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Comparison of Separate and Integral Spacecraft

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Nomenclature

C = cost per flight
c = unit cost
g = acceleration of gravity
I = specific impulse
R = (refurbishment plus amortized replacement cost)/unit cost
v = ideal velocity
w = weight
λ = stage structure factor (empty weight less payload/gross weight less payload)

Subscripts

g = gross
i = integral
p = throw payload
s = separate
u = useful payload

Introduction

EXPANSION of manned space activity can be expected to bring about growing requirements for manned spacecraft capable of performing such missions as logistic shuttle, rescue, reconnaissance and inspection, more flexibility and economically than is possible with current available or funded systems. This has been recognized for some time during which many vehicle concepts have been proposed and evaluated. It has come to be generally agreed that such a spacecraft should be inherently capable of aerodynamic maneuvering by being configured for hypersonic lift-to-drag ratios not less than ~1.5 to provide flexibility in landing site selection, to provide horizontal landing capability, and to reduce thermal shield temperatures during re-entry. It is not so generally agreed, however, whether the spacecraft should provide its own capability for space maneuvering including participation in the launch ascent, orbital change, and rendezvous, in the form of an integral propulsion system, or whether such propulsive capability should be provided by a separate expendable stage.

If total "throw" payload is used as the criterion, as is frequently the case because experience to date is predominantly with nonrecoverable and ballistic systems where total "throw" payload is a reasonable criterion, then the separate vehicle concept often appears preferable because of the favorably low structure factor associated with expendable stages. If, however, "useful" payload is used as a criterion, (and this may be a more direct measure of mission capability where much of the "throw" payload must be allocated to aerodynamic lifting and thermal structure), then under many conditions, the integral vehicle concept may be preferable, both from a weight and cost viewpoint. This will be demonstrated by a parametric performance and cost comparison of representative separate and integral spacecraft concepts.

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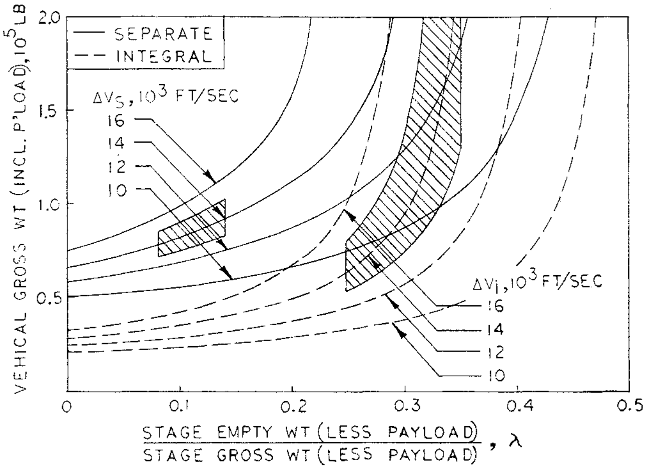

Fig. 1 Gross weight vs structure factor (I_s = I_i = 450 sec).

Separate and Integral Vehicle Concepts

The separate vehicle is assumed to consist of a lifting body and an expendable propulsive stage which provides velocity gains necessary for major maneuvers such as optimized participation in launch ascent, orbital change, rendezvous and deorbit. The integral vehicle is assumed to consist of a lifting body containing an integral propulsive capability for maneuvers such as listed above. In general, because of the lower re-entry platform loading of the empty integral spacecraft, that vehicle can utilize radiative thermal shielding, while the nonpropulsive re-entry element of the separate spacecraft would be expected to require an ablative shield.

Since more design data are available for separate spacecraft, that case will be used as the reference in establishing a standard payload applicable to a spectrum of manned near orbit missions. Studies have shown that a wide variety of manned mission requirements can be met with a throw weight in the order of the Titan-IIIC near orbit payload capability of 25,000 lb. Therefore, it will be assumed that w_p = 25,000 lb.

Useful payload will be defined as including cargo, crew, life support and an appropriate allocation of power and other accessory subsystems. Design data from various sources indicate good general agreement that for a nonpropulsive, medium L/D lifting body having a gross weight of 25,000 lb., the useful payload as here defined will be in the vicinity of 11,000 lb. Thus, assume w_u = 11,000 lb.

With regard to reasonable values of structure factor, design studies to date seem to justify the following assumed ranges for hydrogen-oxygen propelled vehicles in the size ranges considered: $0.08 \leq \lambda_s \leq 0.14$ and $0.25 \leq \lambda_i \leq 0.35$.

The velocity gain required of each vehicle concept depends on the mission. All missions for this type of spacecraft require launch, however, and except where extensive orbital maneuvers are involved it seems reasonable to assume that almost all the required velocity will be that associated with the latter portions of the ascent to orbit. Launch trajectory optimization studies have shown that the ideal velocity requirements for many cases are about equal for both the separate and integral cases, in the range of 13,000–15,000 fps, which will be the assumed range for both Δv_s and Δv_i.

Thus, the separate and integral concepts can be compared on an essentially equal performance basis without reference to preceding propulsive stages (assuming of course, that

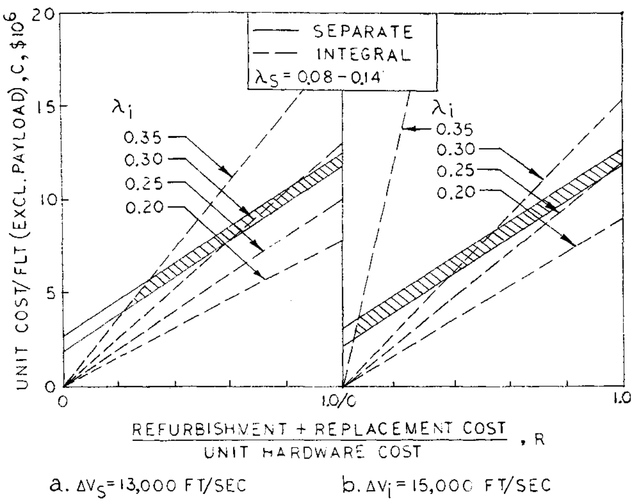

Fig. 2 Unit cost per flight vs refurbishment-replacement factor ($I_s = I_i = 450$ sec).

their gross weights are about equal, which as shown later they are, for many cases of interest). Both separate and integral concepts can therefore be assumed to have equivalent preceding stages, whether they be of recoverable, expendable, drop tank or other variety.

Performance Comparison

The performance of the two vehicle concepts can be described using the ideal rocket equation.

$$\Delta v_s = gI_s \ln[(w_p + w_s)/(w_p + \lambda w_s)] \quad (1)$$

$$\Delta v_i = gI_i \ln[(w_u + w_i)/(w_u + \lambda w_i)] \quad (1a)$$

Since the gross weights of the vehicles are $w_{gs} = w_p + w_s$ and $w_{gi} = w_u + w_i$, it can be shown by substitution in Eqs. (1) and (1a) that

$$w_{gs} = w_p(\lambda_s - 1)e^{\Delta v_s/gI_s}/(\lambda_s e^{\Delta v_s/gI_s} - 1) \quad (2)$$

$$w_{gi} = w_u(\lambda_i - 1)e^{\Delta v_i/gI_i}/(\lambda_i e^{\Delta v_i/gI_i} - 1) \quad (2a)$$

Equations (2) and (2a) are plotted in Fig. 1 for various values of Δv and λ, and for $I_s = I_i = 450$ sec to represent high-pressure hydrogen-oxygen propulsion. Typical regions of interest are shown by the shaded areas. The most prominent indication of Fig. 2 is the strong sensitivity of the integral concept, to uncertainties in structure factor λ_i. If $\lambda_i <$ about 0.3 can be achieved, the integral concept is competitive with or preferable to the separate concept on a gross weight basis; but if λ_i much exceeds the vicinity of 0.3, then the integral concept quickly becomes not only relatively heavy, but progressively even more sensitive to λ_i.

Cost Comparison

The flight hardware cost per flight for the two vehicles can be written

$$C_{p+s} = c_pR + c_s \quad (3)$$

$$C_i = c_iR \quad (4)$$

where c_p does not include useful payload cost and neither c_s nor c_i include propellant costs, which in any case are negligible compared to hardware costs.

The hardware costs were developed using cost estimating relationships involving materials cost and manufacturing hours per pound, air frame dry weight, and expended propellant weight for the expendable stage, and involving historical and point design vehicle costs and manufactured hardware weight of ablative and radiative manned spacecraft for the reusable vehicles. Using these estimated costs, Eqs. (3) and (4) are plotted in Fig. 2a and 2b for ideal velocity gains of 13,000 and 15,000 fps, respectively. Because of the uncertainties associated with them, structure factors and refurbishment-replacement factors are varied parametrically.

These results indicate that if a λ_i of 0.25 can be achieved, then the integral spacecraft is more economical in essentially all cases considered. If $\lambda_i = 0.30$, the integral concept is more economical for values of R less than 39 to 84%. Even if λ_i is as high as 0.35, the integral concept is more economical for values of R less than 15 to 32%.

Observations

1) Integral spacecraft are indicated to be competitive with, or preferable to, separate spacecraft in both cost and gross weight for a considerable range of structure factors and refurbishment-replacement factors, for velocity gains of 13,000 to 15,000 fps. At lower velocity gains, separate spacecraft may be at a greater advantage because of structure factor penalties to the integral concept which result inevitably from smaller vehicle sizes, and from the fixed structural weights needed to enclose and carry the payload. At higher velocity gains, integral spacecraft probably have the advantage because of the more favorable structure factors accompanying larger sizes.

2) Because of the performance sensitivity of the integral concept to structural dead weight, it appears mandatory that the structure factor as defined here be held to the vicinity of 0.30 or less. This emphasizes the importance of improvements in the technologies of high volumetric efficiency lifting bodies and lightweight structures and radiative thermal shields.

3) Aside from cost and performance advantages, integral manned spacecraft appear to offer increased reliability by eliminating staging and increased flexibility from more favorable cross range, landing characteristics and refurbishment (all by virtue of decreased planform loading during and after entry).

4) Although integral spacecraft may involve somewhat higher research and development costs than separate concepts, and their characteristics are more sensitive to structure factor, their potential advantages appear to render integral spacecraft concepts worthy of increased interest and study.

Effects of Oblique Shock Waves in the Near Field of Rocket Plumes

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IN Ref. 1 we described a technique for laboratory simulation of rocket plumes in space. Some sample results showed that stagnation-point heating along the thrust axis was higher

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