even larger savings are indicated for the future. It is concluded that, when large numbers of small, solid propellant rocket motors are to be tested under simulated altitude conditions, substantial gains in testing economy and data accuracy can be achieved through use of automatic testing equipment.

References

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Pressure-Fed Liquid Rocket Payload Potential

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Nomenclature

- c_0 = oxidizer heat capacity, Btu/lb-°F
- g = gravitational const (earth), ft/sec²
- I_{sp} = propellant specific impulse in vacuum, sec
- k_i = effective thermal conductivity, Btu/hr-ft-°F
- R = mixture ratio
- t_m = mission time, hours
- ΔT_0 = oxidizer temperature rise, °F
- ΔT_{0i} = temperature difference, oxidizer to insulation surface, °F
- ΔV = velocity increment, fps
- W_v = initial gross vehicle weight, lb
- W_{pe} = weight of propellant expelled, lb
- W_{pr} = residual propellant weight, lb
- W_{tkg} = weight of propellant tanks, gas tank, and gas, lb
- W_i = insulation weight, lb
- α = tankage factor, lb/ft³
- β = propellant payload fraction
- δ_v = propellant fraction
- η_e = propellant expulsion efficiency
- ρ_p = propellant bulk density, lb/ft³, g/cm³
- ρ_0 = oxidizer density, lb/ft³, g/cm³
- ρ_f = fuel density, lb/ft³, g/cm³
- ρ_i = insulation density, lb/ft³
- $\dot{r}$ = insulation fraction per unit storage time, h_r^{-1}

Introduction

THE pressure-fed liquid rocket is an extremely versatile design concept. In the lower limit, the design can be as simple as a solid motor for applications requiring a single programed impulse. In the upper limit, it can be designed for any range or sequence of thrust and/or impulse modulation required for the most sophisticated space missions. This note compares the potential payload capabilities of high-energy cryogenic and storable propellant combinations in pressure-fed rockets for spacecraft missions. Payload fractions were determined with respect to mission time and velocity requirements.

Minimal Storage Missions

Comparison of pressure-fed propulsion systems can be made relatively absolute because of the nearly exact equivalency that can be established between systems utilizing different propellant combinations. The detailed evaluation of a multitude of design cases can therefore be circumvented by the device of comparing the payload capabilities on the "common denominator" of equal initial vehicle weight, velocity increment, and propellant storage and expulsion subsystem design criteria. The common design criteria selected are 95% theoretical propellant specific impulse for 100 psia chamber pressure and a 50/1 nozzle expansion ratio, 98% propellant expulsion efficiency, and a tankage factor of 2.8 lb/ft³ based on spherical tanks operating at a pressure of 150 psia. The propulsion systems employing liquid hydrogen are an exception only to the extent that they optimize at a lower chamber pressure of approximately 70 psia, with corresponding tank pressures of 120 psia.

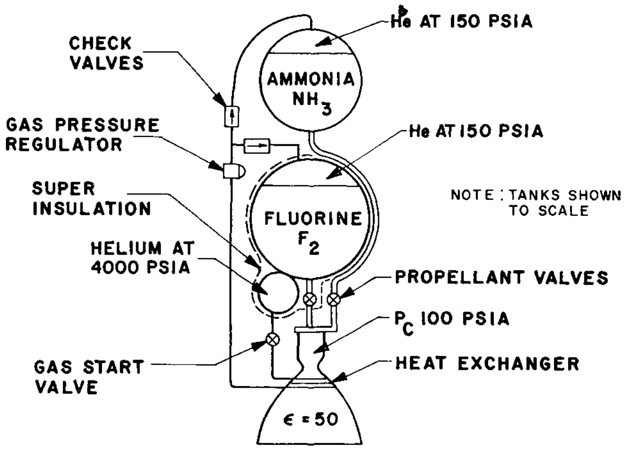

Fig. 1 Pictorial schematic of typical propulsion system.

The tankage factor (α) is the weight of pressurizing gas, gas tankage, and propellant tankage per cubic foot of propellant expelled; 2.8 lb/ft³ is the potential value for earth-storable propellant systems pressurized with unheated helium and for the cryogenic propellants with helium gas storage at the temperature of the colder propellant and with a heat exchanger to raise the temperature of the delivered helium to that of the warmer propellant. The propellant expulsion efficiency (η_e) of 98% is the over-all value, taking into account the residual propellants (liquid and vapor) resulting from operating mixture ratio error and tank expulsion efficiency. A typical propulsion system designed to this criterion is shown schematically in Fig. 1.

The rocket performance criterion chosen is a propellant payload fraction parameter,

$$\beta \equiv (W_v - W_{pe} - W_{pr} - W_{tkg})/W_v \tag{1}$$

This fraction (β) includes all elements of rocket vehicle weight not primarily determined by the propellant selection, i.e., net payload, guidance system, control system, stage structure, and rocket engine weight. The engine weight is included in β because it is determined primarily by vehicle/flight-plan considerations such as required thrust-to-weight ratio. It can be shown that the definitive Eq. (1) reduces to

$$\beta = 1 - \delta_v[(1/\eta_e) + \alpha/\rho_p] \tag{2}$$

where

$$\delta_v = W_{pe}/W_v = 1 - e^{-x} \qquad x = \Delta V/gI_{sp}$$

Thus β is directly expressed in terms of the propellant properties (I_{sp} and ρ_p), the system design constants (η_e and α), and the mission requirement (ΔV). The conventional specific impulse (I_{sp}) vs mixture ratio (R) data for propellant performance was converted to I_{sp} vs ρ_p data by

$$\rho_p = (R + 1)/[(R/\rho_0) + 1/\rho_f] \tag{3}$$

Figure 2 shows the basic performance of eight propellant combinations for one-day storage time and a ΔV requirement of 16,000 fps. The highest payload line (β) intercepted by

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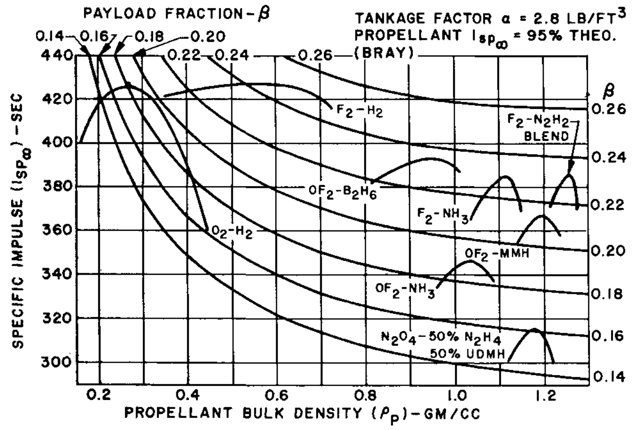

Fig. 2 Propellant payload fraction β for ΔV = 16,000 fps.

the propellant combination performance lines indicates the maximum payload obtainable; the point of intercept defines the I_{sp_∞} and ρ_p (i.e., mixture ratio) required to maximize payload. The constant β lines also show the $I_{sp_\infty}\rho_p$ tradeoff for a constant payload. The propellant performance curves are shown for the terminal propellant storage densities and 95% of theoretical I_{sp} calculated by Bray analysis methods.[1] Figure 3 gives the payload potential (β) as a function of ΔV for the optimum mixture ratio (i.e., ρ_p) of each propellant. These figures apply to minimal propellant storage missions such as imparting earth orbit injection velocity to a spacecraft and/or imparting the velocity change required for flight from earth orbit to the moon or near planets.

Extended Storage Missions

As the storage time of the system in the ready condition increases, the net payload fraction β_n for deep cryogenics will degrade as insulation weight displaces payload weight. The determination of β_n as a function of storage time was based on the following thermal design considerations: it was assumed the effective equilibrium temperature of the external tank surfaces could be passively controlled to any temperature between +100 and −200°F in the portion of space between earth and the near planets. The required absorptivity to emissivity ratio of the tank surface coatings, ranging from approximately 1.0 at the high temperatures to approximately 0.1 at the low temperatures, appears to be within the present state of the art. In this temperature range, the only propellants under consideration that require insulation in excess of the minimal for α = 2.8 lb/ft³ are LO_2, LF_2, and LH_2. The heat leakage rate to these propellants was, therefore, based on an effective insulation surface temperature of −200°F.

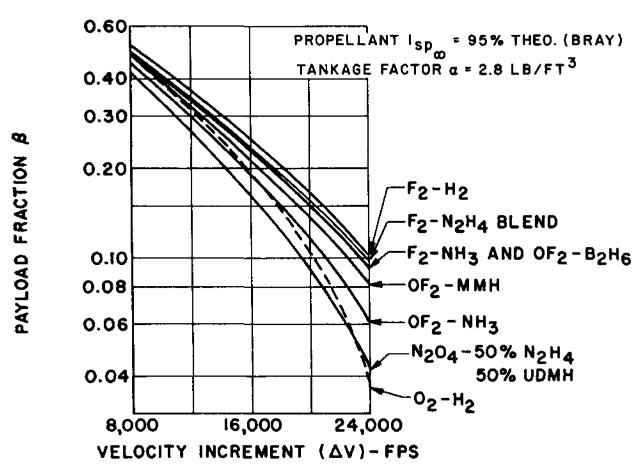

Fig. 3 Payload fraction vs velocity increment.

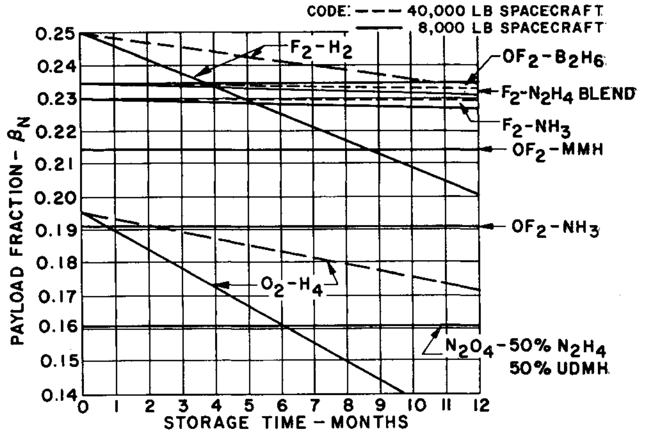

Fig. 4 Payload fraction vs mission storage time for 16,000 fps ΔV.

The maximum propellant temperature was established by a vapor pressure limit of 50 psia for nonvented storage. Initial conditions, for the cryogenics, were taken at the end of the boost period with the propellants at their normal boiling point. (Subcooling was assumed to absorb the heat load during the prelaunch and boost periods.) Sufficient insulation was provided to limit the total heat leakage during the storage period to the liquid heat sink of the propellants. An insulation density of 6 lb/ft³ and conductivity of 1.5 and 2.1×10^{-5} Btu/hr-ft-°F for LH_2 and LO_2/LF_2 service, respectively, were considered to be representative of the potential of current superinsulations. It was assumed that the total heat load consisted of equal parts through the insulation and through the tank supports and piping.

By equating the heat leakage rate and mission time to the heat sink of the propellants, and expressing insulation and propellant weights as functions of R, ρ, δ_v, and W_v, the insulation weight fraction required ($\dot{\tau}$) per unit storage time can be expressed in the form shown for the propulsion system of Fig. 1:

$$\dot{\tau} \equiv W_i/W_v t_m = \frac{46.8\delta_v^{1/3}\rho_i R^{1/3} k_i \Delta T_{0i}}{\eta_e^{1/3} W_v^{2/3}(R+1)^{1/3}\rho_0^{4/3} c_0 \Delta T_0} \quad (4)$$

The solution of equations of this form permits the net propellant payload fraction to be determined with respect to time as follows:

$$\beta_n = \beta - \dot{\tau} t_m = \text{net propellant payload fraction} \quad (5)$$

The effect of the insulation weight penalty associated with increasing mission time on the potential payload (β_N) of these propellants is shown in Fig. 4.

Conclusion

The relative payload differences between systems employing different propellant combinations are more significant and more accurate than the absolute results because they are based on the major determinants of payload for equivalent system design criteria, inertial velocity, mission time, and vehicle weight.

The analysis results show that the potential payload of the high-energy propellants, involving one or two cryogenics, is significantly higher than the earth storables out to and including-one-year missions. The best general purpose propellants for the ΔV and storage time spectrum of missions ranging from earth orbit to the near planets were determined to be F_2-N_2H_4 blend, F_2-NH_3, and OF_2-B_2H_6. However, F_2-H_2 is superior for short storage times and/or high spacecraft weights.

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