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83 Since the beginning of man's exploration and utilization of outer space, many types of propulsion systems for space vehicles have been developed. In the past the major controlling factor in the choice of a propulsion was that of the technical capability on hand for designing and producing the system. Nowadays, however the propulsion engineer has at his disposal such a wide variety of materials and manufacturing processes that the proper Choice of a propulsion system for a given mission is not entirely clear. Here we present a simple method for the rapid determination of the best system given a minimum knowledge of the mission requirements and the materials on hand.

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# A SIMPLE METHOD FOR THE SELECTION OF SATELLITE PROPULSION SYSTEMS

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## INTRODUCTION

Since the beginning of man's exploration and utilization of outer space, many types of propulsion systems for space vehicles have been developed. In the past the major controlling factor in the choice of a propulsion system was that of the technical capability on hand for designing and producing the system. Nowadays, however the propulsion engineer has at his disposal such a wide variety of materials and manufacturing processes that the proper choice of a propulsion system for a given mission is not entirely clear. Here we present a simple method for the rapid determination of the best system given a minimum knowledge of the mission requirements and the materials on hand.

## THE METHOD

Propulsion systems are studied by defining various parameters that relate certain masses of the system. We assume that the total mass of a space vehicle is composed of

$$M_T = M_P + M_S + M_{PL} \quad (1)$$

where  $M_P$  is the propellant mass carried for use during the life of the vehicle,  $M_S$  is the mass of the structure associated with the propulsion system and  $M_{PL}$  is the rest of the mass of the vehicle. We define

$$\epsilon = \frac{M_S}{M_P + M_S} \quad , \text{ a structure coefficient}$$

$$\eta = \frac{M_{PL}}{M_T} \quad , \text{ a mass efficiency}$$

The  $\epsilon$  parameter can be found in any text book of rocketry, but our mass efficiency,  $\eta$ , is new. We believe it to be specifically applicable to space vehicles which must be launched and therefore subject to a more or less rigid total mass constraint, i.e.,  $M_T$  is fixed.

The equation of motion for a vehicle in orbit subject only to the propulsion system thrust force can be re-written as

$$\eta = \frac{\frac{-\Delta V}{g_0 I_{sp}}}{\epsilon - 1} \quad (2)$$

where  $\Delta V$  is the total velocity increment required during the vehicle's mission life,  $I_{sp}$  is the specific impulse and  $g_0$  the gravitational constant.

If it is assumed that the propulsion system structure is dominated by the propellant tanks and, further, the tanks are spherical, it can be shown that

$$\epsilon = \frac{(S_F + (OF) S_O) \rho_s}{(S_F + (OF) S_O) \rho_s + \rho_F + (OF) \rho_O} \quad (3)$$

where

$$S_F = \left(1 + \frac{P_F}{2\sigma_s}\right)^3 - 1 \quad (4)$$

$$S_O = \left(1 + \frac{P_O}{2\sigma_s}\right)^3 - 1 \quad (5)$$

and  $OF$  is the oxidant/fuel mixture ratio by volume,  $\rho$  is the density,  $P$  is tank pressure, and  $\sigma$  is the tank material maximum permitted tensile stress. The subscripts  $F$ ,  $O$ , and  $s$  refer to fuel, oxidant and structural material, respectively.

In Table 1, estimates of these parameters are presented for various propellants. The compressed cold gases are considered to be at 300K and 20atm - while the condensed cold gases are at the vapor pressure corresponding to that

temperature. The monopropellant hydrazine is assumed to be stored at 20atm and the bipropellants are stored at 200atm with chemical reaction taking place in the combustion chamber at 67atm.

Propellant	$\rho_p$ (kg/m <sup>3</sup> )	OP	$p_c$ (atm)	$p_b$ (atm)	Type
$H_2$	801	-	0.0040	-	SOLID
$Al$	1780	-	0.0023	-	
$H_2$	80	-	0.0027	-	
$H_2/H_2O$	81	-	0.072	-	
$LiH_2$	100	-	0.000	-	
$H_2/H_2$	100	-	1.000	-	LIQUID
$H_2/H_2/H_2O$	177	1.31	1.000	1.440	
$H_2/H_2/H_2O$	303	0.70	1.000	1.404	
$H_2/H_2/H_2O$	277	0.70	1.000	1.382	
$H_2/H_2/H_2O$	270	1.10	1.000	1.369	

Table 1. Propellant parameters for some different propulsion systems.

Figure 1 compares the mass efficiency for the various systems as a function of the  $\Delta V$  mission requirement. It can be seen that at low  $\Delta V$  there is not much difference in mass efficiency between the systems. This is because the propulsion system is not a major component in the total mass of the vehicle and for such missions the propulsion unit should be selected on the basis of some criteria other than mass efficiency.

However, as the  $\Delta V$  requirement increases, the propulsion system mass begins to dominate, and differences in mass efficiency between available systems becomes extremely important and should be the determining factor in system selection.

#### SENSITIVITY ANALYSIS

In comparing the results in Figure 1 with the differences in propellant parameters in Table 1, it can be seen that the structure coefficient  $c$  can be more important than the propellant  $I_{sp}$ . The sensitivity of the vehicle mass efficiency to these parameters can be studied by taking the mathematical derivative of Eq (2) to obtain

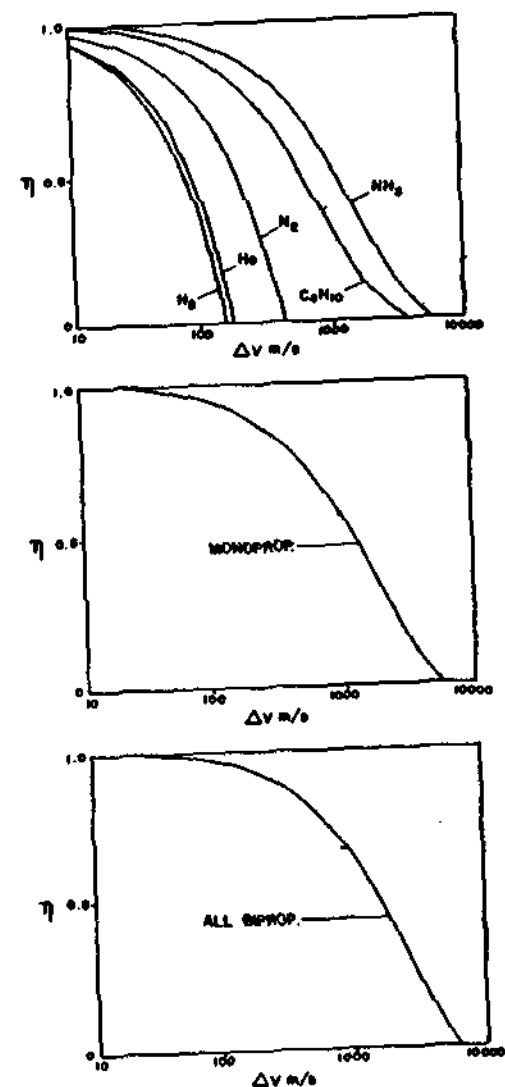


Figure 1. Comparison of the mass efficiency for the different propulsion systems of Table 1.

$$\frac{d\eta}{\eta} = \frac{c}{\eta} \left( \frac{\partial \eta}{\partial c} \right) \frac{d}{c} + \frac{I_{sp}}{\eta} \left( \frac{\partial \eta}{\partial I_{sp}} \right) \frac{d I_{sp}}{I_{sp}}$$

or

$$\frac{dn}{n} = x_c \frac{dc}{c} + x_{Isp} \frac{dIsp}{Isp} \quad (6)$$

Eq. (6) shows the sensitivity of the mass efficiency of some given base line design to changes in the parameters. The sensitivity coefficients are easily found to be

$$x_c = \frac{c}{c-1} \frac{\left( \frac{-\Delta V}{e^{g_0 Isp} - 1} \right)}{\left( \frac{-\Delta V}{c - e^{g_0 Isp}} \right)}$$

and

$$x_{Isp} = \frac{-\Delta V}{g_0 Isp} \frac{\frac{-\Delta V}{e^{g_0 Isp}}}{\left( \frac{-\Delta V}{c - e^{g_0 Isp}} \right)}$$

The ratio of these two coefficients is then

$$\frac{x_c}{x_{Isp}} = - \frac{g_0 Isp}{\Delta V} \left( \frac{c}{c-1} \right) \left( 1 - e^{\frac{-\Delta V}{g_0 Isp}} \right)$$

which is shown in Figure 2.

It is worth noting that the scale is in  $\log_{10}$  and that at small values of  $\frac{g_0 Isp}{\Delta V}$  the  $x_c$  is some 400 orders of magnitude greater than  $x_{Isp}$ . As  $\frac{g_0 Isp}{\Delta V}$  increases the two coefficients rapidly approach a common value, i.e., their ratio becomes 1.

It is easy to see that the idea of sensitivity coefficients can be extended to the parameters that make up  $c$ . For example, the sensitivity of a given design to a change in structural material could be studied by forming

$$x_\sigma = \frac{\sigma}{n} \frac{\partial n}{\partial \sigma} = \frac{\sigma}{n} \frac{\partial n}{\partial c} \frac{\partial c}{\partial \sigma} = \frac{\sigma}{c} x_c \left( \frac{\partial c}{\partial \sigma} \right)$$

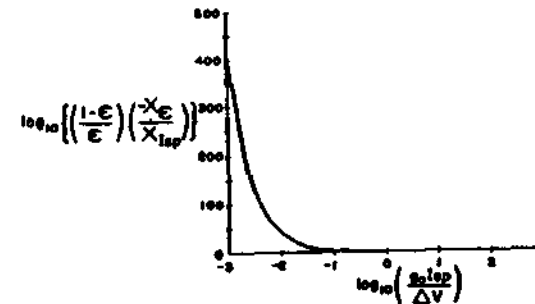


Figure 2. Comparison of the sensitivity coefficients in terms of orders of magnitude.

or for a change in propellant density

$$x_\rho = \frac{\rho}{c} x_c \left( \frac{\partial c}{\partial \rho} \right)$$

and so forth.

## CONCLUSION

In closing we point out that the methodology presented here includes all of the pertinent parameters of a propulsion system such as the operating pressure, structural material, and propellant characteristics in one simple expression that can be mathematically manipulated. The formulation can be incremented if desired to include such things as interactions between  $Isp$  and  $c$  through the adoption of different nozzles, combustion chambers, injections systems, and the like.