5.5	3.25	7.8	1.68	1.98
44	SCOUT AND BLUE SCOUT	Kidde	150		12.6	0°	2.31	5.5	4.02	8.58	1.49	4.8
48	SCOUT SLV-1A	Kidde	150		12.6	120°	2.56	4.0	4.25	7.5		2.98
50	CENTAUR	Bell	158	154	15	0°	2.12	5.05	3.25	9.36	1.05	3.05
51	BOOST GLIDE RE-ENTRY	Kidde	150	233	15	90°	4.7	4.4	4.7	8.5	1.85	2.8
55	BURNER II SESP	Kidde	163	200	40	90°	5.8	5.5	5.8	9.4	2.3	3.25

TABLE 6.3-3 (Cont.)

THRUST (LBS)	PROGRAM	VENDOR	I_S (SEC)	P_C (PSIA)	ϵ	NOZZLE/ CHAMBER ANGLE	DIMENSIONS (INCHES) W/O VALVE WIDTH	W/O VALVE LENGTH	WITH VALVE WIDTH	WITH VALVE LENGTH	WEIGHT (LBS) W/O VALVE	WITH VALVE
18-90	LLRV	Bell	122	325	3.0	0°	3.21	5.25	4.7		1.91	3.56
113	X-15	Bell	158	292	15	0°	1.9	5.4			1.94	2.3
126	LL SIM.	Kidde	120		3.5	0°	3.60	6.3	7.93	8.625		5.6
520	LLRV	Bell	122	325	3.0	0°	4.675	8.5			7.92	9.57
500	SCOUT, SLV-1A	Kidde	150		9.8	90°	5.5	7.0	8.05	12.0		15.9
600	LITTLE JOE II	Kidde	143		6.9	90°	5.2	6.5	6.54	14.75		16.85
630	LL SIM.	Kidde	120		3.5		6.75	8.85				36
1200	CO SPONS.	Bell	157	250	15	0°	8.38	16.0	8.38	20.8	22.5	

(Refs. 2, 3, 14, 25, 27)

TABLE 6.3-4 DEVELOPED HYDRAZINE THRUSTORS

THRUST (LBS)	PROGRAM	VENDOR	I_s (SEC)	P_c (PSIA)	P_{in} (PSIA)	ϵ $(\frac{A_e}{A_t})$	CATALYST	DIMENSIONS (INCHES) W/O VALVE WIDTH	W/O VALVE LENGTH	WITH VALVE WIDTH	WITH VALVE LENGTH	WEIGHT (LBS) W/O VALVE	WEIGHT (LBS) WITH VALVE	MINIMUM IMPULSE BIT (LB-SEC)	POWER (WATTS)
.05	IR&D	R. Res.	220	90		50	S-405	1.1	1.8	1.1	3.8	0.1*	0.3*		
.08	S/C MOD 35	TRW	100											.0002	
.50	USN-NRL9	R. RES.					S-405								
.50	AF-HPM	R. RES	228	200		100	S-405	2.0	3.0	2.0	5.4	0.37	0.90	0.01	5
1.0	IR&D	MARQUARDT	230	140		50	S-405	1.8	3.2	1.8	5.6	0.5	0.75		
1.0**	IR&D	HAM STD	212	160		40	S-405	1.12	3.28	2.25	5.66	0.2	0.7		28
1.5	H^3MU	R. RES.	130*	72		50	S-405	1.4	3.75			0.75			
3.0	H^3MU	R. RES.	130*	72		50	S-405	2.15	4.9			0.50			
3.5-1.4	S/C MOD 35	TRW	229				S-405	1.0		1.0			0.53	0.35	
3.5-1.2	INTELSAT III	TRW	229	250-80	465-115	50	S-405	1.0	3.1	3.0	6.5	0.1	0.53	0.02	5
5.0	IR&D	R. RES.	233	200		100	S-405	2.0	4.81	1.35	6.75	0.49	0.88	0.1	20
5.0	IR&D	R. RES.	233	80		60	S-405	1.1	5.8	2.1	8.5	0.79	1.24	0.1	20
5.0	LTV	R. RES.	175(S.L.)	200		1.5	S-405	3.25	3.0				0.55		
5.0*	IR&D	HAM STD													
5.5-2.0	IR&D	KIDDE	235-230	400-50		40	S-405	0.90	4.2	1.12	7.3	0.4			14
5.0	IR&D	MARQUARDT	232	155		50	S-405	2.4	6.0	2.4	8.3	1.2	1.45		
5.0	IR&D	MARQUARDT													
8.0-2.5**	IR&D	KIDDE	220-230	200-85											
10	IR&D	R. RES.	233	200		100	S-405	2.0	5.0	1.35	7.0	0.5	0.9		
12.0-2.5	IR&D	KIDDE	232-234	250-50									1.25	0.05	
16-8	IR&D	KIDDE	234-232	250-50			S-405					1.25*		.05	
23	BSD-PBPS	R. RES.	230	150		50	S-405	5.0	5.85	5.0	8.0	2.7	3.56	Class.	28
23	BSD-SAMSO	HAMSTD	Classified				S-405								
25*	MARTIN	HAM STD													
25-16	IR&D	TRW	233-228	125-78	304-152	50	S-405	2.73		4.4	12.0			0.016	
27	TRANSTAGE	R. RES.	228	200		50	S-405	3.4	8.0	3.4	12.3	2.0	4.49	0.28	30
35-5	IR&D	KIDDE	235-230	300-50			S-405	2.0	8.0						
50	NASA-JPL	R. RES.	230	150		50	S-405	3.15	10			2.8*			
50	RANGER/ MARINER	JPL					H-7					2.65			
50**	IR&D	JPL					S-405								
50	MARINER 69	TRW				44	S-405					2.5			
60-12	IR&D	KIDDE	220-211	270-80								2.8		0.60	
75	RPL	R. RES.	Classified			50	S-405	3.3	6.0			2.81		Class	28
75**	IR&D	TRW	230				S-405							0.75	
75*	IR&D	HAM STD		185		60	S-405	4.27	9.84	4.27	17.2	4.7	6.6		40
100	IR&D	HAM STD													
100*	IR&D	HAM STD													
100	IR&D	MARQUARDT	236	165		50	S-405	5.5	11.0	5.5	14.0	6.12	7.21		
150	IR&D	HAM STD		225		60	S-405	5.45	12.0	5.45	19.0	7.3	9.2		40
170	IR&D	R. RES.	236	170		100	S-405	8.5	17.5	8.5	23.0	9.0	13	200 QR	
200*	IR&D	HAM STD	235	250		60	S-405	6.	12.94	6.	20.2		10.7		40
200-12	IR&D	KIDDE	225-217	220-80			S-405	4.5	9.38						
287-16	AF-612	TRW	235	229-13	500-14	50	HA3+S405	5.++	15.8++	8.4	24	19.0		150 QR	200QR
300	IR&D	R. RES	232	300		70	S-405	6.7	15	6.7	19	34.0	39		

LEGEND:
- * $N_2H_4 + H_2O$
- ** $N_2H_4 + N_2H_5(NO_3) + H_2O$
- \+ HEAVY WEIGHT
- ++ ATE-80
- S.L. SEA LEVEL
- QR QUAD REDUNDANT VALVE
- H^3MU HYDRAZINE HAND HELD MANEUVERING UNIT

(Refs. 14, 16, 20, 28)

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6.4 BIPROPELLANT THRUSTORS

6.4.1 GENERAL

Liquid bipropellant engines produce thrust by chemically reacting two propellants, called the oxidizer and the fuel, in a thrust chamber and directing the resulting exhaust through a nozzle. Bipropellant engines are used in the Lunar Orbiter, Surveyor, Agena, Mercury and Apollo spacecraft.

Early bipropellant engines employed hydrogen peroxide, liquid oxygen or nitric acid as "oxidizers" with alcohol, anilene, or kerosene "fuels". These engines were used in missiles, rocket-powered aircraft, and boosters. Satellites have only recently used bipropellant engines to any extent. These applications have occurred in conjunction with the trend to the so-called "earth-storable" propellants. Spacecraft applications strongly contributed to this trend by using the good storability and performance characteristics of these propellants to advantage in the space environment.

Bipropellant engines for spacecraft generally use nitrogen tetroxide (N_2O_4) or mixed oxides of nitrogen (MON) as oxidizers and Aerozine-50 or monomethylhydrazine (MMH) as fuels. Occasionally, inhibited red fuming nitric acid (IRFNA) is used as an oxidizer, and "neat" hydrazine (N_2H_4) or unsymmetrical dimethylhydrazine (UDMH) are used as fuels. MON is a mixture of nitrogen tetroxide with 10-25% nitric oxide. Aerozine-50 is a trade name for a 50-50 mixture, by weight, of hydrazine and UDMH. Chlorine trifluoride (ClF_3) is increasing in use with the fuels mentioned. Beyond these propellants a generation of higher-energy, storable propellants is envisioned which includes the boranes, "Compound A", Hybaline", and similar compounds. However, a real need for these propellants has yet to be established. Consequently, this section will cover the currently popular nitrogen tetroxide, IRFNA, MON, and chlorine trifluoride oxidizers, and hydrazine, UDMH, MMH, and Aerozine-50 fuels.

Combinations of these fuels and oxidizers are hypergolic, that is, they ignite on contact in the thrust chamber and require no auxiliary ignition provisions. This is desirable from the standpoint of system simplicity but it complicates service and handling procedures in that inadvertent contact of fuel to oxidizer must be avoided.

6.4.2 OPERATING PRINCIPLE

Bipropellant rocket engines consist of a propellant control valve, propellant supply lines, injector, thrust chamber, and nozzle assembly. Provisions may also be included to mount the engine to the spacecraft to maintain a predetermined thermal condition, (thermal control) and to control the engine thrust vector direction (TVC).

Propellants are provided to the engine injector on demand by propellant flow control valves in both the fuel and oxidizer engine feed lines. Fuel and oxidizer are kept separate continuously as they pass through the engine propellant manifold into, and through the injector assembly. The injector distributes and mixes fuel and oxidizer within the chamber to promote efficient reaction, to prevent and suppress pressure instabilities, and to assist in engine cooling.

Chemical reaction of these bipropellants produces gases at high temperatures (5500-6000°R) which exceeds the structural capabilities of practical chamber materials. Propellant mixture ratio (ratio of oxidizer to fuel, by weight, at the chamber walls is deliberately controlled by injector design so that substantially cooler, fuel-rich gases contact the walls. Heat rejection from bipropellant engines is controlled by radiation, material ablation, or by regenerative cooling. Almost all early bipropellant engines used regenerative cooling or material ablation to protect the engine during, and immediately following, engine operation. Regeneratively cooled engines are not commonly used where rapid response and pulsing capability are desired because of the large hold-up volume involved. Ablative engines have been used for some time, so the technology has reached a considerable degree of refinement. But, ablative engines are life limited by ablation rate and the amount of ablative material which means that long operating periods require heavier engines. Some radiation cooling has been used in later ablative engine designs to reduce weight. Radiation cooled bipropellant engines have recently been used with success in space. The bipropellant Marquardt MA-109 engine has been successfully flown in the five Lunar Orbiter missions. Radiation cooled engines are fairly lightweight, much less sensitive to life limitations, and can be closely coupled to the propellant valves for fast response and good pulsing performance. Radiation cooled, bipropellant, engine designs currently cover the 5 to 2000 lb. thrust range, and this trend to radiation cooling is expected to continue. Ablative engine designs persist, especially in the larger bipropellant engines which are sometimes augmented with some radiation cooling. The Surveyor spacecraft is the only currently known use for low thrust, regeneratively-cooled, bipropellant engines. Table 6.4-1 shows a comparison of these bipropellant engine types based on those factors which are particularly mission or spacecraft related.

CONSIDERATION	CHAMBER TYPE		
	ABLATIVE	REGENERATIVE	RADIATION
WEIGHT - THRUST CHAMBER		BEST	*
- OVER-ALL SYSTEM	GOOD	**	* GOOD
RESPONSE	GOOD		GOOD
PERFORMANCE (TRAPPED PROPELLANT)	GOOD		GOOD
THROTTLING ABILITY	GOOD	***	GOOD
SHORT DEVELOPMENT TIME	BEST		
COST	BEST		
ENVIRONMENTAL - METEORITE DAMAGE	BEST		
- HARD VACUUM		GOOD	GOOD
- RADIATION		GOOD	GOOD
VEHICLE COMPATIBILITY	GOOD	GOOD	

* WEIGHT COMPARISON DEPENDS ON THRUST LEVEL

** JACKET PRESSURE DROP INCREASES TANK PRESSURE FOR PRESSURE-FED SYSTEMS

*** REASONABLE RANGE OBTAINABLE WITH H_2

TABLE 6.4-1 BIPROPELLANT THRUST CHAMBER COMPARISON

(Courtesy Rocketdyne Div., North American/Rockwell)

6.4.3 BIPROPELLANT THRUSTOR CONSTRUCTION

The three major cooling techniques used in small bipropellant engines involve regenerative cooling, materials ablation or radiation cooling. Ablative engines are particularly life limited by the amount of ablative material present. Ablative and regeneratively cooled thrust chambers have low exterior surface temperatures (normally less than 500°F) which permits them to be recessed within the spacecraft. Radiation cooled engines (surface temperatures above 2000°F) are usually not recessed since they induce heavy thermal loads on adjacent equipment. Sometimes they are even affixed to extended mounts as a thermal standoff to minimize radiation effects. These mounts, plus associated plumbing and wiring can, however, impose significant weight penalties to the system. Recently research and development programs have been conducted to adapt radiation cooled bipropellant chambers to buried installations.

6.4.3.1 ABLATIVE ENGINES

A typical ablative engine is shown in Figure 6.4-1 complete with valves. [illegible]mplified terms, an ablative thrust chamber is constructed of resin-impregnated [illegible]berglass. Cooling is accomplished by boiling out the resin. This ty[illegible] thrust chamber assembly has been employed on the Gemini, Apollo Command Module, and the Titan III Transtage. Ablative thrust chambers are used with radiation cooled nozzle extensions in larger propulsion units (8000-20000 lb-thrust class).

Ablative thrust chambers are usually designed for chamber pressures between 100 to 130 psia since pressure related weight penalties become prohibitive at higher pressure levels. Above a pressure level of 130 psia, a refractory-type throat insert is considered mandatory to minimize thrust variations due to throat area growth. A somewhat common design criteria involves restricting external surf[illegible]

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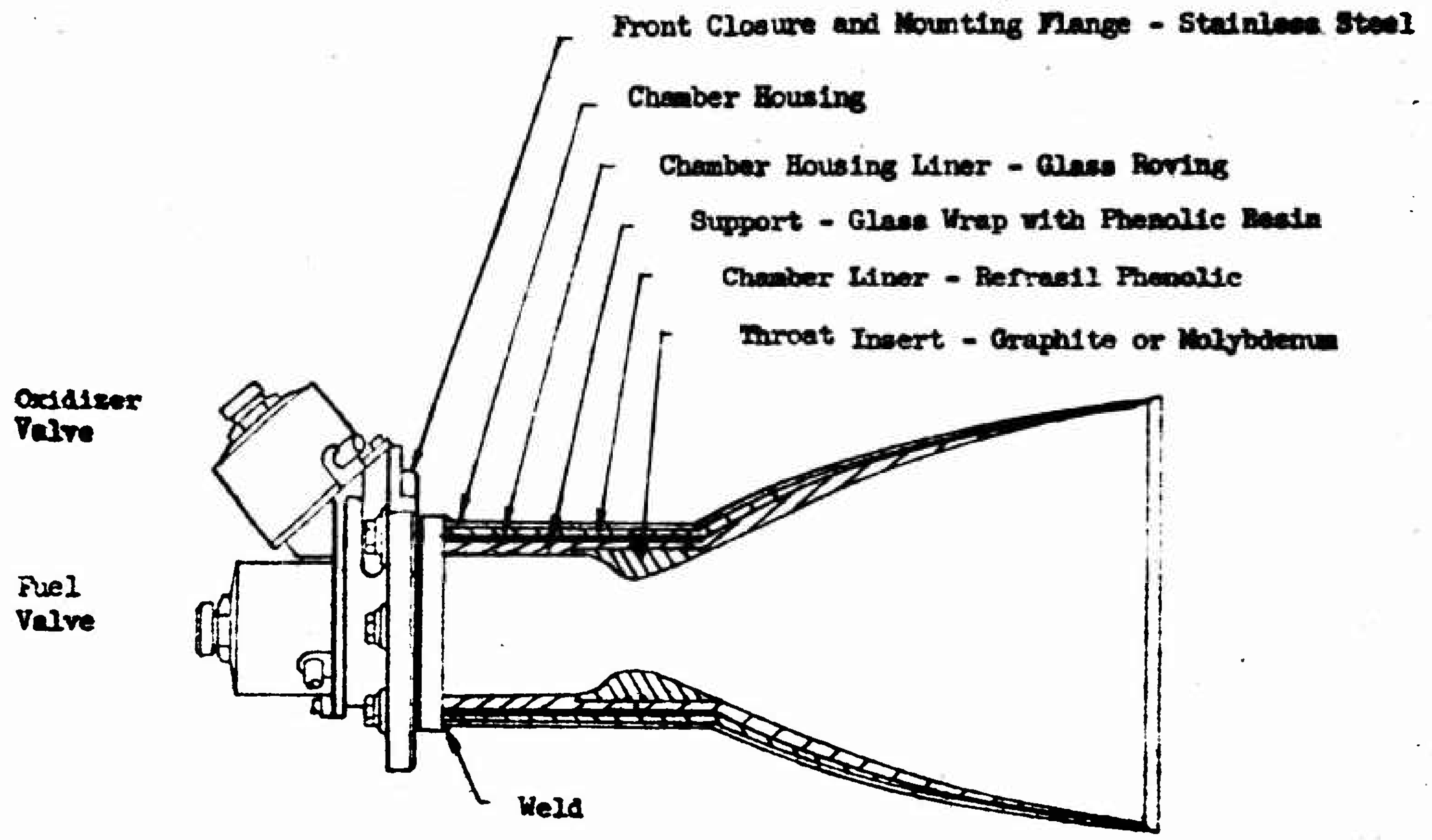

Figure 6.4-1 Ablative Bipropellant Engine

external surface temperature below 400°F to permit a buried installation.

6.4.3.2 RADIATION COOLED ENGINES

A typical radiation cooled bipropellant engine is shown in Figure 6.4-2. . Engines similar to this are used on Lunar Orbiter, Apollo Service and Lunar modules, and on the Agena secondary propulsion system. The high combustion temperatures of bipropellant engines require special materials including the refractory metals. Currently, the most frequent applications involve molybdenum-titanium or tantalum-tungsten alloys. Government funded research and development efforts are currently being conducted to apply columbium alloy and beryllium metals to thrust chamber construction.

Radiation cooled thrust chambers are coated with materials (such as an aluminide or silicide) to prevent oxidation of the base metal. External surfaces are also coated to improve emissivity, and to prevent oxidation during test. Currently, these coatings limit chamber wall temperatures to about 3000-3200°F, which in turn limits maximum chamber pressure to about 100 psia. Current operational throat temperatures are on the order of 2000-2500°F, a value achieved by operating the engine slightly off optimum mixture ratio and/or the incorporation of film cooling. A considerable margin is generally maintained between actual and allowable wall temperature to provide long engine life.

6.4.3.3 REGENERATIVELY COOLED ENGINES

A typical regeneratively cooled, small bipropellant engine is shown in Figure 6.4-3. . At this time, the only known application of this principle is on the Surveyor spacecraft vernier engines. This spacecraft uses three radially distributed engines for midcourse correction, attitude control retro maneuvers, and terminal maneuver vernier control. Attitude control is accomplished by differential throttling.

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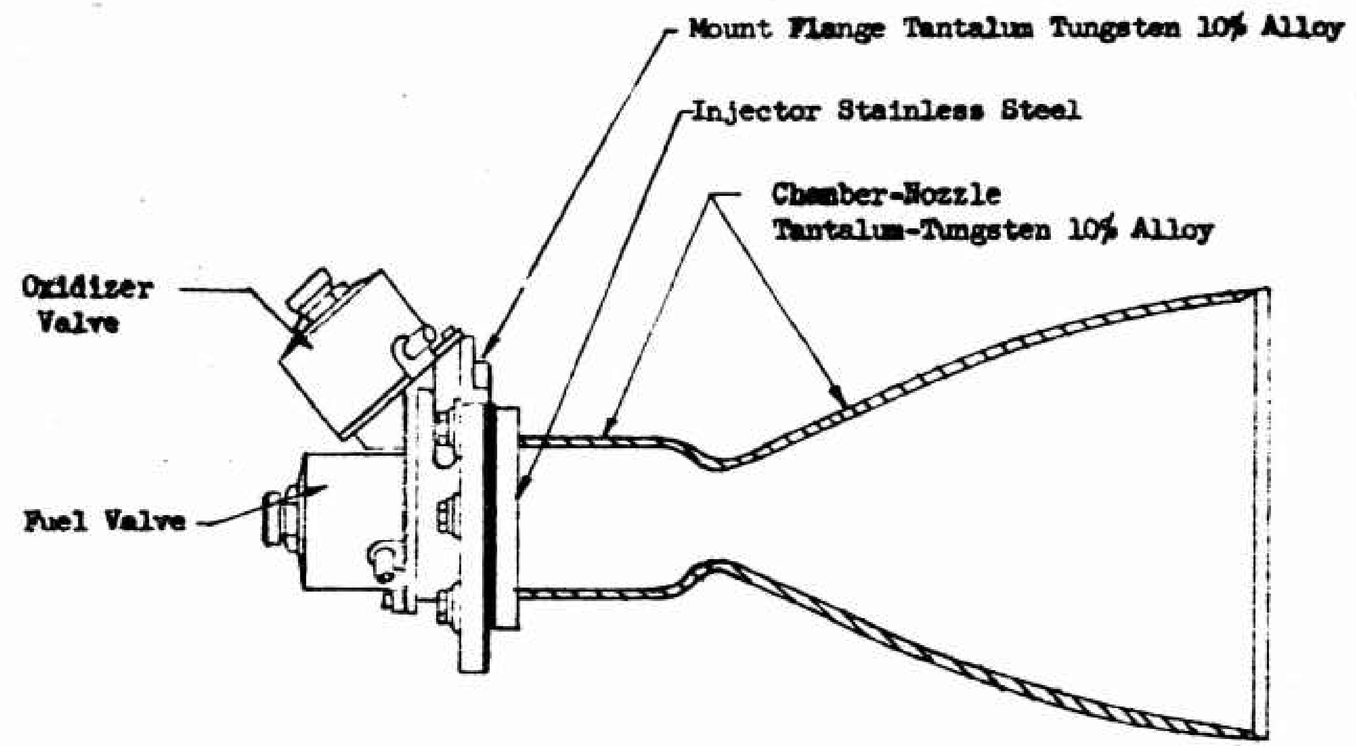

Figure 6.4-2 RADIATION COOLED BIPROPELLANT ENGINE

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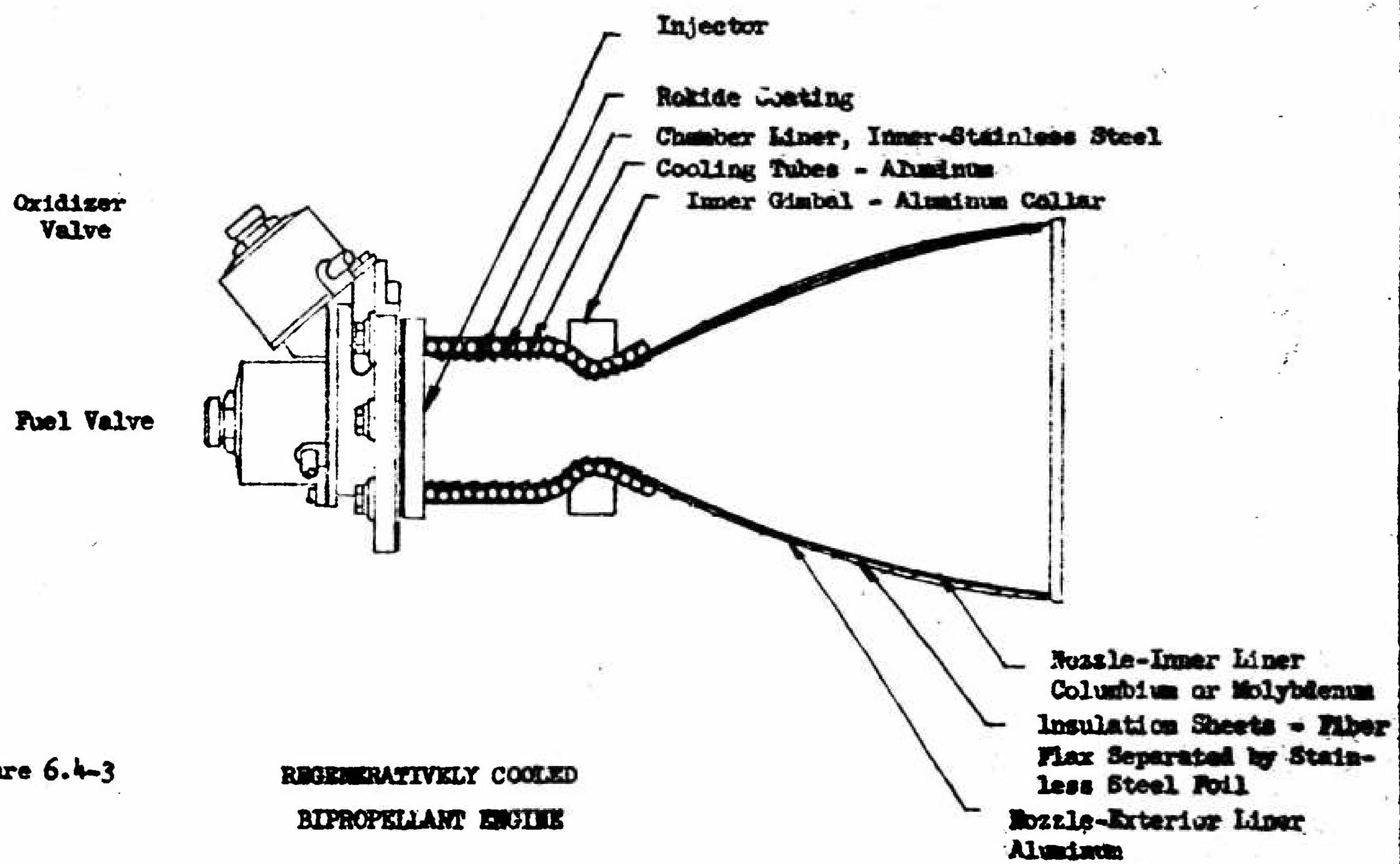

Figure 6.4-3

REGENERATIVELY COOLED
BIPROPELLANT ENGINE

The regenerative cooling circuit contributes a relatively large trapped propellant volume below the engine control valve. This prolongs engine operation during any valve actuation until the cavity is cleared. Thus, transient response and minimum operating times are inferior to those possible in the more closely coupled ablative or radiation cooled engines. It is possible to design regenerative engines specifically for transient performance by locating the propellant flow control valve between the jacket and the injector. However, post-firing heat soakback becomes a significant problem in this case if it can cause the propellants to boil. This is not desirable in that it may overpressurize propellant plumbing briefly, and it may result in entrained gas problems during subsequent operation.

6.4.4 BIPROPELLANTS

Bipropellants in extensive space usage currently involve the so-called "earth storable" propellants. Common oxidizers are nitrogen tetroxide and the mixed oxides of nitrogen (MON), nitric acid. Chlorine trifluoride, already in Navy shipboard use, may also be used in space application. Common fuels are hydrazine, unsymmetrical dimethylhydrazine, Aerozine-50, and monomethylhydrazine. Significant physical properties of these propellants are summarized in Table 6.4-2. They will be discussed separately in greater detail.

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TABLE 6.4-2

PHYSICAL PROPERTIES OF PROPELLANTS

Oxidizers	Freeze Point, °F	Boil* Point, °F	Specific+ Gravity	Absolute+ Viscosity, lb/ft-sec	Specific+ Heat, Btu/lb-°F
N_2O_4	11.8	70	1.49	2.95×10^{-4}	0.365
MON-10	-10	45	1.46	2.04×10^{-4}	
MON-15	-24	35	1.41		
MON-25	-61	17.5	1.39		
IRFNA	-65	142	1.59	9.50×10^{-4}	0.418
ClF_3	-118	53	1.83	3.10×10^{-4}	0.308
Fuels					
N_2H_4	34	236	1.00	6.90×10^{-4}	0.734
UDMH	-71	146	0.785	4.10×10^{-4}	0.647
MMH	-62.3	189	0.871	6.70×10^{-4}	0.699
Aero-50	18.8	158	0.908	7.10×10^{-4}	0.689

* Boiling temperature, referenced to 14.7 psi

\+ Specific gravity, viscosity, and specific heat referenced to 60°F

(Ref. 4)

6.4.4.1 OXIDIZERS-NITROGEN TETROXIDE AND THE MIXED OXIDES OF NITROGEN

Nitrogen tetroxide (N_2O_4) consists principally of the tetroxide in equilibrium with a small amount of nitrogen dioxide (NO_2). As obtained commercially, it contains less than 0.1% water by weight. Nitrogen tetroxide is very reactive and toxic, but is not sensitive to mechanical shock, heat, or detonation. Specific heat, absolute viscosity, density and vapor pressure of nitrogen tetroxide are shown as functions of temperature in Figure 6.4-4.

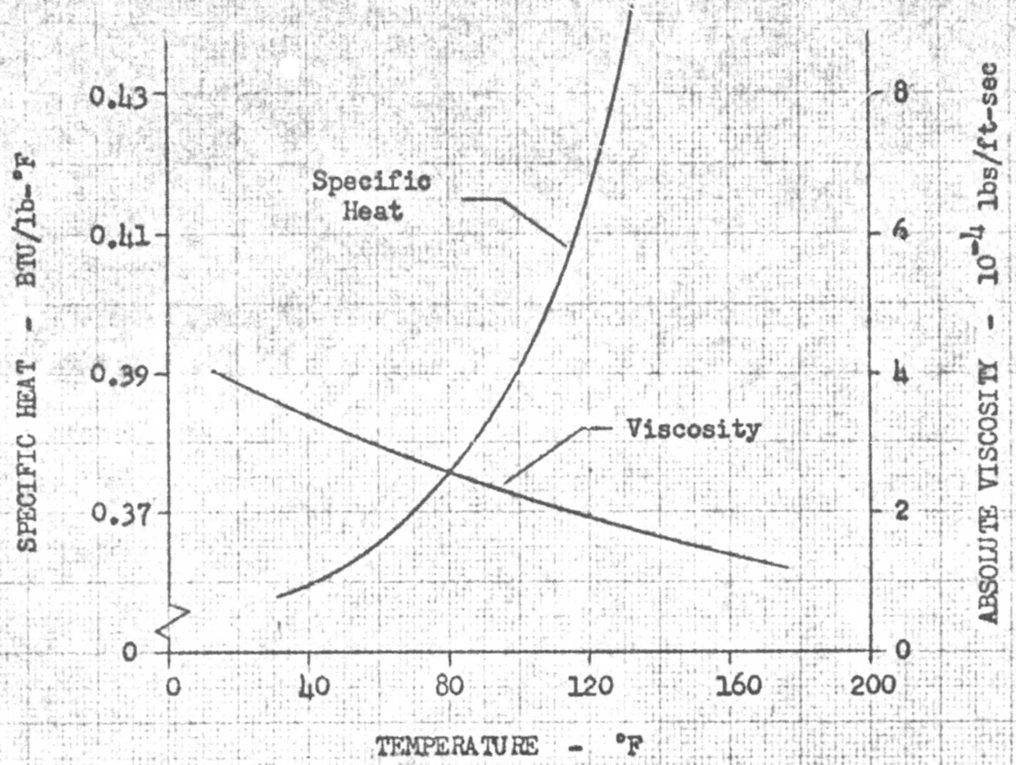
The relatively high freezing temperature of nitrogen tetroxide may be reduced by forming a solution with nitric oxide (NO), producing the so-called "mixed oxides of nitrogen" or MON. These solutions are generally designated by the percentage of NO, hence MON 10, MON 15, and MON 20. Boiling point is also changed, as indicated in Table 6.4-2. The density, absolute viscosity and vapor pressure of MON 10 solutions are shown in Figure 6.4-5.

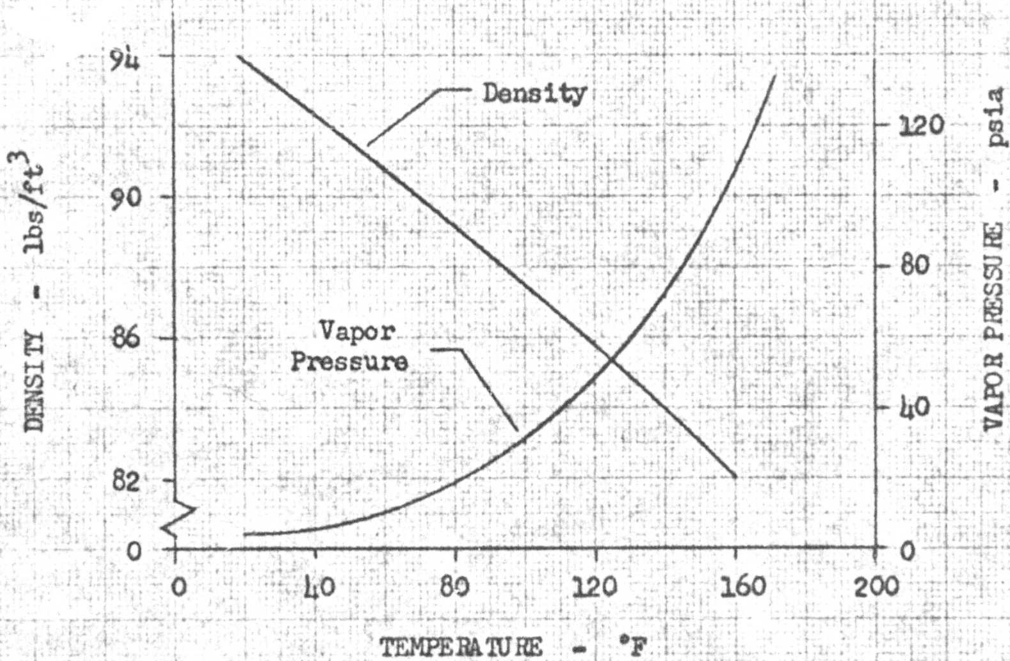
6.4.4.2 Oxidizer-Nitric Acid

Inhibited red fuming nitric acid (IRFNA) consists of 83.4% nitric acid (HNO_3), 13% nitrogen tetroxide (N_2O_4), 3% water and 0.6% hydrogen fluoride (HF). It is highly corrosive, toxic, and reacts with most metals and organic materials. It is currently being employed in several launch vehicle stages, such as Agena, and in several air-launched missiles. Figure 6.4-6 shows specific heat, absolute viscosity, density, and vapor pressure of IRFNA as a function of temperature.

6.4.4.3 Oxidizers-Chlorine Trifluoride

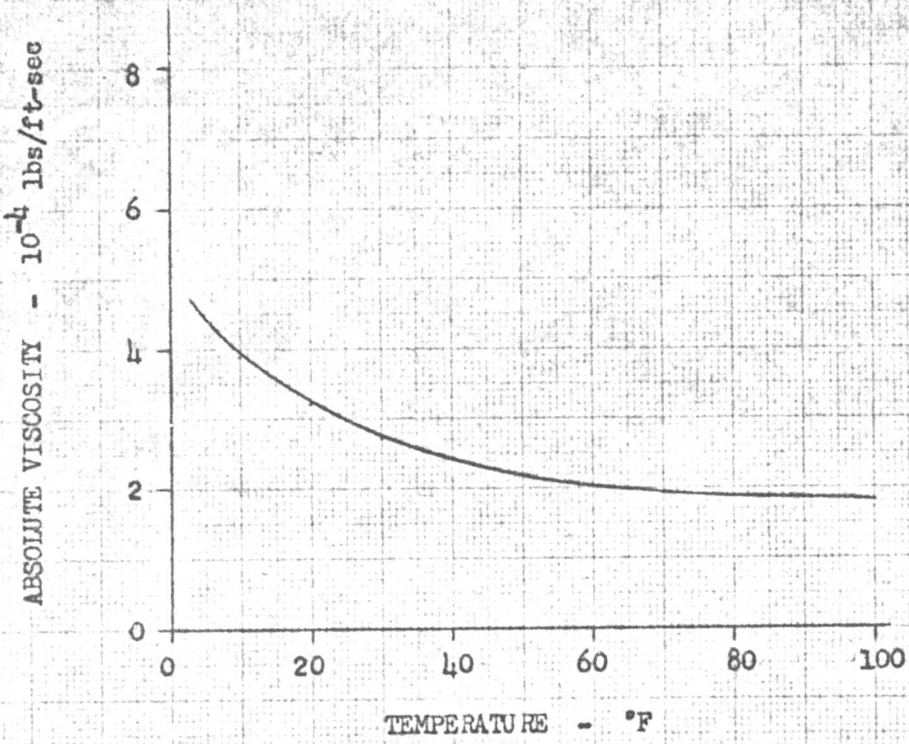
Chlorine trifluoride (ClF_3) is commercially available with purities greater than 99%. It is a toxic and corrosive oxidizing agent similar to fluorine. It reacts with water and will support combustion with almost all organic vapors and liquids. It reacts with all elements except the rare gases and nitrogen. However, it forms a protective film on certain metal surfaces which

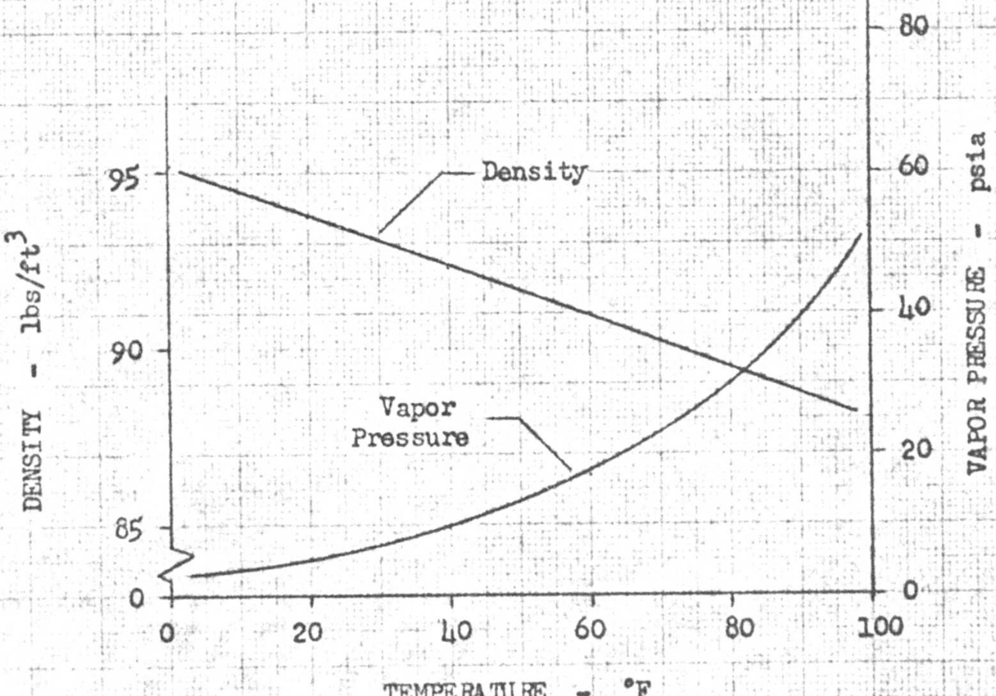




(Ref. 4)

CALC			REVISED	DATE	PHYSICAL PROPERTIES NITROGEN TETROXIDE	FIGURE 6.4-4
CHECK						
APR						
APR					THE BOEING COMPANY	PAGE 141





(Ref. 4)

CALC			REVISED	DATE	PHYSICAL PROPERTIES MON-10	FIGURE 6.4-5
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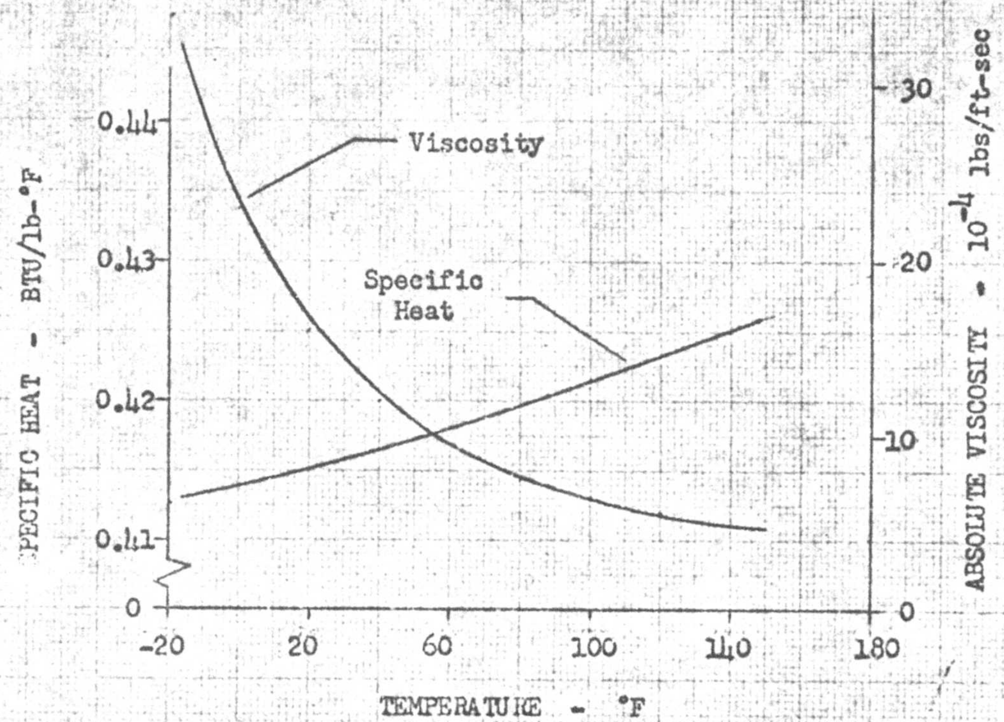

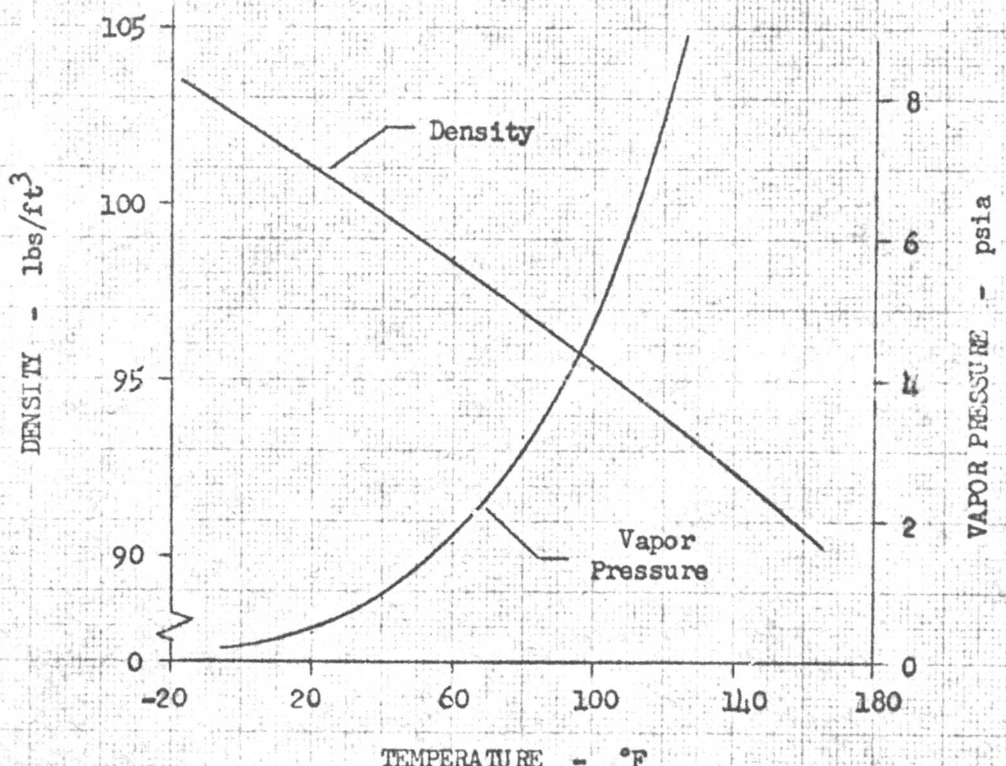

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CALC			REVISED	DATE		FIGURE 6.4-6
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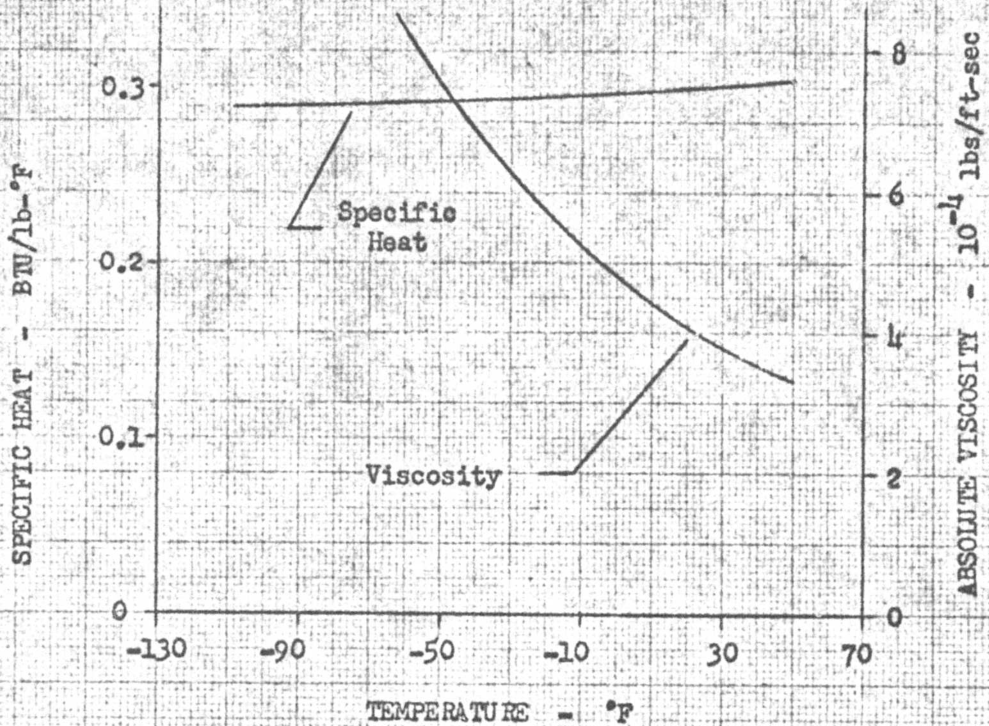
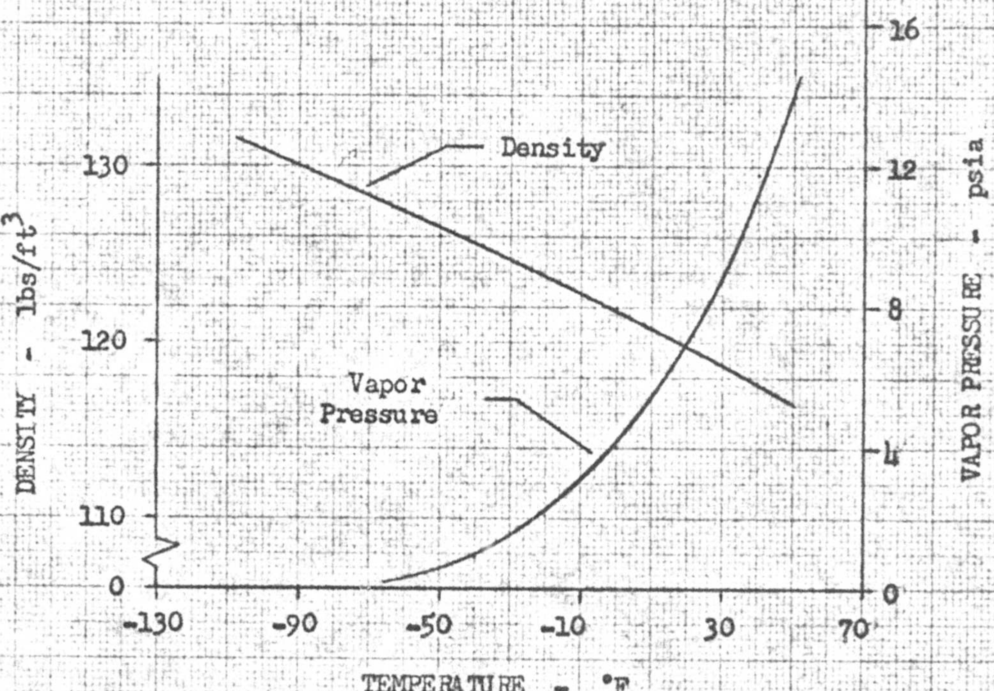
					THE BOEING COMPANY	PAGE 143





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CALC			REVISED	DATE	PHYSICAL PROPERTIES CHLORINE TRIFLUORIDE	FIGURE 6.4-7
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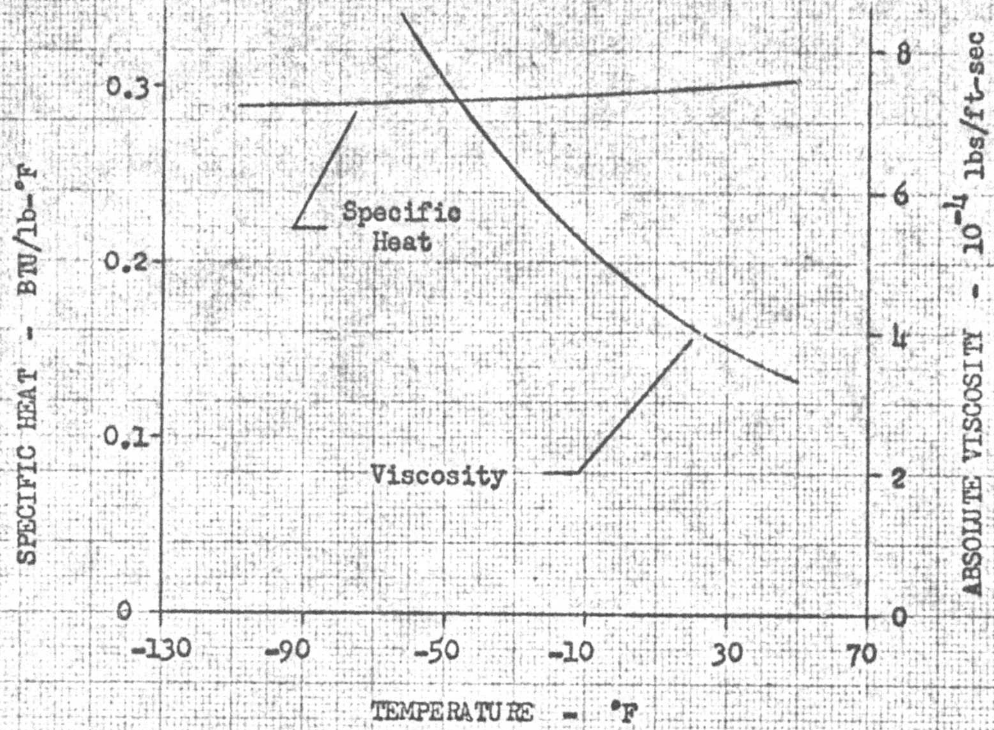
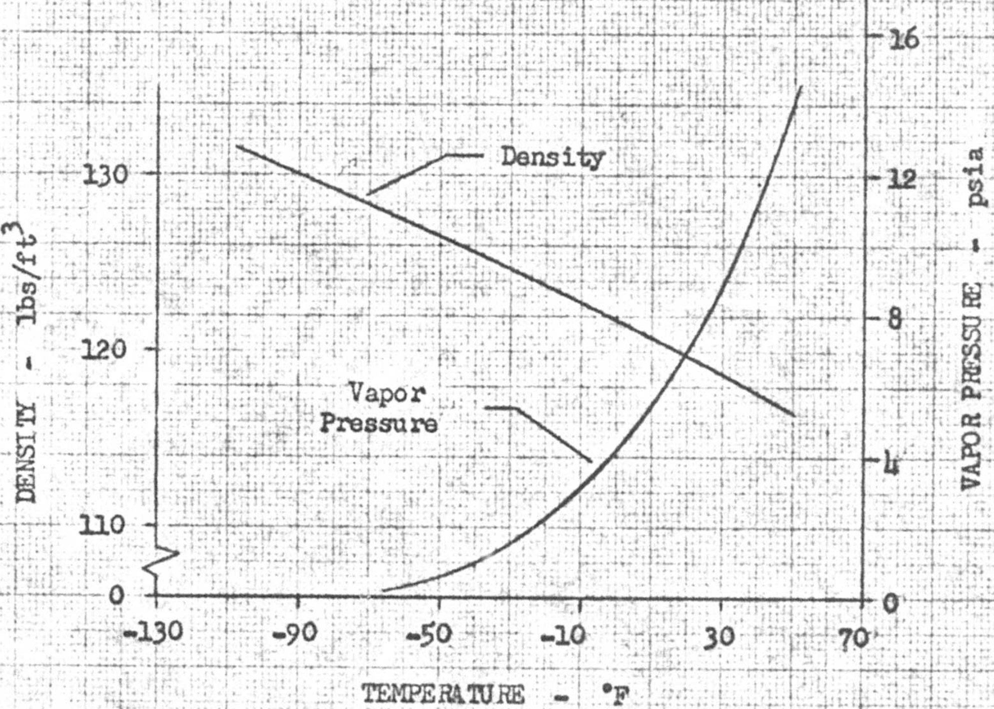
APR					THE BOEING COMPANY	PAGE 145





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CALC			REVISED	DATE	PHYSICAL PROPERTIES CHLORINE TRIFLUORIDE	FIGURE 6.4-7
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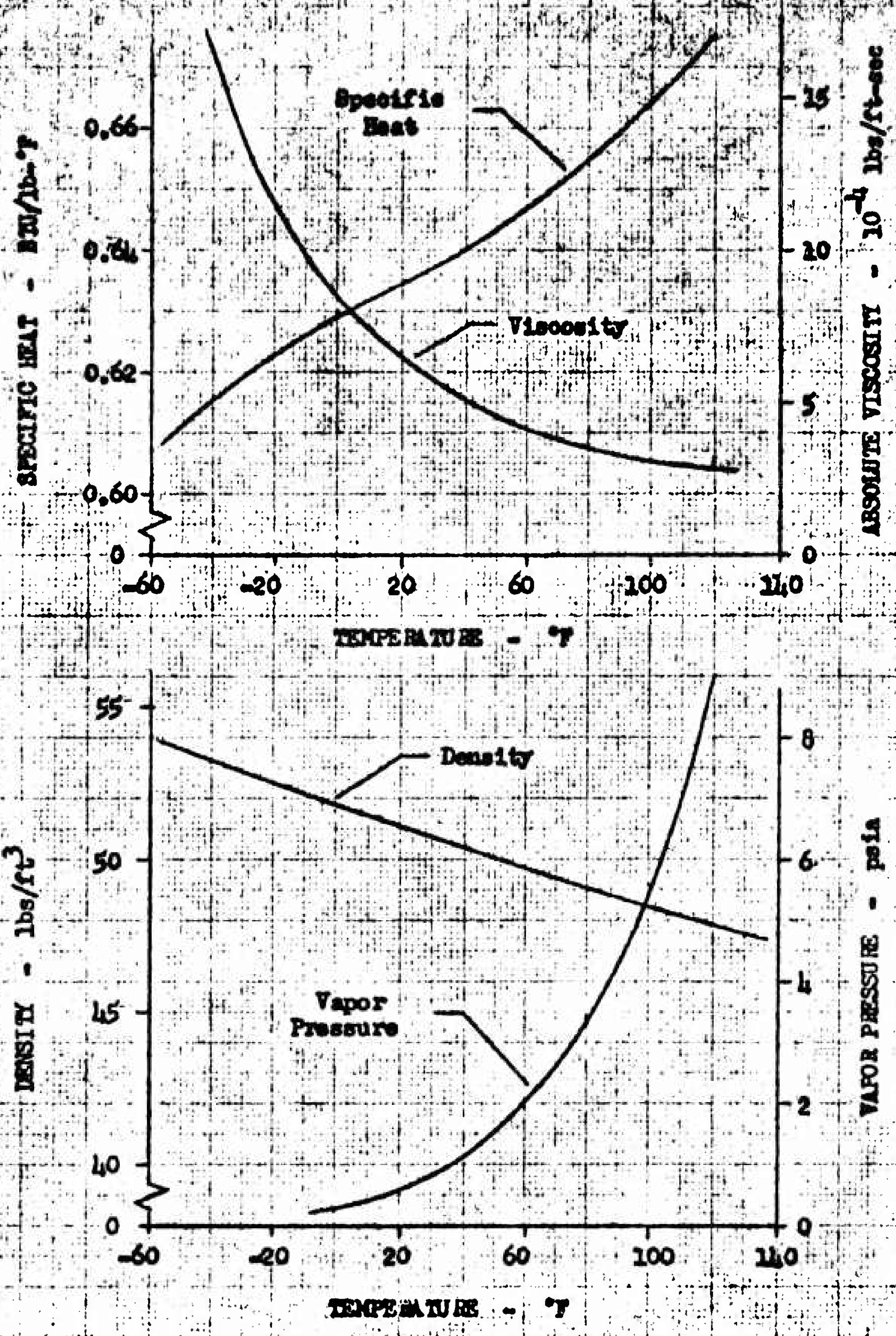

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CALC			REVISED	DATE	PHYSICAL PROPERTIES UNSYMMETRICAL DIMETHYL HYDRAZINE	FIGURE 6.4-8
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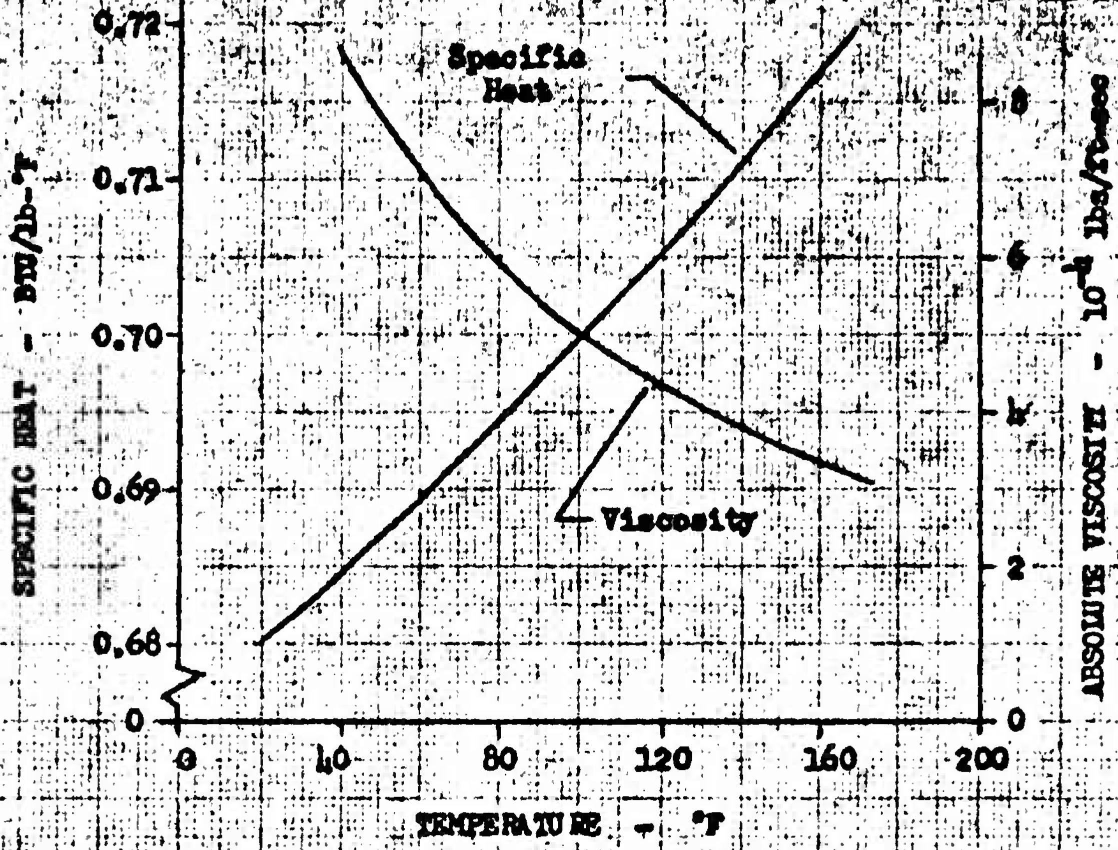

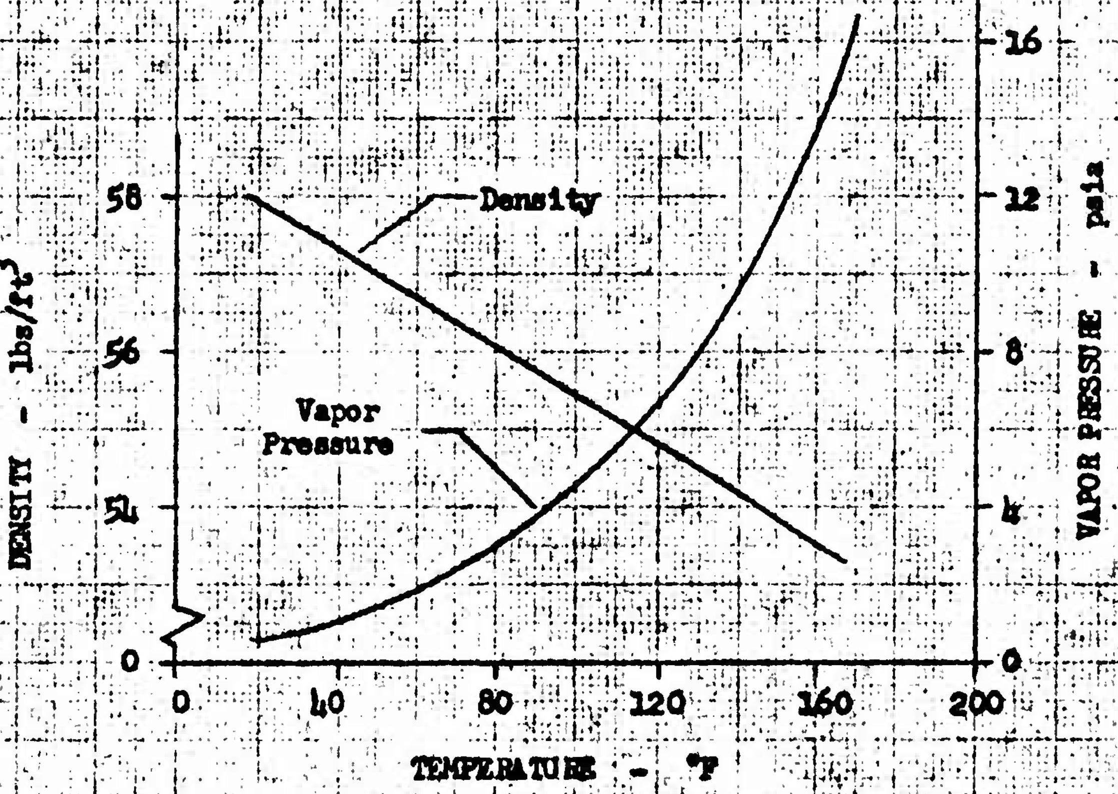

(Ref. 4)

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APR					THE BOEING COMPANY	PAGE 147

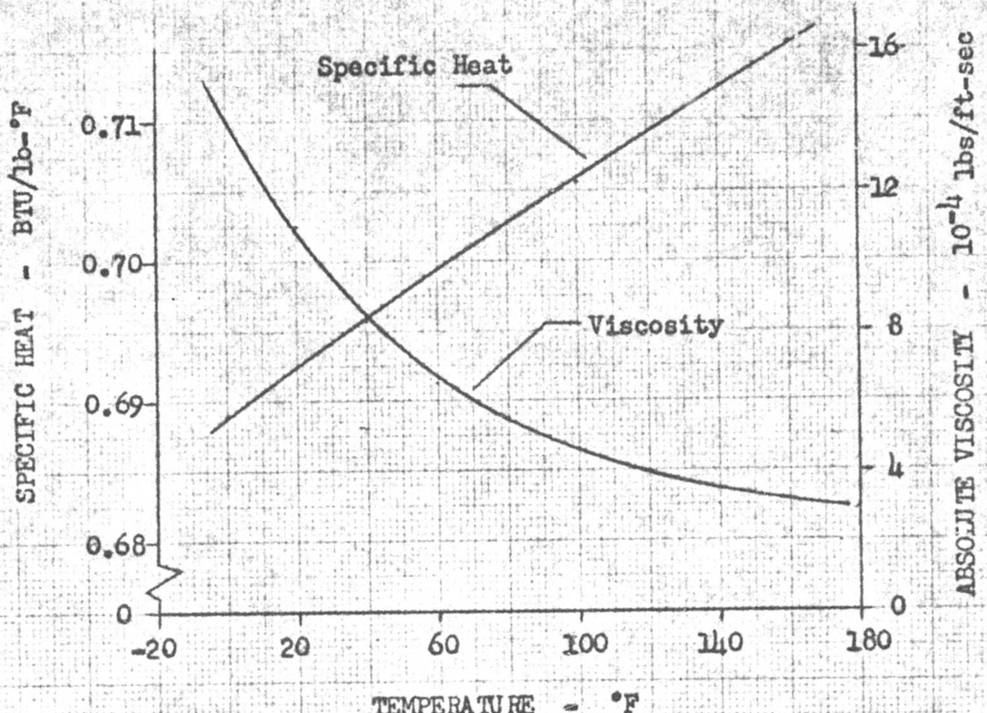
6.4.4.7 FUELS-MONOMETHYLHYDRAZINE (MMH)

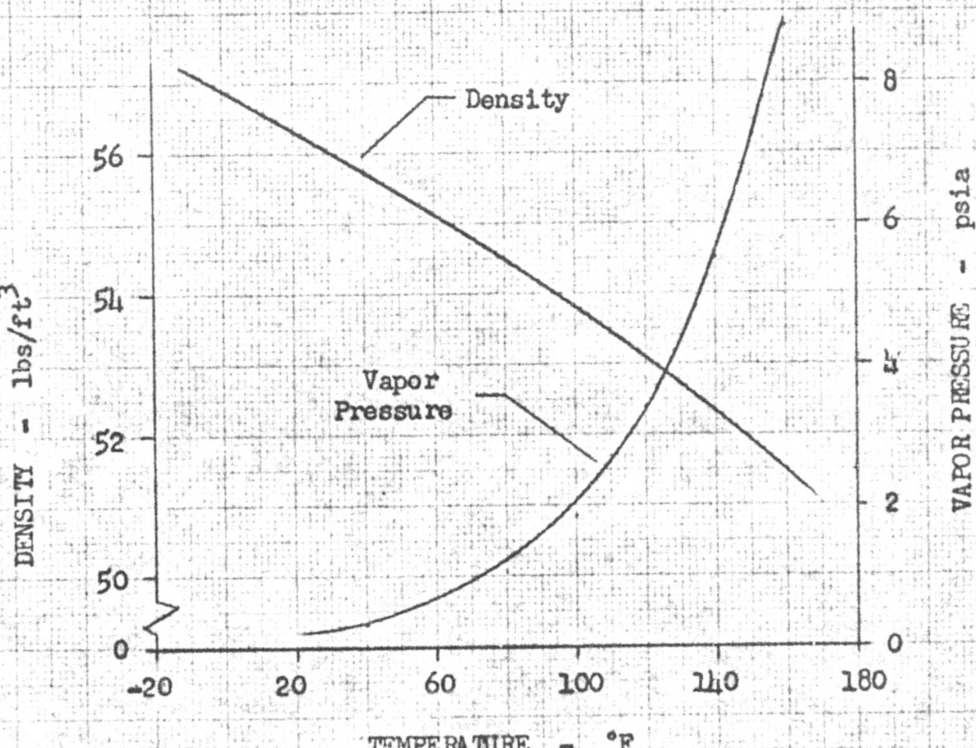
As with other hydrazine-type fuels, MMH is toxic and volatile. It is not sensitive to impact or friction. It is more stable than hydrazine, but is similar to hydrazine in sensitivity to catalytic decomposiiton. The viscosity, specific heat, density and vapor pressure of MMH is shown in Figure 6.4-10 as a function of temperature.

6.4.5 PERFORMANCE OF BIPROPELLANT ENGINES

Figure 6.4-11 shows the theoretical performance of bipropellant combinations considered in this section as a function of propellant weight mixture ratio, and referenced to a 100 psia chamber pressure and 50:1 nozzle expansion ratio. Propellant density is sufficiently different for most combinations to affect spacecraft size, so Figure 6.4-12 shows overall propellant bulk density as a function of mixture ratio for these same combinations.

The performance actually delivered by these combinations is reduced from theoretical values by losses attributable to the mixing and combustion process, thermal and cooling conditions, friction drag, variations in propellants and the geometrical influences of the injector, chamber and nozzle. These factors vary withengine design size and operating duty cycle. The latter two factors are the most significant. Fig. 6.4-13 shows performance as a function of thrust for many different bipropellant engines having different propellants, mixture ratios, chamber pressures and expansion ratios. This performance is actually normalized to a percentage of theoretical vacuum specific impulse (shifting equilibrium) at the engines nominal operating point. The





(Ref. 4)

			REVISED	DATE		
CALC					PHYSICAL PROPERTIES MONOMETHYL HYDRAZINE	FIGURE 6.4-10
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APR					THE BOEING COMPANY	PAGE 149

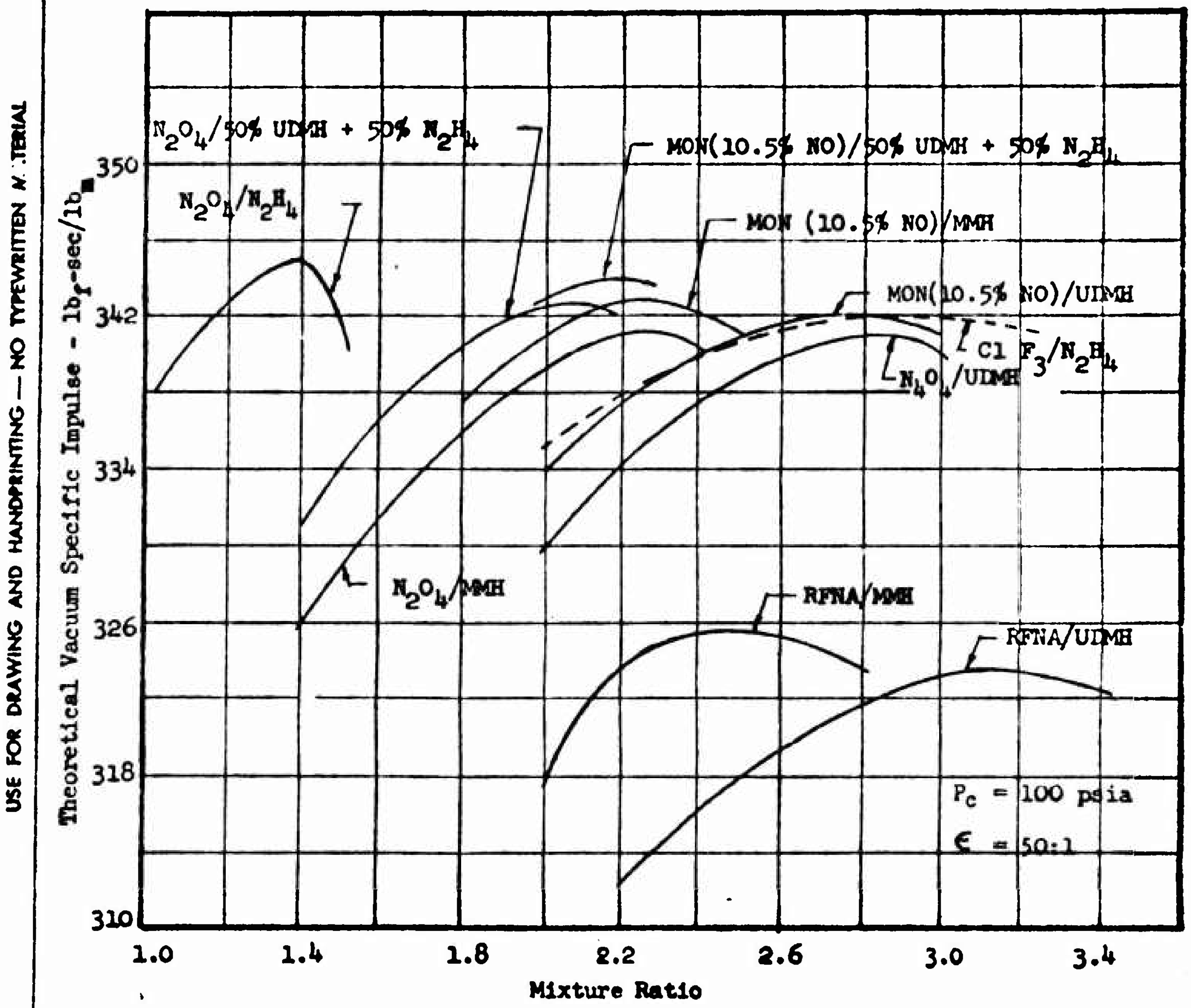

FIGURE 6.4-11 THEORETICAL PERFORMANCE OF BIPROPELLANT COMBINATIONS (Ref. 3)

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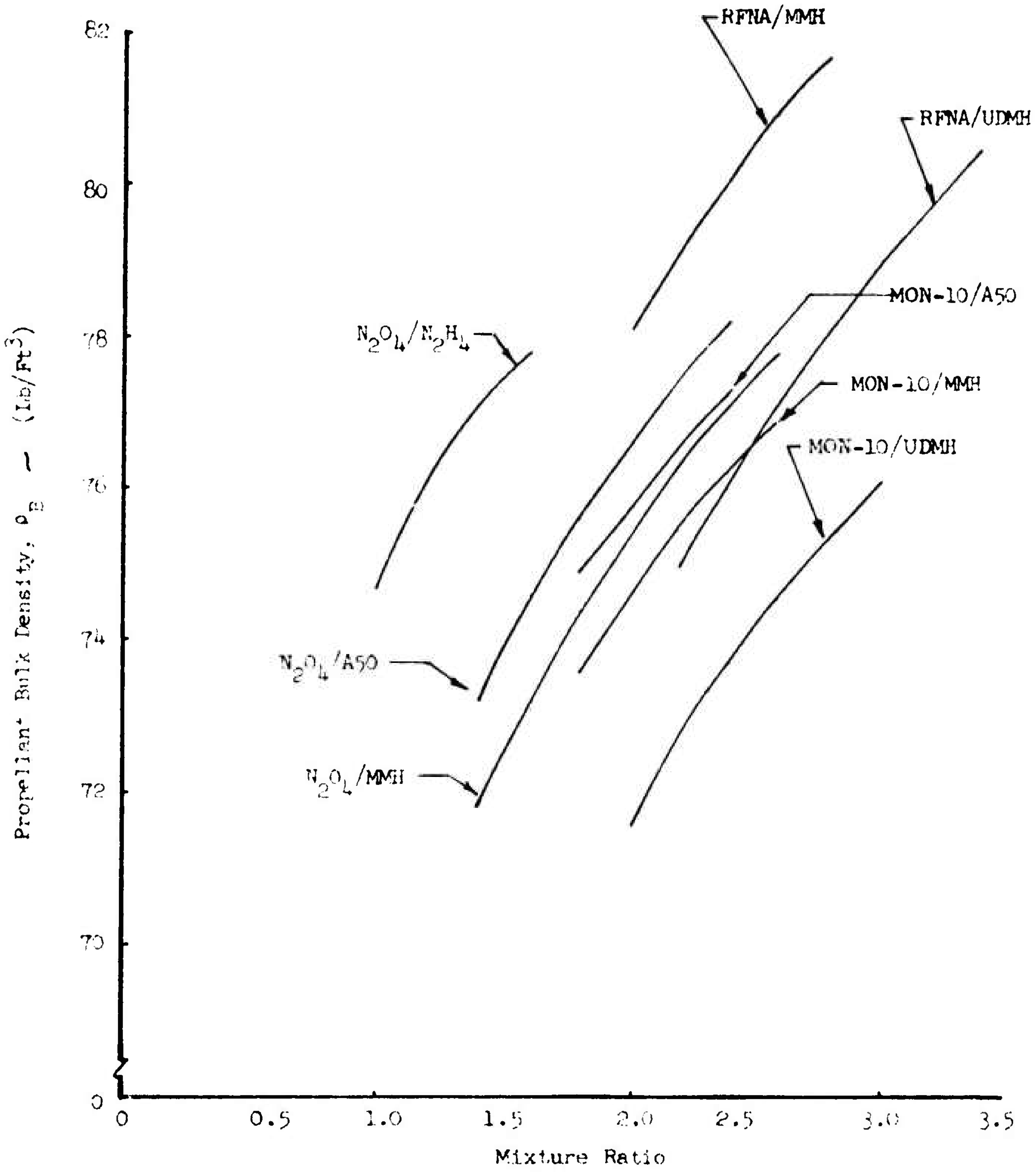
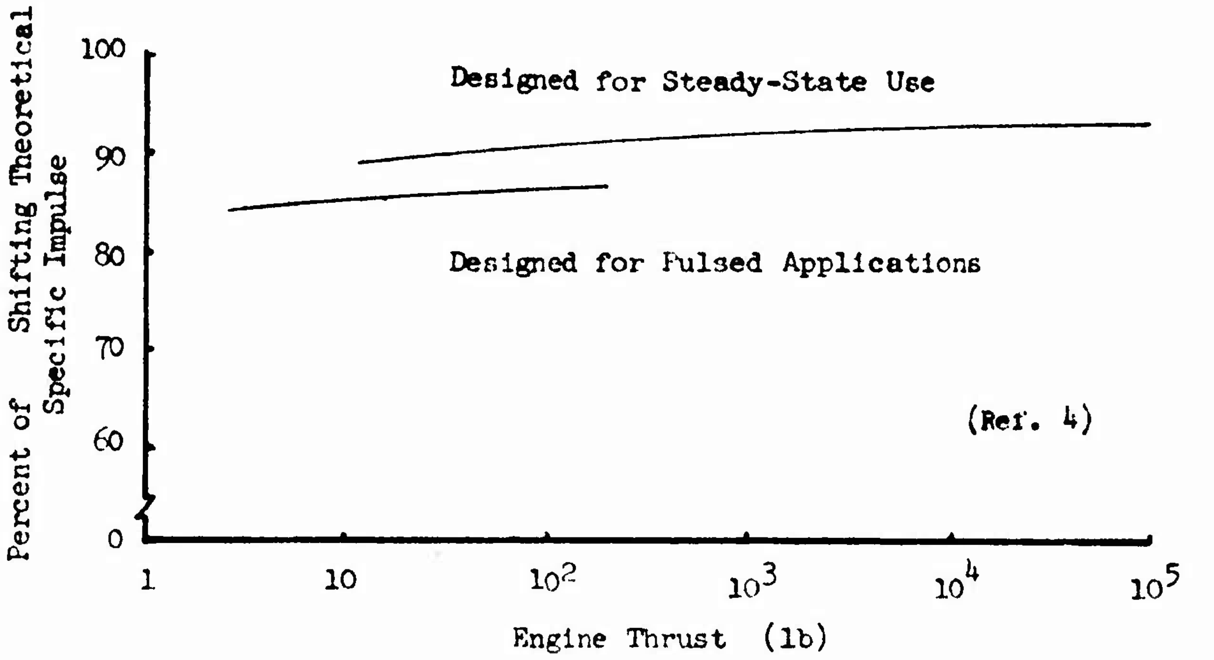
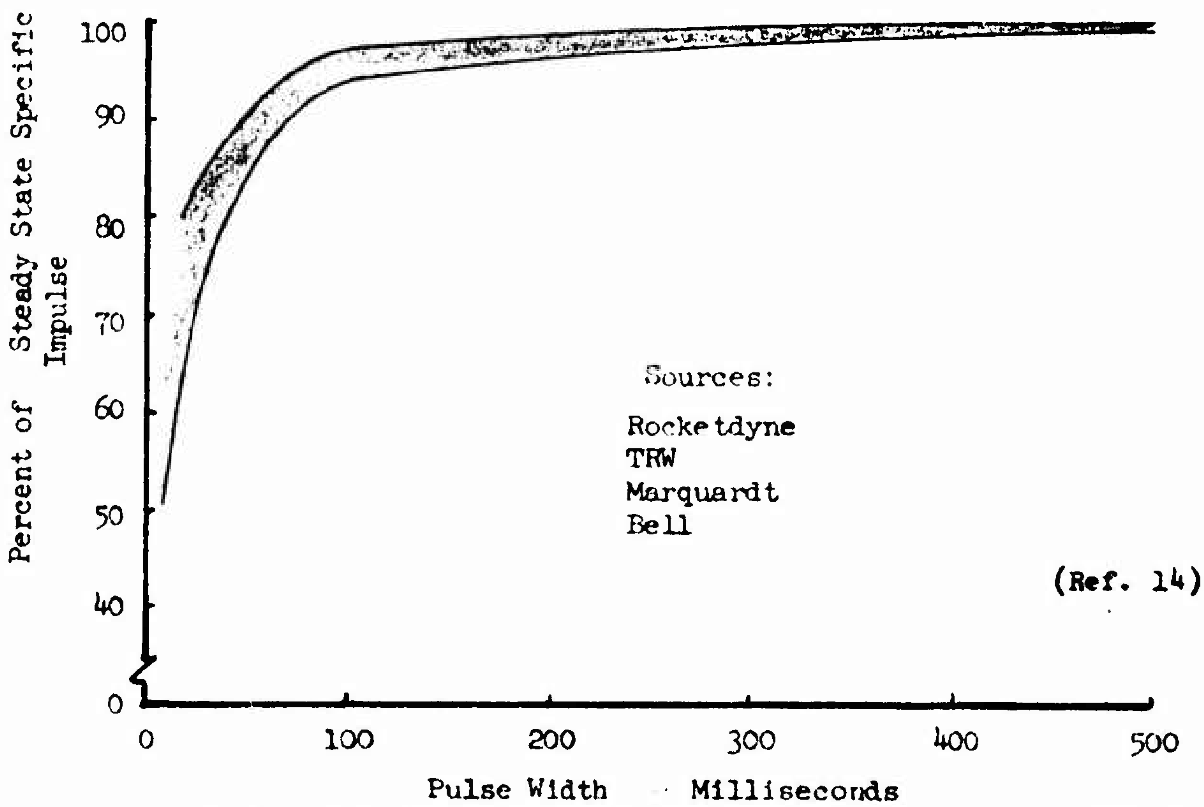


FIGURE 6.4-12 BULK DENSITY OF BIPROPELLANT COMBINATIONS



6.4-13 DELIVERED PERFORMANCE OF BIPROPELLANT ENGINES



6.4-14 PERFORMANCE OF RADIATION COOLED PULSE ROCKET ENGINES

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discontinuity observed in these data can be attributed to whether the engine was designed for primarily steady state, or pulsing operation.

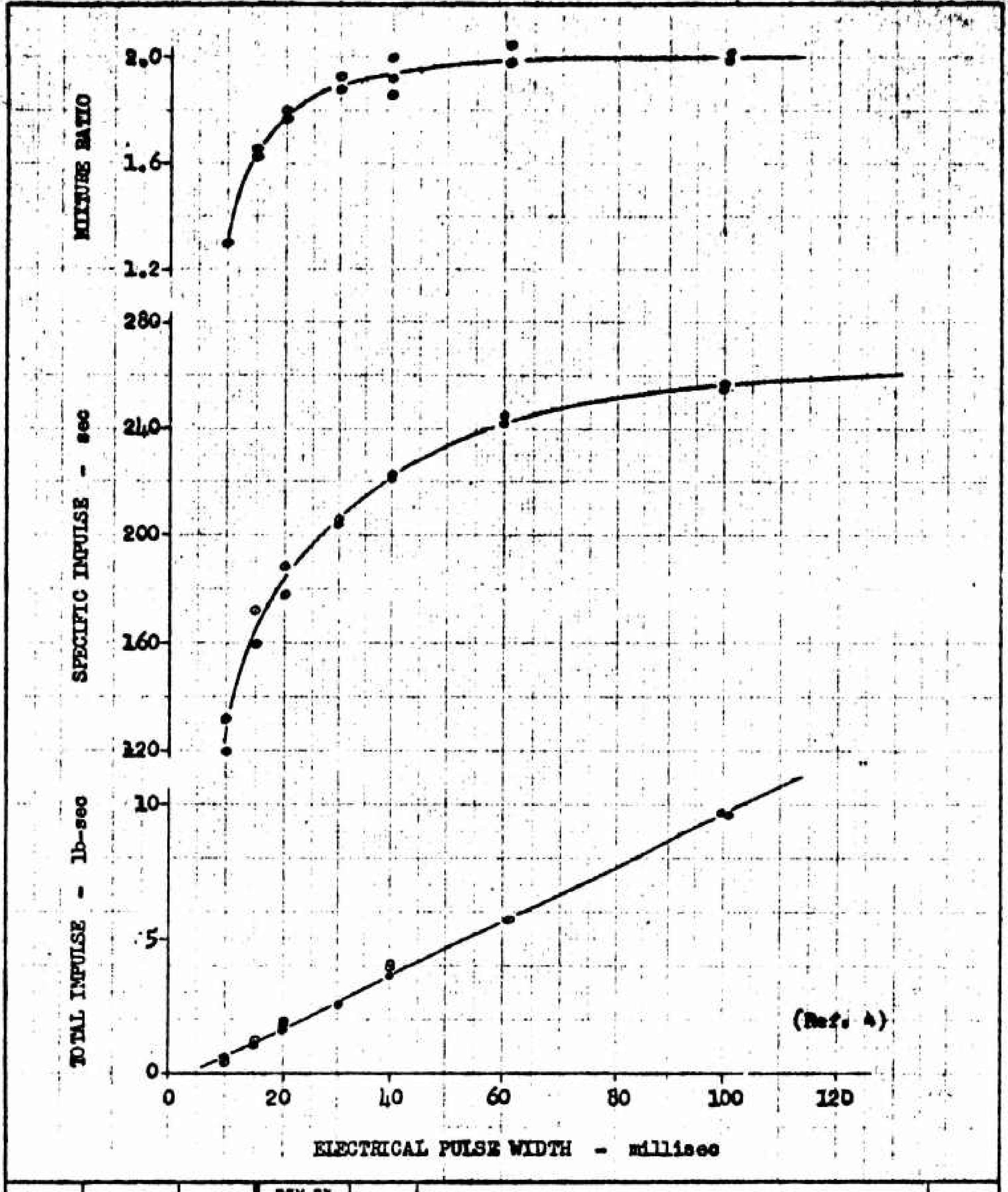
As operating duration is decreased, the relatively constant thrust build-up and decay transients represent a greater portion of the delivered impulse. Transient operation is less efficient since operating characteristics greatly exceed the conditions for which the engine was designed for greater performance, including propellant mixture ratio. This results in decreasing performance as pulse size (i.e., width) decreases. The variations in performance for these conditions is shown in Figure 6.4-14 as a function of delivered steady state performance.

Figure 6.4-15 shows the performance of a typical bipropellant rocket engine using earth storable propellants.

It is sometimes desirable to impart very small impulse bits to perform very small maneuvers. Consequently, the engine capability to provide this control, known as "minimum impulse bit" capability, frequently becomes a very important factor in engine design or selection. The capability for very small impulse bits is determined primarily by how fast the engine valves can be operated. The "hold-up volume," or line volume between the valves and the injector is also important. Engines designed with this capability and good pulsing performance in mind are termed "close-coupled, fast response" engines. As a gross rule, minimum impulse bit capability for these close coupled, fast response engines can generally be estimated by:

$$I_{T_{Min}} = .01 + .004\,F \qquad (6.4\text{-}1)$$

Minimum impulse bit tolerance generally will not exceed $\pm$ 10% of $I_{T_{Min}}$.



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
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6.4.6 BIPROPELLANT THRUSTOR DESIGN

Bipropellant engines can be configured for spacecraft preliminary design exercises by using the following generalized procedure:

1) Assume engine thrust level, F, and duty cycle. These are established for velocity control engines by evaluating limits to maneuver duration, acceleration, gravitational environment, control authority, engine location and duty cycle. Maximum thrust levels are set by acceleration limits for structural or control purposes, single pulse minimum maneuver velocity limits, and engine size and weight. Minimum thrust levels are set by maximum maneuver time limits imposed by engine life, performance penalties associated with finite burn time effects, and thermal, power and communication limits associated with being in the maneuver position.

   Thrust level limits for reaction control are evaluated from disturbance torques, minimum impulse bit, response rate, and engine location for limit cycle operation and for all attitude positioning maneuvers.

2) Assume thrust chamber pressure, P_c, consistent with the following values representative of current practice:

   a) P_c (radiation cooled engines) = 50 to 100 psia
   Current radiation cooled bipropellant engines in the 0.2 to 200 lbs thrust range are evenly distributed throughout this chamber pressure range.

   b) P_c (ablative type engines) = 100 to 150 psia
   Current ablative bipropellant engines in the 5 to 150 pound thrust range are rather evenly distributed between 80 to 150 psi chamber pressure.

   c) P_c (regeneratively cooled engines) = 100 to 300 psia
   The few regeneratively cooled bipropellant engines designed fit in the

20 to 200 lb thrust range and are generally throttling engines. Chamber pressures vary, consistent with thrust, with approximately 15 to 150 psia.

3) Assume nozzle expansion ratio, ϵ. A good starting point is $\epsilon = 40$ at which the majority of small bipropellant engines are designed.

4) Determine propellant specific impulse, Isp for steady state operation from Figures 6.4-11 thru 6.4-15.

5) Determine propellant flow rate from:

$$\dot{w} = F/I_s \tag{6.4-2}$$

6.4.7 BIPROPELLANT ENGINE GEOMETRY

Dimensions of bipropellant engines will vary somewhat depending on the basic cooling technique, amount of insulation, and location of engine valves. In preliminary design exercises, the thrust chamber and nozzle major diameter are usually the nozzle exit diameter, d_e, which can be estimated by:

$$d_e = K_e + 0.84 \left[\frac{F\epsilon}{P_c}\right]^{\frac{1}{2}} \tag{6.4-3}$$

where: K_e = a constant relating nozzle exterior to interior dimensions in the exit plane, and having a value of :

K_e = 0.3(radiation and regeneratively cooled engines)

= 1.0(ablative engines)

Some latitude is available in setting the major engine assembly diameter, d_T, which is frequently affected by such items as engine mounts and propellant values. However, current practice essentially follows:

$$d_T = d_e + K_d \tag{6.4-4}$$

where K_d = a constant relating major assembly diameter to nozzle exit diameter, and having a value of:

$$K_d = 2.9 \text{ (radiation and regeneratively cooled engines)}$$
$$= 2.2 \text{ (ablative engines)}$$

Total length of a typical bipropellant engine assembly, L_T, including propellant valve can be estimated in preliminary design exercises with:

$$L_T = 3 + \left[\frac{F}{25.5}\right]^{1/2} \left[(\epsilon)^{1/2} - 1 \right] \tag{6.4.5}$$

6.4.8 BIPROPELLANT ENGINE WEIGHT

Bipropellant engine weight is affected by the cooling techniques used. Engine vendors have conducted numerous parametric studies to relate, for each cooling method, engine weight to thrust, chamber pressure, expansion ratio, and operating duration. Such studies attempt to show weight relationships and, frequently, to indicate specific regions of preference. Experience has shown, however, that they frequently do not correlate well with actual engine designs. This can happen because (1) some factors are really not amenable to parametric treatment, (2) some parameters can be valued differently depending on cooling concept, (3) cooling concept can affect the weight of non-propulsive spacecraft equipment, and (4) there is a degree of optimism which accompanies studies not immediately related to hardware. Vendor differences in cooling concept, design, construction and materials are difficult to relate parametrically. Standard material gages and component sizes actually introduce step functions into weight comparisons. There are also installation and mission related weight factors pertaining to environment, engine mounting, and duty cycle which may obviate direct comparisons. To include these effects, the following weight expressions have been derived from the weight and design characteristics of more than 40 different bipropellant rocket engines between 0.2 and 200 pounds thrust, and using earth storable propellants.

a) <u>Full Ablative Engines</u> - The weight of a full ablative engine, W_a, that is, an engine having an ablative thrust chamber, an ablative nozzle assembly, propellant valves, injector, inlet plumbing, fittings, wiring, cabling, and engine mounts, can be roughly estimated from the following expression:

$$W_a = 2.5 + .05\,F \qquad (6.4\text{-}6)$$

where: F = engine thrust level between 5 and 100 lbs and the nozzle expansion ratio is about 40.

This expression does not resolve engine weight variations as a function of operating duration.

b) <u>Radiation Cooled Engines</u> - The weight of a radiation cooled bipropellant engine, W_R, including valves, injector, inlet plumbing, fittings, wiring, cabling, and engine mounts can be estimated from the following expression:

$$W_R = 0.161 \quad 5 + F^{0.4} + \frac{F^{0.85}\,(\epsilon + 10)}{P_c} \qquad (6.4\text{-}7)$$

c) <u>Regeneratively Cooled Engines</u> - No mathematical model can be given for small, regeneratively cooled, bipropellant rocket engines since too few such engines exist upon which the analysis could be based.

6.4.9 DEVELOPED BIPROPELLANT THRUSTORS

Table 6.4.3 lists bipropellant engines developed for spacecraft applications and for research purposes.

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TABLE 6.4-9 DEVELOPED BIPROPELLANT THRUSTORS

THRUST (LBS)	PROGRAM CONTRACT OR AGENCY	VENDOR	MODEL	TYPE CHAMBER/ NOZZLE	PROPELLANT OX FUEL	MIXTURE RATIO (O/F)	I_s (SEC)	P_c (PSIA)	P_{in} (PSIA)	ϵ (A_e/A_t)	DIMENSIONS-INCHES W/O VALVE WIDTH	DIMENSIONS-INCHES W/O VALVE LENGTH	DIMENSIONS-INCHES WITH VALVE WIDTH	DIMENSIONS-INCHES WITH VALVE LENGTH	WEIGHT (LBS) W/O VALVE	WEIGHT (LBS) WITH VALVE	τ_T MIN	POWER (WATTS)	MAX LIFE (SEC)	L*
0.5	IR&D	BELL		RADIATION	O_2CH_4		360	42		50	1.0	2.6	5.6	2.8		0.20				
1.0	IR&D	BELL	8999	RADIATION	N_2O_4/A50		255	50		60				5.95	0.29	0.89		15		30
1.0	NAS-5458	BELL	8360	RADIATION	N_2O_4/A50	1.4	255	50		60	2.73	4.8	2.73	7.2	0.52	1.10		20	3400	
5.0	IR&D	BELL	8348T	RADIATION	N_2O_4/A50, MMH	1.6	275	60		40	2.63	5.8	2.63	8.0	1.07	1.65		20	3600	23
5.0	IR&D	BELL	8348C	RADIATION	N_2O_4/A50, MMH	1.6	275	60		40	2.63	5.8	2.63	8.0	0.72	1.30		20	3400	23
5.0	ADV SYN-COM	MARQ.	MA-124	RADIATION	N_2O_4/MMH, MHF	1.6	283	96	170	40	2.2	4.4	3.95	5.2		1.5	0.29	35	9600	
5.0	IR&D	MARQ.	R-9A	ABLATIVE	N_2O_4/MMH, MHF	1.6	280	135	210	40	2.8	5.0	3.6	6.55		3.2	0.028		500	
5.0		UTC		ABLATIVE	N_2O_4/MMH	1.6														
16	AGENA, GEMINI	BELL	8101 8250	RADIATION	MON/UDMH	1.1	252	81		55.6		7.3				2.2			2650	32
16		BELL	8500	RADIATION	N_2O_4/A50	1.0	276	81		55.6	4.2	6.55	4.2	10.8	2.7	3.5		20	3600+	32
20	IR&D	MARQ.	R-7B	ABLATIVE	N_2O_4/MMH, MHF	1.6	282	130	210	40	2.8	6.8	4.8	7.9		3.9	0.10	35	360	
20	IR&D	UTC		ABLATIVE	N_2O_4/MMH															
20	NAS-B-20795	AGC																		
22	IR&D	BELL		ABLATIVE	N_2O_4/UDMH	1.0	290	40		S.L.					2.4					
22	IR&D	BELL	8999-T	RADIATION	N_2O_4/A50, MMH	1.3	290	40		50	4.0	7.5	4.0	10.38		3.2				
22	IR&D	BELL	8999-C	RADIATION	N_2O_4/A50, MMH	1.3	290	40		50	4.0	7.5	4.0	10.38		1.8				
22	MOL	MARQ	R-1E	RADIATION	N_2O_4/MMH, MHF	1.6	276	93		40	3.5	6.1	6.0	10.1		2.88	0.012	35	13,200	
23.5	GEMINI	RD	SE-6	ABLATIVE	N_2O_4/MMH															
25.0	GEMINI	RD	SE-7	ABLATIVE	N_2O_4/MMH															
25.0	TRANSTAGE	RD	SE-9	ABLATIVE	N_2O_4/MMH															
25.0	GEMINI	TRW			N_2O_4/MMH	1.3	295	150	295	40	4.8	4.8	4.8	7.2		4.75	0.025	24.7	600	
25	IR&D	TRW	URSA25R	RADIATION	N_2O_4/MMH	1.6	292	90	185	40			3.7	11.2		4.2	0.20	70	10^6	
25	IR&D	BELL	8441	RADIATION	N_2O_4/A50	1.6	290	80		40	3.62	7.2	3.62	10.5	2.1	2.9		28	3400	12
45	TRANSTAGE	RD	SE-9	ABLATIVE	N_2O_4/A50															
48	AGENA	RD	SE-5	ABLATIVE	N_2O_4/.75N_2H_4-.25 MMH	1.42	272	150	280	40	4.5		4.5	14.0					600	
72	S-IVB ULLAGE	RD	SE-7	ABLATIVE	N_2O_4/MMH	1.72	274	102		40										

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TABLE 6.4-3 DEVELOPED BIPROPELLANT THRUSTORS (Cont'd)

THRUST (LBS)	PROGRAM, CONTRACT OR AGENCY	VENDOR	MODEL	TYPE CHAMBER/NOZZLE	PROPELLANT OX/FUEL	MIXTURE RATIO (O/F)	I_s (SEC)	P_c (PSIA)	P_{in} (PSIA)	ε (A_E/A_T)	DIMENSIONS - INCHES W/O VALVE WIDTH	W/O VALVE LENGTH	WITH VALVE WIDTH	WITH VALVE LENGTH	WEIGHT (LBS) W/O VALVE	WEIGHT (LBS) WITH VALVE	I_T MIN	POWER (WATTS)	MAX LIFE (SEC)	L*
75	AF-FO-4611-67-C0095	AGC																		
85	GEMINI	RD	SE-7	RADIATION	N_2O_4/MMH															
93	IR&D	TRW	APOLLO 93	RADIATION	N_2O_4/MMH	2.0	266	150	295	9.1	4.8	4.8	4.8	7.2		6.75	0.25	24.7	[illegible]	
100	UNIV. ENG. NAS7-305	BELL	8374	RADIATION HEAT SINK	N_2O_4/MMH	1.6	298	80	175	40	8.35	16.0	8.35	19.0		6.82	[illegible]		3600	
100	NASA-MSC NAS9-1522	BELL	8290	RADIATION	N_2O_4/A50	2.0	290	80		40	5.9	13.7	5.9	16.7	5.6	6.8		28		[illegible]
100	IR&D	BELL	8999-C	RADIATION	N_2O_4/A50	1.6	290	80		40	4.0	8.8	4.0	16.56	3.5	4.7		30		
100	IR&D	BELL	8999-T	RADIATION	N_2O_4/A50	1.6	290	80		40	4.0	8.8	4.0	16.56	5.3	[illegible]		30		
100 20	MSC-LTV	BELL	8414	RADIATION	N_2O_4/A50	1.6	295	85-16	200	40	7.4	16.25								15
100	APOLLO. LO, MOL	MARQ.	MA-109 R-4D	RADIATION	N_2O_4/A50	2.0	276	96	170	40	5.74	11.26	6.0	13.42	3.04	4.9	[illegible]	26	[illegible]	
100	ORIGINAL APOLLO	MARQ.	R-4B	RADIATION	N_2O_4/MMH, MHF	1.6	307	96	180	48	5.74	11.28	6.0	13.42	3.04	4.9		26	5600	
100	IR&D	MARQ	R-30	ABLATIVE	N_2O_4/MMH, MHF	1.6	286	126	210	40	5.4	10.5	5.7	13.5		9.5	0.48	26	300	
100 87	IR&D	TRW	URSA 100R	RADIATION	N_2O_4/MMH	1.6	295	90	185	52			5.9	13.8		4.7	0.4	30	[illegible]	
100	NAS8-20795	AGC																		
100	GEMINI	RD	SE-7	RADIATION	N_2O_4/MMH															
100	IR&D	UTC		RADIATION	N_2O_4/MMH															
100	UNIV. ENG. C-1	THIOKOL	TD-345	ABL + REGEN.	N_2O_4/MMH	1.6														
104-30	SURVEYOR	THIOKOL		REGEN + RAD.	MON/MMH + H_2O															
130-80	SCOUT, TGT MISSILE	RD	P-4	REGENERATIVE	IRFNA/HYDYNE															
150	LUNAR HOPPER	TRW						85-17		40			5.82		3.83					22
150-30	SURVEYOR BACK-UP	TRW		ABLATIVE	MON/MMH	1.5	291	110-22	370	32.2	5.725		5.725	13.3		8.3			[illegible]	17
180-20	SURVEYOR BACK-UP	TRW	MIRA 180	ABL + REG	N_2O_4, MON/MMH	1.6	295	150-16		38.2				8.5					285	20
200	AGENA/GEMINI	BELL	8101, 8250	HEAT SINK	MON/UDMH	1.15	254	96		15.6	6.0	11.0		15.54	9.9	10.4			680	35
200	IR&D	TRW	URSA 200	RADIATION	N_2O_4/UDMH	1.6	295	90	185	40			8.0	12.4		8.0	1.0	50	[illegible]	

(Refs. 14, 20, 29, 30)

7.0 SYSTEMS CONSIDERATIONS

Certain systems considerations are important in a discussion of velocity control and reaction control thrustors. The thrustor itself is actually a small weight and geometry penalty to the system. But thrustor design significantly affects propulsion system characteristics which, in turn, are usually important to the spacecraft. Spacecraft design is especially sensitive to power requirements, command and control procedures, data provisions, thrust vector and thermal control requirements, propellant performance, and propellant storage pressures. These factors will be briefly discussed.

7.1 INTERFACES

The thrustor itself interfaces with the propulsion system directly and the spacecraft. The propulsion system also interfaces directly with the spacecraft. Both the thrustor and the propulsion system have an interface with the mission as it relates to trajectory duration and sequencing. Primary interfaces are shown in Table 7.1-1, and discussed below:

a) Propellant Feed System -- The thrustor assembly connects directly to the propellant feed system at the inlet side of the engine or engine valves. This connection directs propellants from the feed system to the engine injector. It is affected by propellant flow rate and by propellant supply pressure. Engine chamber pressure is related to feed system pressure drops and propellant supply pressures in the tanks. Hence, engine chamber pressure affects system weight to a large extent since propellant tanks, and frequently,plumbing lines, are designed as pressure vessels.

   The engine to feed system interface also may include torsional and translational loads imparted to feed system plumbing through engine gimbally for

TABLE 7.1-1 ELECTRICAL, MECHANICAL AND STRUCTURAL INTERFACES

Type of Interface	Description	Boundary Definition
Propellant Feed System		
Mechanical	Plumbing attachment to valve inlets	Valve inlet
Mechanical	Torque, translation on gimballing	Valve inlet
Thrust Vector Deflection		
Mechanical	Actuator linkage attachment	Actuator linkage bosses
Mechanical	Pivot location	Engine gimbal assembly
Thermal	Exhaust plume to jet vane	Exhaust plume temperature profile
Structural		
Mechanical	Engine attachment to thrust mount	Attach plane surface, gimbal mount
Mechanical	Clearance in all gimballed positions	Engine assembly total envelope in all deflected positions
Pneumatic	Exhaust plume to spacecraft impingement	Exhaust plume pressure profile
Thermal Control		
Thermal	Heat soak back through thrust mount	Watts at thrust mount attach plane
Thermal	Heat transfer through propellant line to valve inlet	Valve inlet heat flux
Thermal	Exhaust plume to spacecraft heat transfer	Exhaust plume heat rate profile
Thermal	Engine radiation to spacecraft	Engine heat rate profile
Thermal	Engine heater assembly	Heater location
Electrical		
Electrical	Signal to power, control, sequence engine valves	Electrical input to panel
Electrical	Power, readout of engine assembly instrumentation	Electrical input to instrument portion, electrical panel
Service		
Electrical	Electrical connections to check wiring continuity, operation of thrust level selector, and valve position indicators	Electrical connection to allow testing
Hydraulic	Liquid and gas service provisions to spacecraft	Service coupling
Pneumatic		

thrust vector control. A deliberate attempt should be made in this regard to distribute these loads in such a manner as to minimize their effect on gimbal actuator control authority.

b) Thrust Vector Control -- Engines used for velocity control may require some thrust vector control capability. This capability may take the form of gimballing the engine, deflecting engine exhaust by vanes in the exhaust stream, or differential thrust control of several engines. Engine gimballing involves a moveable engine assembly which requires a large geometrical envelope for clearance reasons, mechanical connections for driving actuators and a hinge or multi-axis assembly as a pivot. Loads are applied at the actuator attachments, hinge points, and propellant supply plumbing. Jet vane systems involve a fixed engine assembly. Jet vanes are primarily sensitive to thermal loads from the propellant exhaust to the vane limiting it to use with lower temperature systems. In multi-engine assemblies used for velocity control, differential throttling or pulsing can be used for control of the mean effective thrust vector. The primary interfaces in this case are electrical for command and control, and thermal, as related to engine behavior in the particular duty cycle.

c) Structural Attachments -- The thrustor interfaces either with the spacecraft or with propulsion system structure at a specific attach point at which thrust loads are transmitted. This interface may also affect clearance requirements when gimballed. Precise location of the engine on the thrust mount is important, especially in fixed engine installations, to insure that the nominal thrust vector is directed through the spacecraft center of gravity.

d) Thermal Control -- Thermally, the thrustor affects the spacecraft environment, and is in turn affected by the spacecraft and the space environment. Thrustor induced thermal loads arise from radiation and conduction from the engine and radiation from the exhaust plume. Thermal interfaces exist at the engine attach points, propellant plumbing, gimbal attach points, thermal shielding, and the radiation profiles of the engine and exhaust plume.

e) Electrical -- The electrical interface includes all electrical provisions for power, control and sequencing of fuel and oxidizer valves, and instrumentation and for data readout.

f) Service -- The service interface includes all electrical provisions for verifying electrical continuity and for all service instrumentation. It also includes hydraulic, mechanical, and pneumatic provisions for preflight service and installation.

7.2 SYSTEM POWER

Electrical power is used by the propulsion system to actuate valves on the engine, in the pressurization system, and in the propellant system, to drive actuators for thrust vector control, to process commands and data concerning system operation, and to elevate temperatures locally by electrical heaters.

Figure 7.2-1 shows electrical power requirements for operating many existing stored gas, monopropellant and bipropellant engines. These are total engine valve electrical power requirements for non-redundant installations. Thus, bipropellant engine requirements involve both the oxidizer and fuel valves. These power values should be doubled when redundant valves are involved. It is important to note that these are not minimum power curves, but are regions which show the power requirements associated with actual installations having

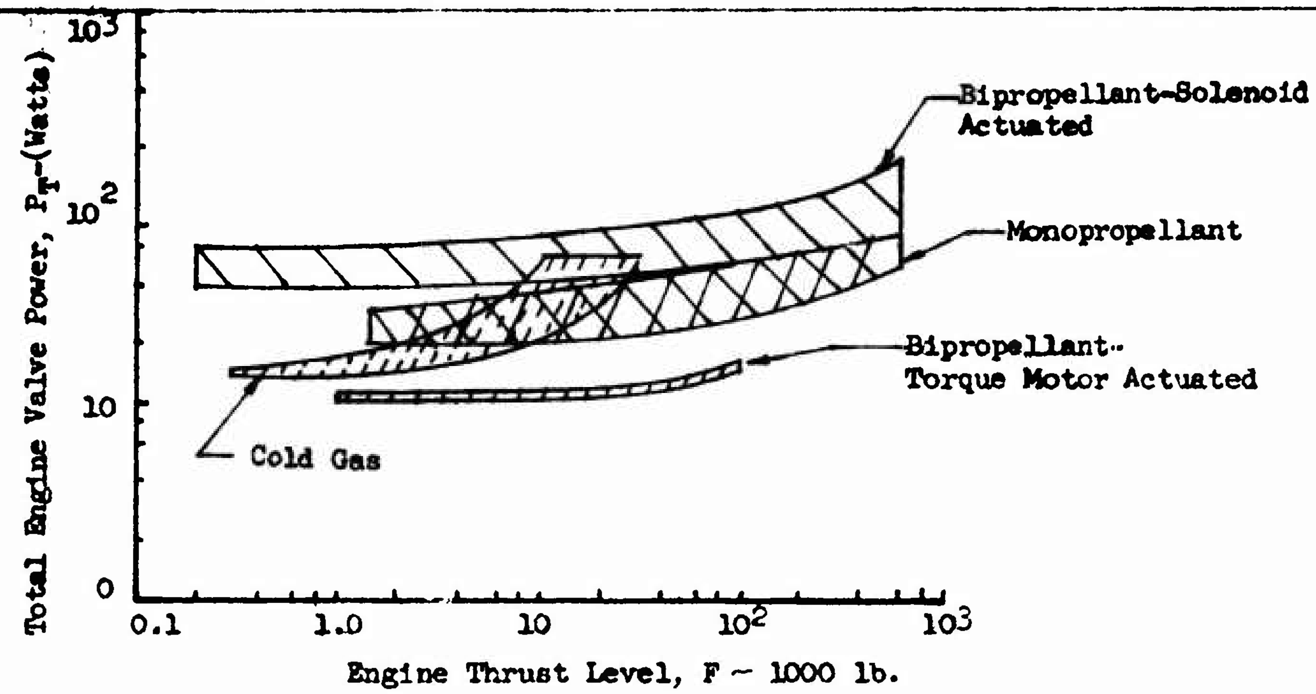


FIGURE 7.2-1 POWER REQUIREMENTS - EXISTING ENGINES VALVES

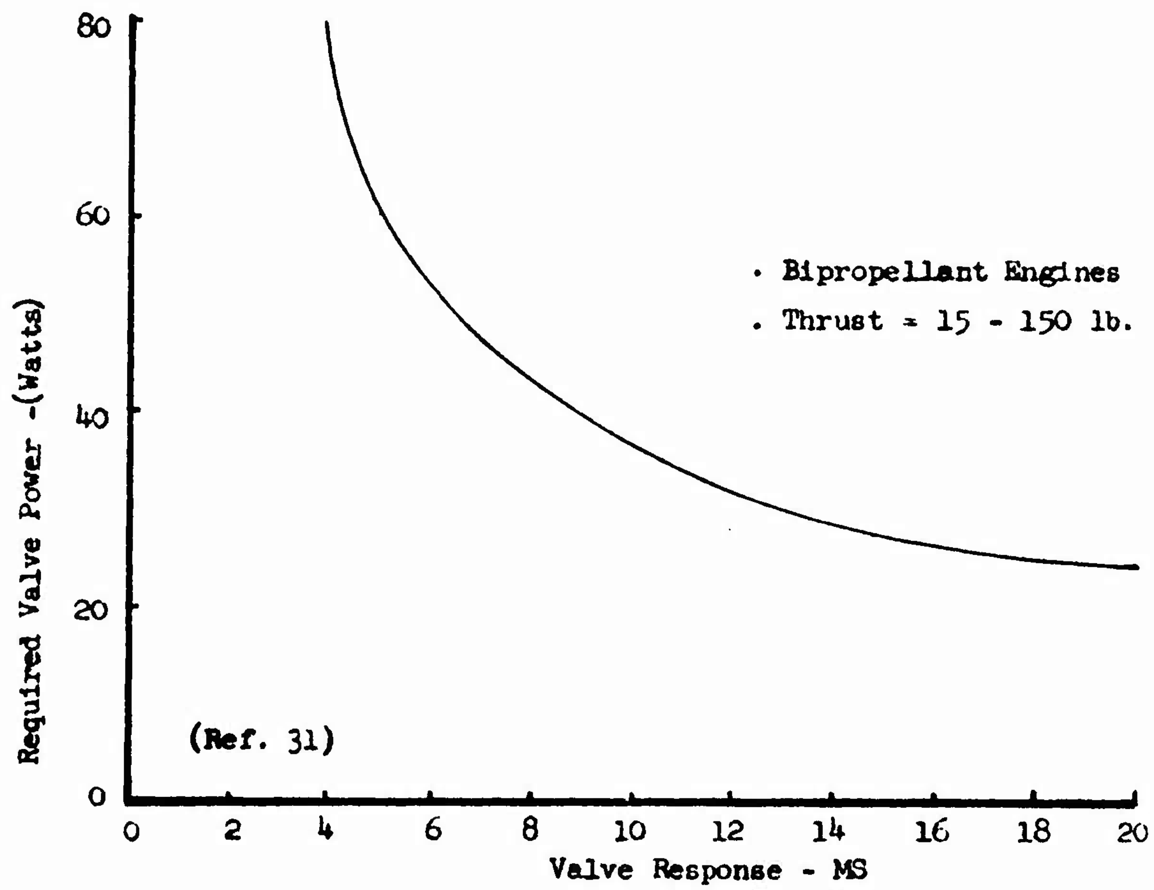


FIGURE 7.2-2 POWER-RESPONSE CHARACTERISTICS OF SOLENOID VALVE

different response conditions. The general flatness of these curves develops from the fact that, in many installations, the same valve is used at different thrust levels and response characteristics. Figure 7.2-2 shows the relationship of valve power to response for a typical solenoid valve.

System power requirements may greatly exceed engine requirements, especially if many valves are used in the pressurization or propellant feed systems.

Power requirements for command, control, and telemetry data concerning propulsion system status are small.

Heated gas thrustor systems using electrical heaters may be expected to have significant weight allotments for power. Similarly, electrical heaters used for thermal control may consume large amounts of energy by operating at low power for extended periods or by operating briefly at high power levels such as during periods of occultation.

In summary, propulsion system power requirments develop from the needs of many components, having different power ratings and operating times. It is important in system design to develop a propulsion system power requirements schedule to identify power consuming elements, to sequence power consuming events, and to help integrate propulsion system and spacecraft power requirements.

7.3 THRUST VECTOR CONTROL

Engines used for velocity control purposes need some sort of directional control over the engine thrust vector to position it properly with respect to the spacecraft center-of-gravity. In numerous bipropellant engines this capability is provided by gimballing the engine in two axes normal to the thrust vector. This procedure requires mounting the engine within a movable assembly of sufficient strength to withstand structural and thrust loads, and to provide actuator attach points of similar capability. Engine inlet plumbing and cabling must also be positioned and designed to minimize their loads on the actuator and to prevent clearance problems.

It is also possible to rigidly fix the engine assembly and direct the effective thrust vector by gimballing the nozzle, or by moving vanes positioned in the exhaust stream at the nozzle exit. Vanes are conveniently used with monopropellant engines since the engine exhaust is free of solid particles and at a low enough temperature to allow simple stainless steel vanes.

Other possible vector control methods include translating the engine or engine components, injecting fluids non-uniformly into the exhaust stream (secondary injection) and by using auxiliary jets. Of all these methods, only the latter has been used in liquid propulsion systems for small spacecraft. Surveyor uses this method by differentially throttling three vernier engines one of which can gimbal in a single axis.

Figure 7.3-1 shows weight estimates of rocket engine thrust vector control systems which are sufficient for preliminary design purposes. The gimbal system consists of a gimbal platform, movable in two axes, which is firmly attached to the engine and to the spacecraft, plus the necessary actuators, wiring and cabling. Alignment adjustments with respect to the spacecraft are commonly provided at the gimbal mount-to-spacecraft mating face. Jet vane assemblies, used in monopropellant engines. consist of four jet vanes, four rotary actuators, a mounting ring, and the necessary wiring and cabling.

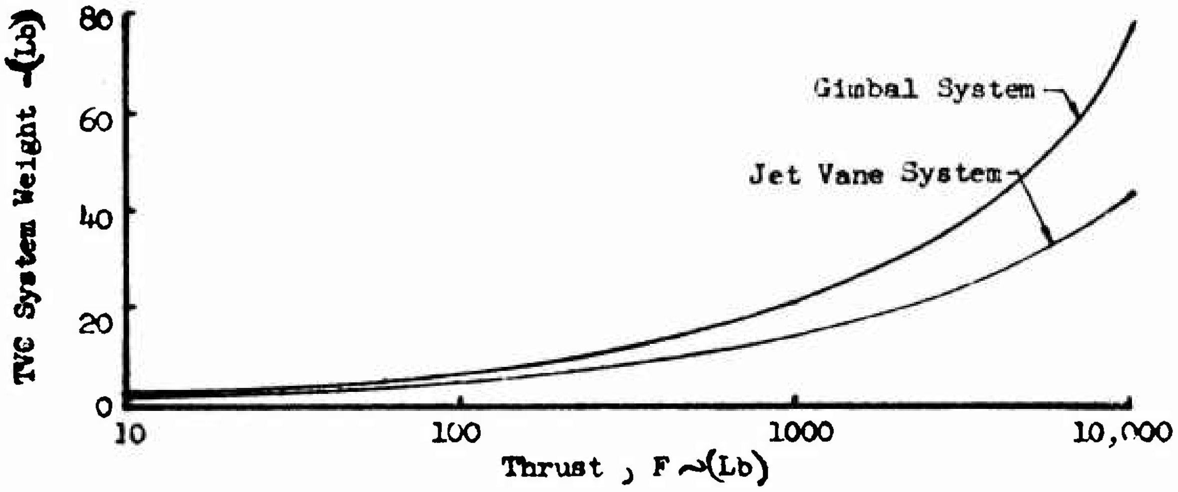


FIGURE 7.3-1 THRUST VECTOR CONTROL SYSTEM WEIGHT

Figure 7.3-2 relates jet vane deflection to effective gimbal angle and specific impulse penalty. Figure 7.3-3 relates jet vane torque motor actuator performance power requirements.

7.4 THERMAL CONTROL

Thermal control of the spacecraft is concerned with, among other things, providing a proper thermal environment to the propulsion system and accommodating the thermal loads which it imposes.

Propellant temperature affects engine performance as it relates to response, impulse bit size, bipropellant mixture ratio, and sometimes, specified impulse. These factors can be resolved at a given temperature. Spacecraft are, however, designed to operate over a range of conditions, hence some variation can be expected in the performance parameters. Absolute limits may also exist such as the 35°F freezing limit of hydrazine under normal conditions. Thermal control provisions, such as heaters, are incorporated to maintain propulsion system operation within specified performance limits.

The engine will absorb or dissipate thermal energy during non-operating periods depending on its orientation with respect to the sun and the character of thermal paths from the spacecraft. Radiation cooled engines are most strongly affected since they are radiators. Proper thermal insulation at engine mounting points is a necessity. The injector in all rocket engines views space through the nozzle throat and can, consequently, exceed its thermal margins unless adequate provisions are made.

The engine and exhaust plume are large radiative heat sources during operation, especially in the radiation cooled engines. Figure 7.4-1 shows the exterior thermal profile of typical rocket engines during steady state operation.

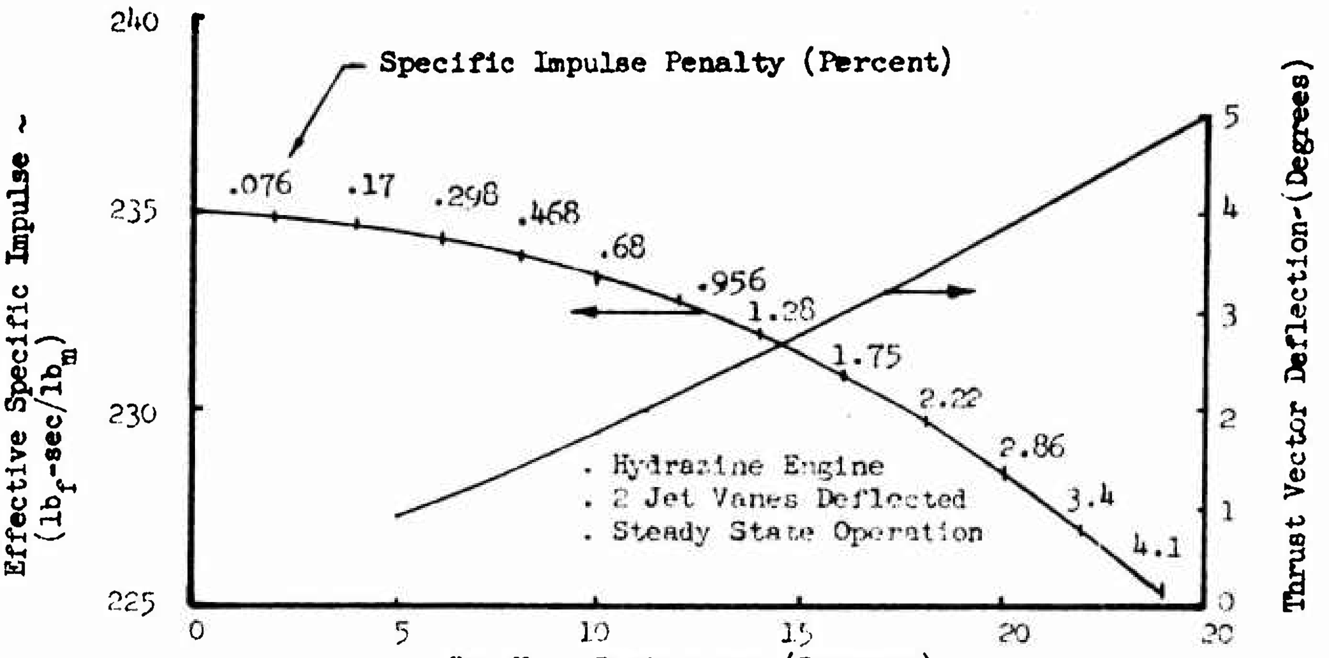


FIGURE 7.3-2 TYPICAL JET VANE TVC PERFORMANCE

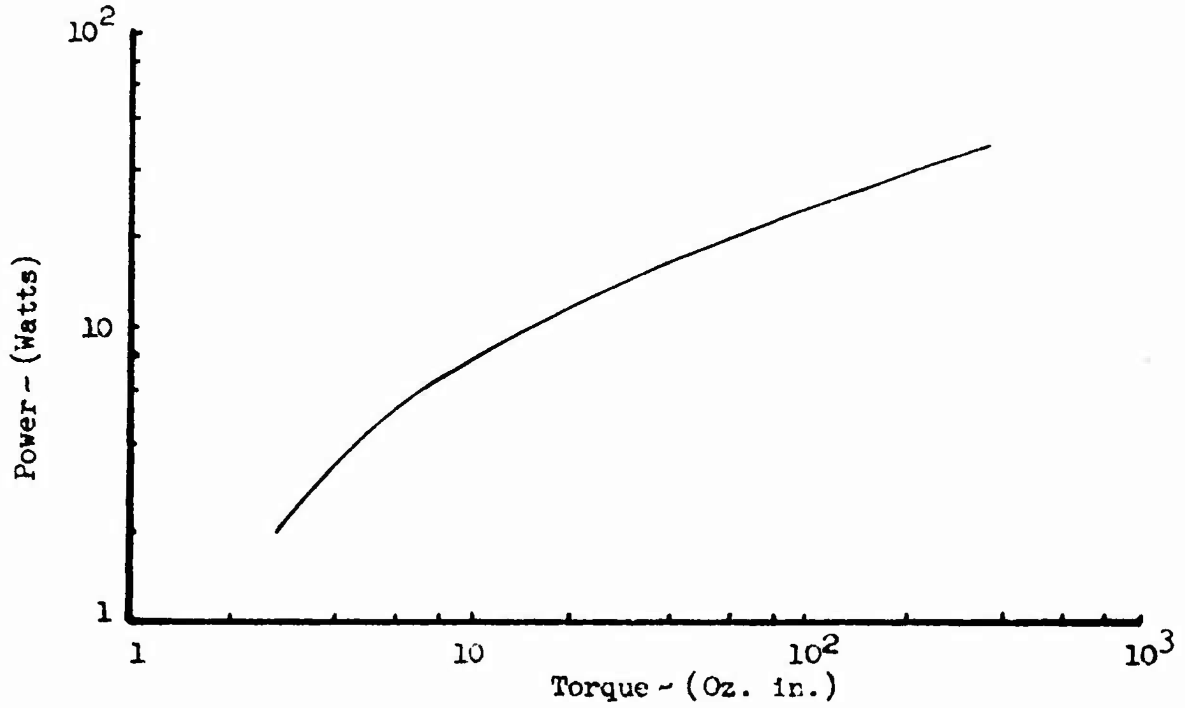


FIGURE 7.3-3 TYPICAL JET VANE ACTUATOR POWER REQUIREMENTS

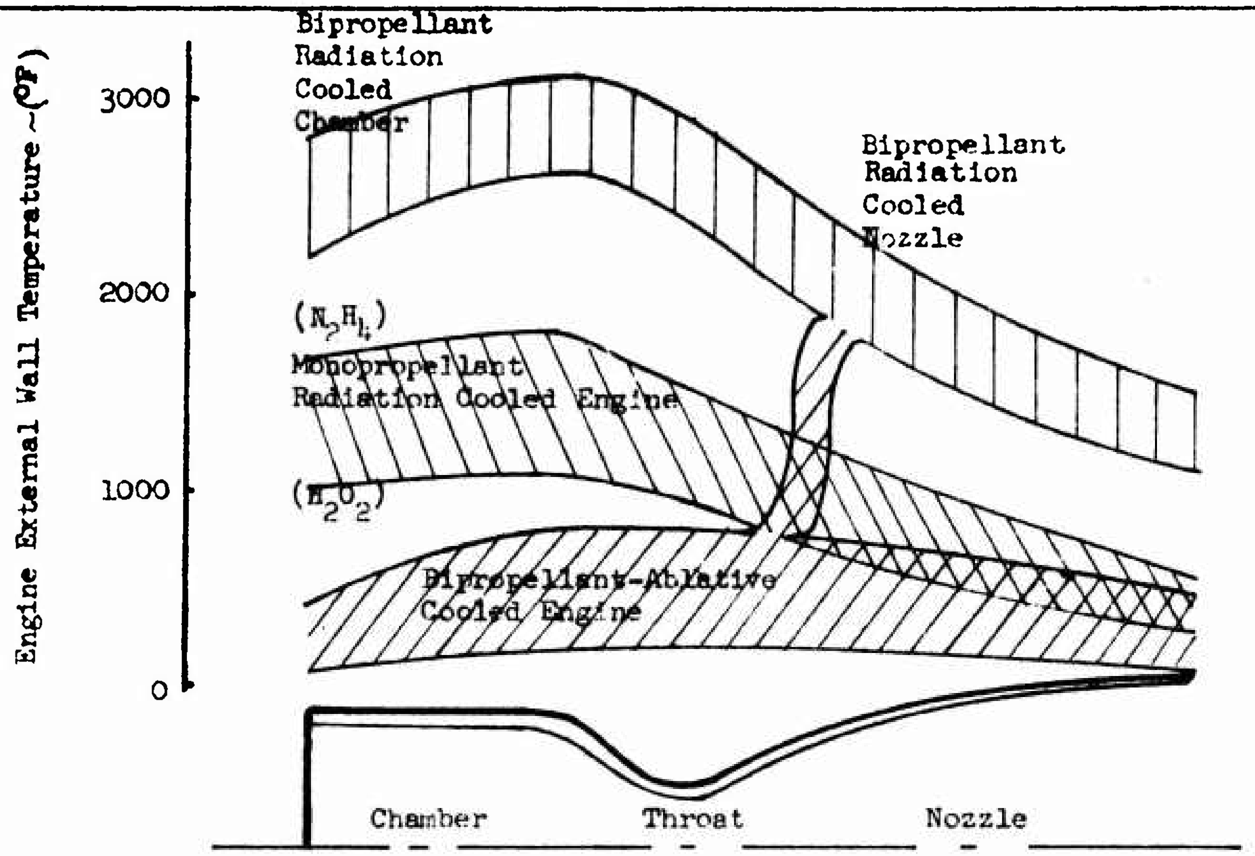


FIGURE 7.4-1 ENGINE EXTERNAL TEMPERATURE PROFILES

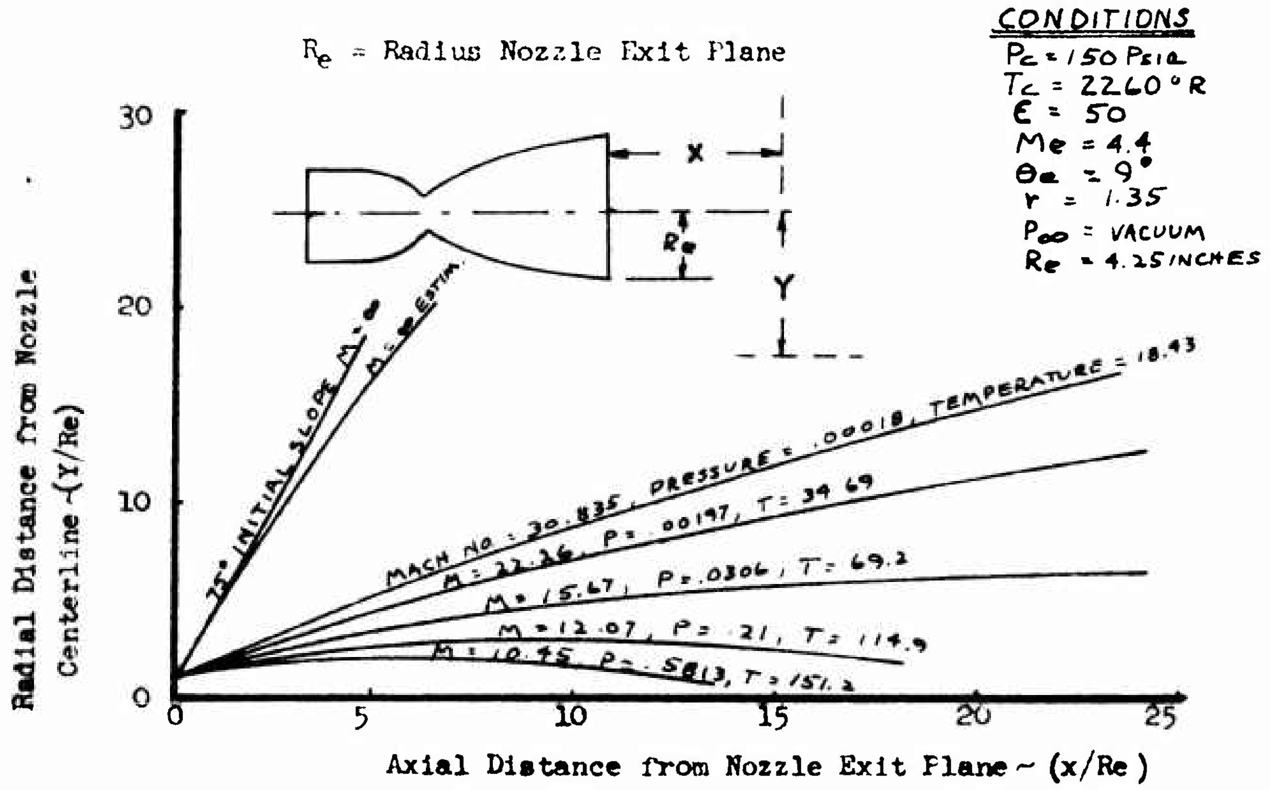


FIGURE 7.4-2 HYDRAZINE ENGINE EXHAUST PLUME (Ref. 32)

Exhaust plume thermal radiation depends on plume size, temperature, and emissivity. Emissivity is especially significant in propellants having solid particles in the exhaust such as with hydrocarbon or aluminized fuels. Plume radiation from stored gases is not significant. With monopropellant engines it can be quite important though emissivity is rather low. Figure 7.4-2 shows the exhaust plume characteristics of a 200 lb. thrust hydrazine engine.

Heat conduction from the engine forward to the inlet plumbing and attach points will occur following shutdown and during certain pulsing duty cycles. This is referred to as heat "soakback" and is undesirable when it causes propellant boiling in the valve or valve inlet or if it induces larger conductive heat loads through attaching structure. Monopropellant engines, which have a high internal chamber mass, store a large amount of "resident heat" which can cause propellant detonation in the injector. These engines are usually designed to insure against post-operative heat soakback to the injector area by providing a thermal short to other areas such as the engine mounting flange.

7.5 LIFETIME

Propulsion system components are subject to life limitations measured by duration or number of cycles. Valves and regulators are primarily subject to cycle limits, usually in the range of thousands to millions of cycles. Propellant tank positive expulsion devices are limited to fewer cycles, possibly no more than one, such as with metallic diaphragms.

Rocket engines are both duration and cycle limited. Cold gas thrustors are sensitive to valve cycle limits. Heated gas thrustors are too, though they are also subjected to heater element lifetime limits. Figures 7.5-1 and 7.5-2 show estimated lifetime characteristics of tungsten wires and tubes

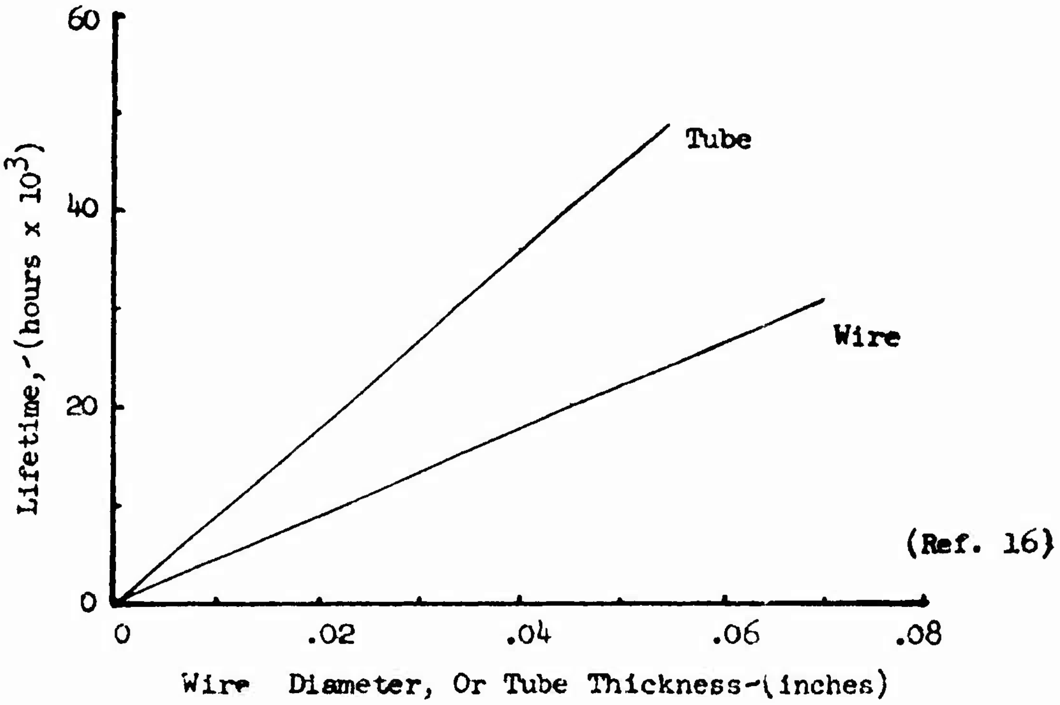
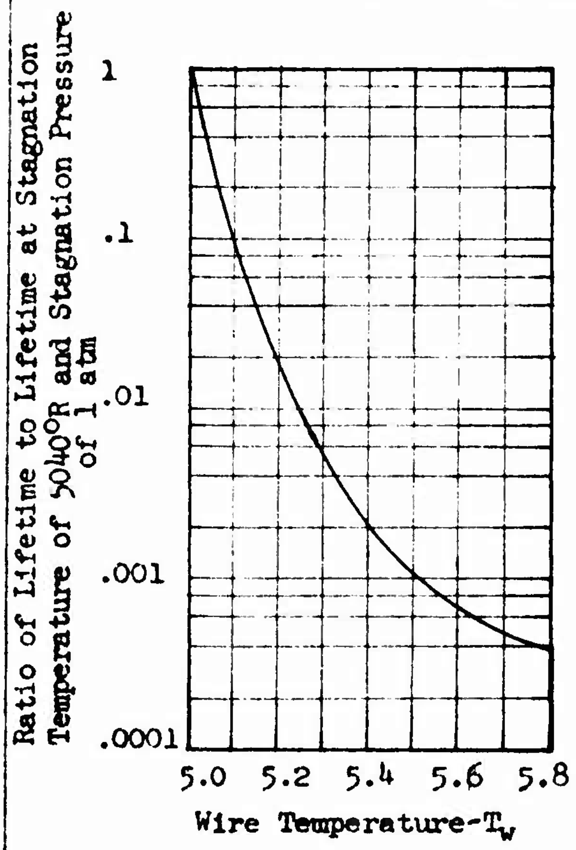


FIGURE 7.5-1 LIFETIME OF TUNGSTEN WIRES AND TUBES IN HYDROGEN



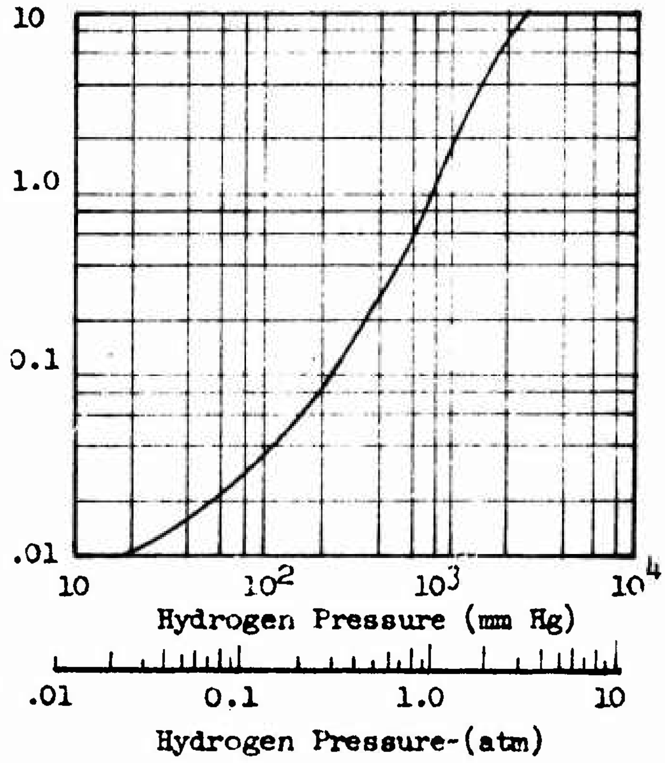


FIGURE 7.5-2 TEMPERATURE AND PRESSURE EFFECTS ON LIFETIME OF TUNGSTEN WIRES AND TUBES (Ref. 16)

in a hydrogen atmosphere. These characteristics are atmosphere dependent, but similar information for other gases was not generally available.

Monopropellant engines are also affected by the cycle limitations of liquid propellant valves, but their lifetime limits are usually discussed in terms of catalyst life. The destructive mechanism which affects catalyst life is erosion coupled with flexing and thermal cycling in wire screens and abrasion in particle catalysts. This is aggravated by extensive pulsing and high temperatures. Figure 7.5.3 shows catalyst life for various hydrogen peroxide thrustors in terms of thrust and operating duration. The lifetime characteristics of a hydrazine thrustor using Shell 405 catalyst is shown in Figure 7.5-4 in terms of chamber pressure and propellant flow time. Lifetime of the non-spontaneous H-7 hydrazine catalyst is better than that of the Shell 405 "spontaneous" catalyst, but hydrazine and peroxide catalyst in general have more lifetime capability than is generally desired from small ΔV engines. Hydrazine monopropellant engines have now demonstrated over a million pulses, and over 8 hours of steady-state operation. Lifetime limits for monopropellant engine parts other than the valves and catalysts have not been identified in test or statistically established.

Bipropellant engine lifetime is primarily related to valve cycling and operating duration. The cycle limits of liquid propellant valves are more significant than with other engines since twice as many valves are involved. Ablative engines are life rated to operating duration for a particular duty cycle. Radiation and regeneratively cooled bipropellant engines have lifetime characteristics similar to monopropellant chambers. Their lifetime rating is usually stated as the qualification test requirement instead of the absolute

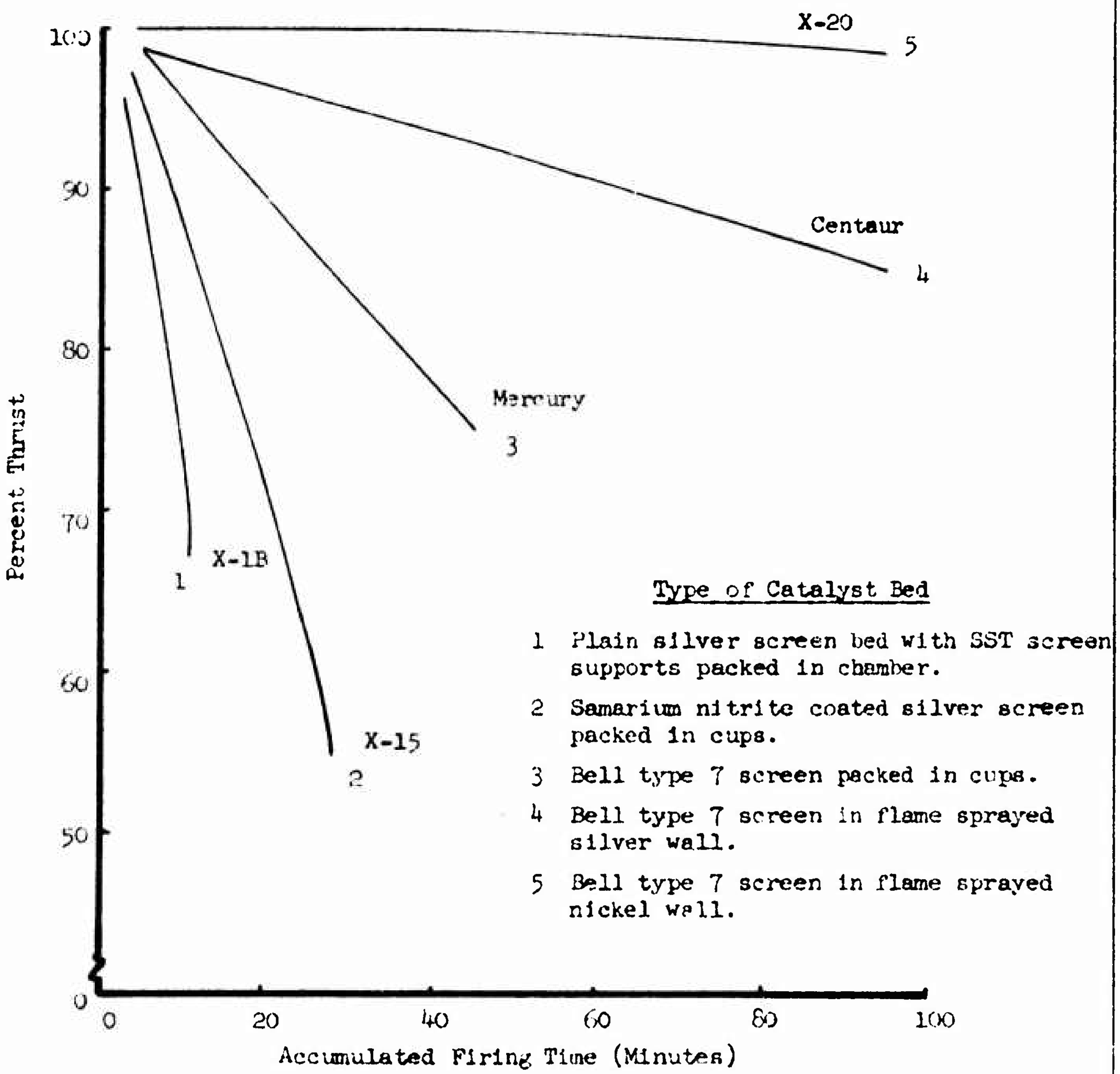


FIGURE 7.5-3 CATALYST BED LIFE-HYDROGEN PEROXIDE (Ref. 27)

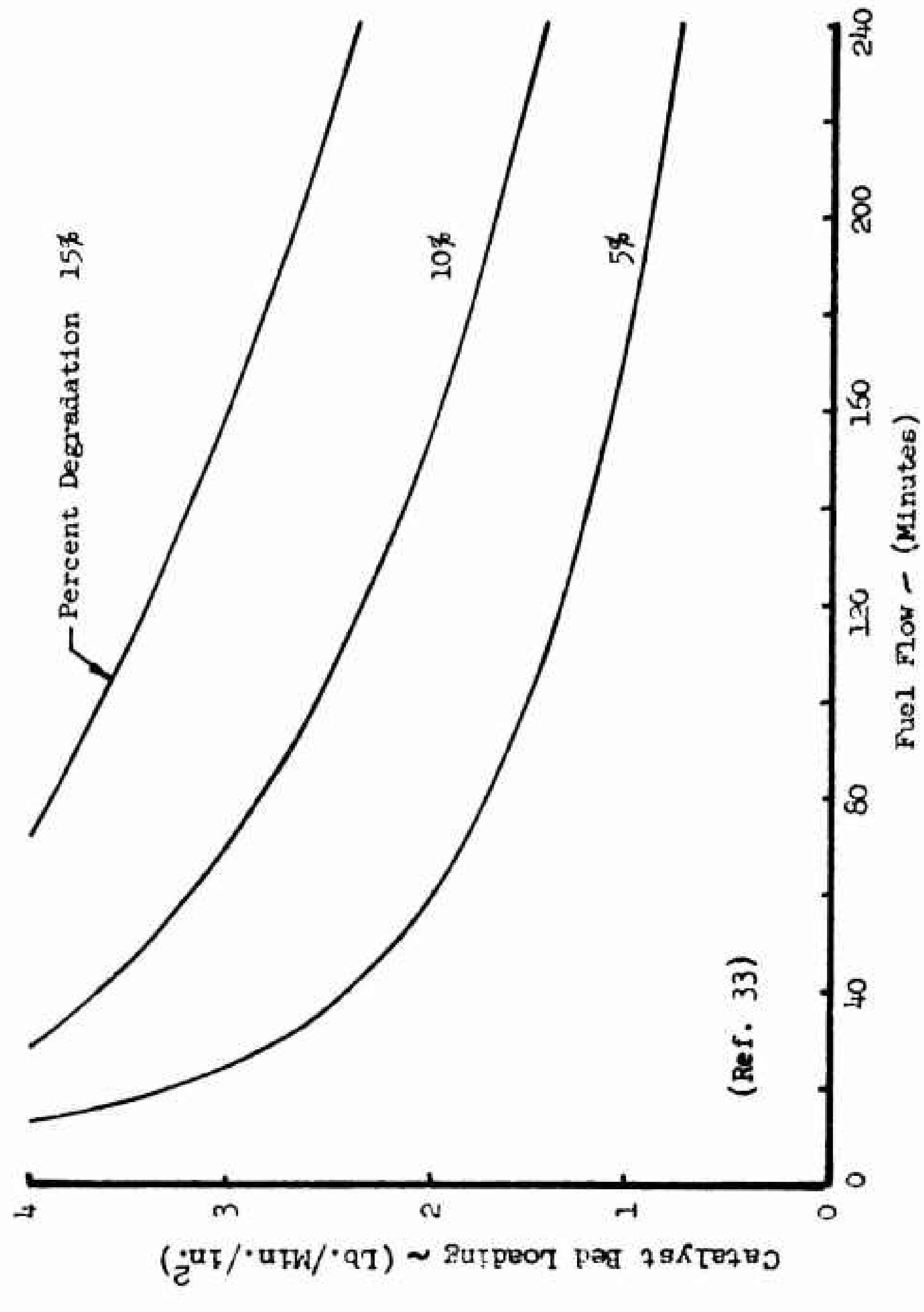


FIGURE 7.5-4 CATALYST BED LIFE CHARACTERISTIC - HYDRAZINE ENGINES

duration to failure since the latter requires more testing, and consequently cost, than is practical for most applications.

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Security Classification

DOCUMENT CONTROL DATA - R&D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author) The Boeing Company	2a. REPORT SECURITY CLASSIFICATION Unclassified
	2b. GROUP

3. REPORT TITLE

ROCKET ENGINES - LIQUID PROPELLANT VOLUME 1 - SMALL ENGINES

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

5. AUTHOR(S) (Last name, first name, initial)

Bader, Jr., H. R.

6. REPORT DATE October 23, 1968	7a. TOTAL NO. OF PAGES 183	7b. NO. OF REFS 33
8a. CONTRACT OR GRANT NO.	9a. ORIGINATOR'S REPORT NUMBER(S) D2-114118-2 Volume I	
b. PROJECT NO.		
c	9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report)	
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13. ABSTRACT

This document contains information on rocket engines using propellants stored as liquids or gases.

Volume 1 contains information on engines used for reaction control systems and low velocity increment propulsion systems. Cold (ambient) gas, heated gas, monopropellant and bipropellant engines are considered. Estimates are made of performance, size and power requirements. Operating principles, hardware details and systems consideration are discussed.

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14 KEY WORDS	LINK A ROLE	LINK A WT	LINK B ROLE	LINK B WT	LINK C ROLE	LINK C WT
Bipropellant						
Cold Gas						
Hot Gas						
Monopropellant						
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Reaction Control Systems						
Thrustors						
Resistojets						
Radioisotope Thrustors						
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