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PROPULSION AND STAGING CONSIDERATIONS FOR AN ORBITAL SORTIE VEHICLE*

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Abstract—Severe weight penalties can occur if the sizes of Orbital Sortie Vehicle stages are not near their optimum values. In particular, single-stage rocket configurations with or without "zero-stage" boosters are necessarily much larger than two-stage configurations. Conventional cryogenic propellant combinations provide a reasonable range of vehicle sizes. By comparison, one storable-liquid propellant combination would require vehicle size to be excessive, while two "exotic" propellant combinations do not appear to provide enough size reduction to warrant their use. Second-stage structural efficiency has a major effect on overall vehicle size. It appears feasible to design a two-stage Orbital Sortie Vehicle whose gross lift-off weight is 50–70 times the payload weight.

1. TECHNICAL DISCUSSION

Sensitivities of vehicle proportions and weights have been evaluated at a trajectory design point that is loosely representative of requirements for an Orbital Sortie Vehicle (OSV), i.e. a small, manned launch vehicle capable of achieving low earth orbit and returning to earth via aircraft-like landing. It is assumed that a total ideal velocity of 33,000 ft/s is to be achieved by a two-stage vehicle. The stages are mounted serially rather than in parallel (as for the space Shuttle). This ideal velocity is a conservative estimate of that which is required to achieve a low-altitude polar orbit, accounting for gravity and drag losses and neglecting possibly beneficial lifting effects. Therefore, this preliminary analysis is a better approximation to vertically launched all-rocket systems than to horizontally launched, air-breathing, lifting systems.

The total ideal velocity, V_I, is computed as the sum of each stage's contribution,

$$\begin{aligned} V_I &= V_I(1) + V_I(2) \\ &= C(1)\ln MR(1) + C(2)\ln MR(2)\ \text{ft/s} \qquad (1) \end{aligned}$$

where $C(1)$ and $C(2)$ are the *exhaust velocities* and $MR(1)$ and $MR(2)$ are the *mass ratios* of the two stages. These are related to the respective *specific impulses*, I_{sp}, by,

$$C = g\, I_{sp} = 32.2\, I_{sp}\ \text{ft/s} \qquad (2)$$

where g is the acceleration due to gravity and I_{sp} is measured in seconds.

The mass ratios, $MR(1)$ and $MR(2)$, depend on the initial and empty weights of each stage:

$$MR = \text{Initial Weight/Empty Weight} = W_I/W_E \qquad (3)$$

(The initial and final weights of the first stage include the initial weight of the second stage.) These equations model the single-stage alternative when the first stage's mass ratio is one, contributing zero velocity increment.

It is assumed that the vehicle's net payload is 7000 lb. If this figure is interpreted as including 2000 lb for a crew and their life-support systems, 5000 lb is left for equipment. The *payload ratio*, L, can be defined as,

$$L = \text{Payload Weight/Initial Weight} = W_P/W_I \qquad (4)$$

where W_I represents either the overall or stage initial weight, as appropriate. Similarly, the *structural weight* ratio, E, is defined as,

$$E = \text{Structural Weight/Initial Weight} = W_S/W_I \qquad (5)$$

By convention, W_S is taken to include weights of the engine, wings, landing gear, control systems, unburned propellants and auxiliary equipment, as well as the airframe itself. Hence, it is not due to the structure alone and might better be called "Inert Weight".

There is an interrelationship among MR, L and E; for this analysis, the most useful expression of this is,

$$L = 1/MR - E. \qquad (6)$$

Rearranging (1) allows $MR(1)$ to be expressed in terms of $MR(2)$:

$$MR(1) = \exp\{[V_I - C(2)\ln MR(2)]/C(1)\}. \qquad (7)$$

Given the total ideal velocity, the specific impulses,

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the structural weight ratios, $MR(2)$, and the payload weight, it then is possible to compute $MR(1)$, the ideal velocity increments of each stage and all the initial and empty weights. Furthermore, an iteration can be set up to find the value of $MR(2)$ that minimizes $W_I(1)$, which is the *gross lift-off weight*, or GLOW.

This procedure has been carried out for a range of structural ratios [$E(1)$ and $E(2) = 0.1$ to 0.2] and for five propellant combinations:

(I) Oxygen:Hydrogen (both stages)
(II) Oxygen:RP-1/Oxygen:Hydrogen
(III) Oxygen:Hydrogen/Fluorine:Hydrogen
(IV) Oxygen:Hydrogen/Oxygen:Hydrogen-Beryllium
(V) Hydrazine-UDMH:Nitrogen Tetroxide (both stages)

I to IV are cryogenic propellants, III and IV could be considered "exotic" due to high reactivity and toxicity, and V is the storable combination used in the Titan II vehicle. The specific impulses for these propellants are taken from [1]. It is assumed that the first-stage rocket provides optimum expansion from 100 psia chamber pressure to standard ambient, while the second-stage rocket provides expansion from 250 psia to vacuum (the chamber pressures are somewhat lower than can be obtained currently; hence, the estimated I_{sp} is conservative.) This provides the combinations given in Table 1.

Four sets of results were obtained and are discussed below. The first deals with the effect of ideal staging velocity on OSV weights. The second illustrates the effect of structural weight ratios on optimum stage sizes. The third shows the effect of propellant combinations on optimum stage sizes. The fourth treats the effect of aerodynamic efficiency and thrust-weight ratio for air-breathing propulsion. In addition, there is a brief discussion of propulsion requirements for on-orbiting maneuvering.

2. IDEAL STAGING VELOCITY EFFECT

The relative sizes and propellant loadings of the vehicle's two stages have direct effect on the ideal velocity at which the first stage burns out and the second begins to thrust. Conversely, it is possible to plot the gross lift-off weight as a function of $V_I(1)$. Figure 1 presents this relationship for Case II, assuming $E(1) = 0.15$ and letting $E(2)$ be either 0.10 or 0.15. For the structurally less-efficient second stage

Table 1.

Case	$I_{sp}(1)$	$I_{sp}(2)$, s
I	390	455
II	300	455
III	390	475
IV	390	536
V	290	340

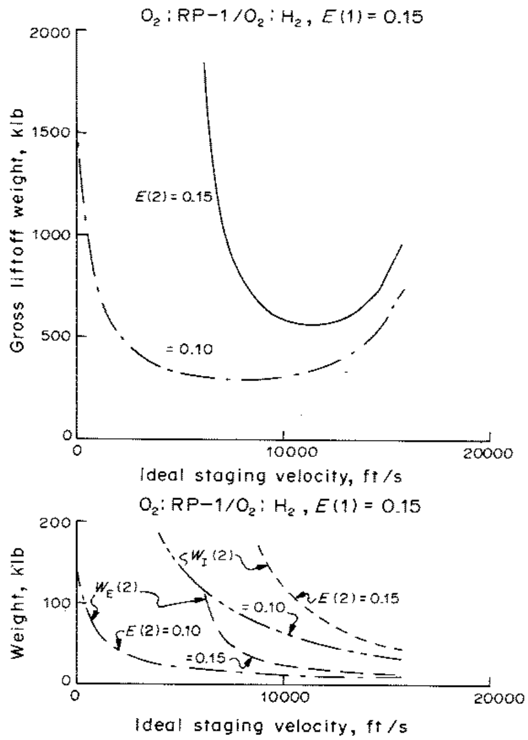

Fig. 1 Effect of ideal staging velocity on vehicle weights for O_2:RP-1 first stage and O_2:H_2 second stage.

[$E(2) = 0.15$], the optimum staging velocity is 10,700 ft/s, which corresponds to a Mach number of about 11. GLOW grows to very large values at non-optimum $V_I(1)$, and there are no solutions for $V_I(1)$ less than 600 ft/s because $L(2)$ goes to zero.

Improving second-stage initial and empty weights of the second stage illustrate the severe penalty paid for low staging velocity. The GLOW of 250,000 to 600,000 lb and second stage empty weights of 15,000 to 30,000 lb set standards against which to judge candidate configurations. The principal significance of $W_I(2)$ is in the sizing of the main engine(s), while $W_E(2)$ has an important effect on the sizing of the wing and landing gear. Staging velocities above 10,000 ft/s call for a maximum thrust in the range of 200,000 lb. The empty weights of such designs would be in the range of current fighter aircraft.

The high lift-off weight of a single-stage OSV with a "zero-stage" booster, i.e. one which provides an initial impulse of a few hundred feet per second or less, is apparent. Furthermore, the wide variations of GLOW at low staging velocity indicate undesirable sensitivity to the inevitable uncertainties of development and production. The generally negative conclusions to be drawn apply principally to vertically launched configurations. The function of the booster in horizontally launched configurations is not so much to supply velocity increment as to help the vehicle get airborne for airplane-like flight.

3. PARALLEL-STAGING EFFECTS

Parallel staging, in which the second stage is thrusting while the first stage is thrusting complicates the analysis because there is no clear-cut demarcation between the contributions that each stage makes to the velocity increment during the first-stage burn. Furthermore, both first and second-stage engines could draw their propellants from the first stage tanks in one design, while in another each might draw from its own tanks. Even if both stages use the same propellants, their specific impulses may be different due to differences in engines. (For example both stages could us Space Shuttle Main Engines with different exhaust expansion ratios.)

Nevertheless, approximations could be made once the parallel staging approach is identified. With engines of both stages using only first stage fuel, an average I_{sp} could be calculated in the ratio of the total thrust due to each stage's engines, and the analysis would proceed as before. A lower "structural" weight fraction could be attributed to the first stage, as the second stage would carry engines that otherwise would contribute to the first stage's inert weight. With each stage using its own propellant (as in the Space Shuttle), the same averaging of I_{sp} must be done, but sizing studies would require further problem definition, e.g. that some predetermined percentage of the first-stage velocity increment would be generated by the second stage. Once the first stage had been dropped, the second stage would begin to thrust alone with partially empty tanks. This would be equivalent to carrying a higher structural weight fraction than could have been obtained with serial staging.

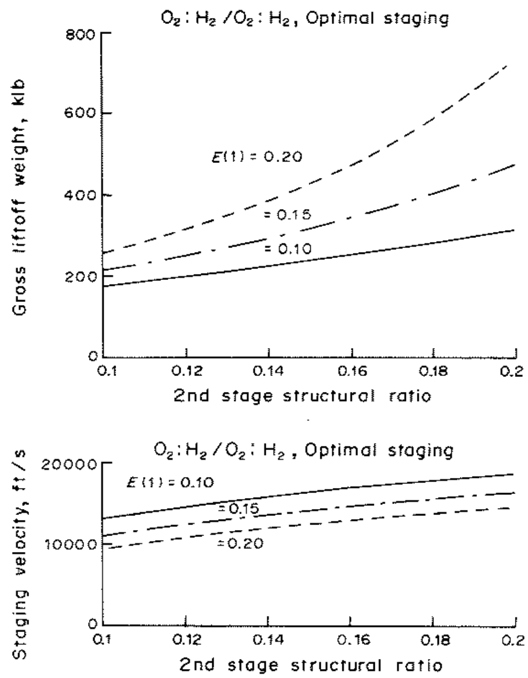

Fig. 2. Optimal gross liftoff weight and staging velocity for two $O_2:H_2$ stages.

4. STRUCTURAL-WEIGHT-RATIO EFFECT

A more detailed examination of the effects of $E(1)$ and $E(2)$ is provided by Figs 2 and 3. Every point on these figures represents an optimum staging condition, i.e. each is equivalent to the minimum of a GLOW curve, such as shown in Fig. 1. These particular figures present Case I, although similar numerical results have been obtained for all five cases. Expected trends can be seen in the figures: increased structural weight ratio leads to increased initial and empty weights. Second-stage structural efficiency tends to be more important than first-stage efficiency because second-stage I_{sp} is higher.

While low structural ratios can be achieved in conventional expendable launch vehicles, the addition of wings and landing gear *to both stages* will raise $E(1)$ and $E(2)$. The range of structural weight fractions considered here (0.1 to 0.2) is rather optimistic, as comparison with the Space Shuttle and high-performance aircraft would suggest that the range be extended to about 0.3. Assuming that the first stage is unmanned but flyable on return, its percentage increase may be less because its maximum reentry velocity and heating are lower and higher "*g*" loads may be tolerated; nevertheless, landing speeds must be consistent with anticipated air traffic environments, as well as with automatic or remotely piloted operation.

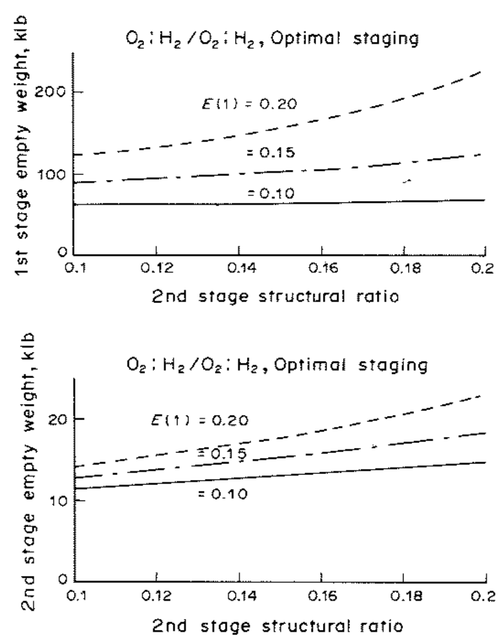

Fig. 3. Optimal empty weight for two $O_2:H_2$ stages.

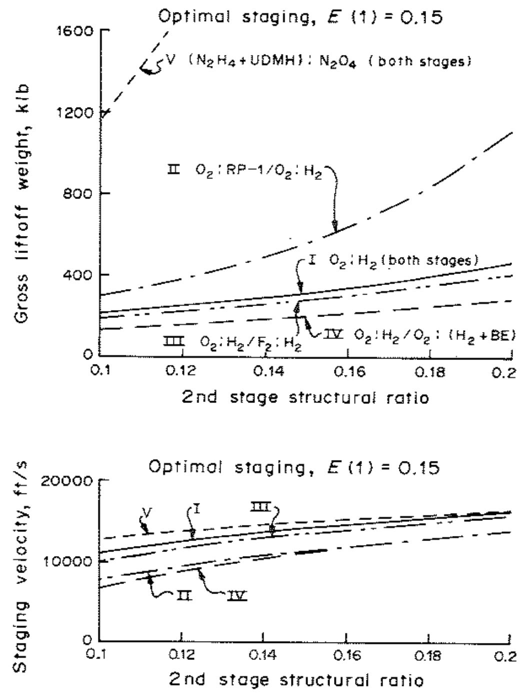

Fig. 4. Optimal gross liftoff weight and staging velocity for five propellant combinations.

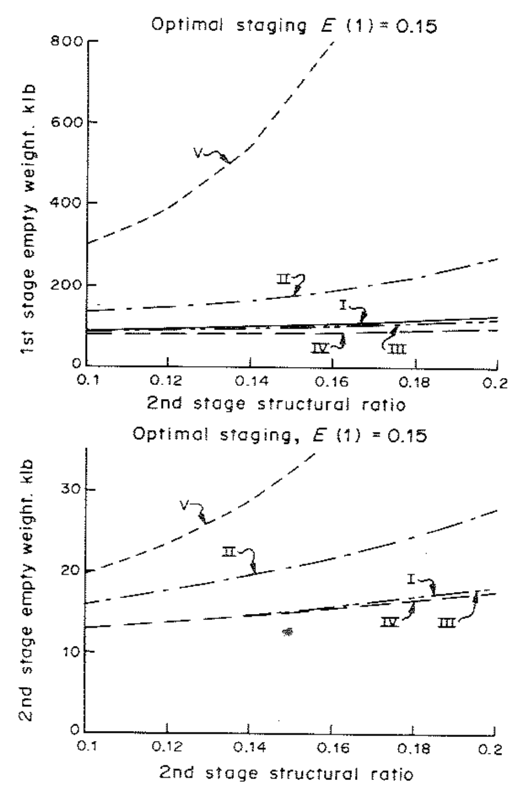

Fig. 5. Optimal empty weight for five propellant combinations.

5. PROPELLANT-COMBINATION EFFECT

The "energetics" of the propellant combinations have a significant effect on OSV sizing. Figures 4 and 5 present additional optimum staging results, and they illustrate that the OSV concept is dependent on second-stage I_{sp} in the vicinity of 450 or more for overall payload ratios exceeding 1/100. All of these examples use cryogenic fuels, possibly limiting the "on-demand" nature of OSV launch; however, it is clear that the all-storable alternative (Case V) is at least three to four times heavier than these options. A hybrid alternative, using storables in the first stage and cryogenics in the second stage could be considered as the storable-propellant I_{sp} is just 3% below that of oxygen and RP-1. It does not appear that the 4% I_{sp} advantage of fluorine over hydrogen warrants the increased hazard connected with its use. The 18% I_{sp} gain afforded by adding beryllium to the hydrogen is attractive, although practical considerations could preclude this approach.

Staging velocities for all combinations are in the 10,000-ft/s neighborhood, generally reflecting the ratios of first- to second-stage I_{sp} (Fig. 3). Overall propulsive efficiency has a predictable effect on first- and second-stage empty weights. Case V is somewhat heavier than the others, and there is close grouping of the weights for the three highest performing propellants. The use of oxygen with RP-1 or hydrazine with nitrogen tetroxide for the first stage causes a significant (but probably acceptable) weight increase.

There are many additional factors to be considered in the selection of propellant combinations, and two of these are noted here. The use of common propellants in both stages is desirable for parallel-burn configurations, as it allows the porting of first-stage propellants to the second stage. The present analysis does not explicitly express *bulk-density effects* on structural efficiency. Higher structural efficiency, i.e. lower structural weight ratio, normally would be associated with high-density propellants, e.g. RP-1 and the storables. The relative difference between Cases I and II would be reduced if this factor were considered.

6. EFFECTIVE SPECIFIC IMPULSE OF AIR-BREATHING ENGINES

While it is clear that air-breathing engines provide substantially greater I_{sp} than rocket engines (1000–4000 s vs 300–500 s), their apparent advantage is diminished by the increased *airframe drag penalty* that must be paid to realize these gains. Rocket acceleration can occur under essentially drag-free conditions, but the vehicle must stay within the atmosphere to accelerate with air-breathing propulsion. Even if the thrust-minus-drag characteristics of the engine alone are good, the net acceleration of the total vehicle must be considered.

The extent of performance reduction due to acceleration within the atmosphere can be predicted by calculating an *equivalent* I_{sp} that accounts for airframe efficiency (L/D) and thrust-to-weight ratio (T/W). Whereas I_{sp} usually is calculated as,

$$I_{sp} = \text{Thrust}/\dot{m}g \tag{8}$$

where $\dot{m}$ is the rate-of-flow of *on-board* propellant, the effective I_{sp} is

$$\begin{aligned} EI_{sp} &= (\text{Thrust} - \text{Drag})/\dot{m}g \\ &= \text{Thrust}/\dot{m}g - \text{Drag}/\dot{m}g \\ &= I_{sp}(1 - \text{Drag}/\text{Thrust}) \end{aligned} \tag{9}$$

If the vehicle is in level flight during its acceleration, then lift equals weight, and

$$\text{Drag} = \text{Weight}/(\text{Lift}/\text{Drag}) = \text{Weight}/(L/D) \tag{10}$$

Consequently, for flight in the atmosphere,

$$\begin{aligned} EI_{sp} &= I_{sp}[1 - (\text{Weight}/\text{Thrust})/(L/D)] \\ &= I_{sp}\left[1 - \frac{1}{(T/W)(L/D)}\right], \end{aligned} \tag{11}$$

Clearly, EI_{sp} is close to I_{sp} only if T/W and L/D are both relatively large numbers. With $T/W = 2$ and $L/D = 4$, EI_{sp} is 12.5% smaller than I_{sp}, and examples of other combinations are shown in Fig. 6. Of course, it is possible to accelerate in level flight with T/W less than one; however, EI_{sp} goes to zero when $T/W = D/L$. Propulsion efficiency is seen to depend on airframe efficiency and installed thrust-to-weight ratio, as well as on the characteristics of the engine.

7. ON-ORBIT MANEUVERING

In the current context, the weight of an orbital maneuvering system (OMS) must be considered as payload, as it does not contribute to orbital insertion and it does not have a major effect on structural weight. Preserving the 7000-lb payload that has been assumed to this point results in an effective payload-weight growth that ripples down to increase all the other weights.

Assuming that the mass fraction dedicated to orbital maneuvering is small compared to second-stage empty weight, $W_E(2)$, the added weight of OMS propellant, W_O, is approximately,

$$W_O = W_E(2)\, V_M/gI_{sp}(\text{OMS}) \tag{12}$$

where V_M is the on-orbit velocity increment (in ft/s) needed for maneuvering and $I_{sp}(\text{OMS})$ is the specific impulse of the OMS. Because $W_E(2)$ is the sum of W_P and $W_S(2)$, W_O can be expressed in terms of $L(2)$ and $E(2)$, as well as the payload weight:

$$W_O = W_P[1 + E(2)/L(2)]\, V_M/gI_{sp}(\text{OMS}) \tag{13}$$

The effective payload weight then is $(W_P + W_O)$.

A simple example illustrates the weight penalties that must be accepted for orbital maneuvering capability. First, it is assumed that the OMS engine and tank weights are included in $W_S(2)$. A one-deg plane change is to be made at an orbital velocity of 25,700 ft/s. Using the storable propellants of Case V, with vacuum $I_{sp} = 340$ s, and assuming that $E(2)$ and $L(2)$ are equal, the plane change requires a velocity increment of 448 ft/s, which translates to a propellant weight that is 8% of W_P; therefore, all the weights of the previous analyses must be increased by the same amount to achieve this capability. A 5-deg plane-change capability implies a vehicle growth of about 40%. For large plane-change capability, "synergistic" maneuvers involving aerodynamic lift at low altitude save weight, although increased thermal protection requirements must be considered.

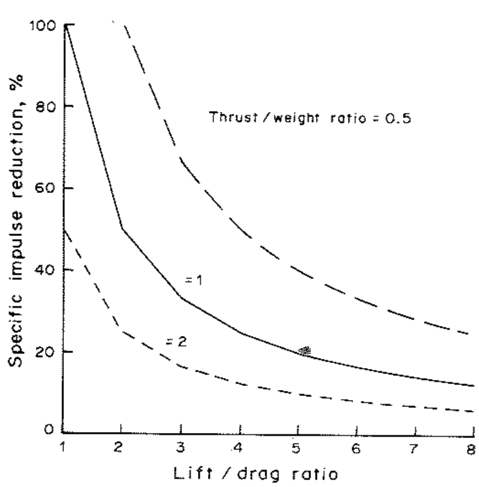

Fig. 6. Effect of aerodynamic efficiency and thrust-to-weight ratio on realizable specific impulses of an air-breathing stage.

8. CONCLUSIONS

Two-stage alternatives for an Orbital Sortie Vehicle have been considered here, and several logical paths for development emerge. By designing for near-optimum staging ratios, the sensitivity to weight growth during development and production is reduced. There does not appear to be a significant advantage in using "exotic" propellants, and a commonly used all-storable propellant alternative has insufficient performance for the assumed flight profile. The leading contenders are oxygen: hydrogen in both stages or oxygen:RP-1 in the first stage and oxygen:hydrogen in the second stage. Extensive orbital-maneuvering capability exacts a heavy penalty in GLOW and empty weights. For 7000-lb payload (including 2000 lb for the crew) lift-off weights on the order of 500,000 lb appear to be feasible. Optimal staging velocities of about

10,000 ft/s are indicated. Single-stage vertical-launch alternatives would be substantially heavier than staged configurations. The present analysis does not model atmospheric effects in any detail; however, preliminary analysis suggests that the high specific impulses produced by air-breathing propulsion would be tempered by the need for high aerodynamic efficiency and thrust-to-weight ratio.

REFERENCE

1. Theoretical Performance of Rocket Propellant Combinations, prepared by Rocketdyne Propellant Performance Unit, Canoga Park, undated.