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-- NOTES --

COMPARISON OF PARALLEL AND TANDEM STAGING OF NUCLEAR ORBITAL LAUNCH STAGES FOR INTERPLANETARY DEPARTURE

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Abstract

A performance comparison between two configurations of spacecraft for interplanetary departure is presented. The two configurations have three propulsion stages arranged in tandem or in parallel. With the stages in parallel, performance is evaluated with three stages used simultaneously and with only two stages used first, followed by the third. Multiorbit injection is used. The comparison indicates that the tandem-configured vehicle will perform better than the parallel-configured vehicle. The characteristics of multiorbit injection are discussed. This study indicates that parallel assembly of nuclear stages does not result necessarily in a performance gain if multiorbit injection is used.

Introduction

When interplanetary missions are being considered, the flight plan that maximizes performance of orbital-launch stages depends on both the stage and the flight-plan characteristics. A study(1) was conducted of the interaction between flight-plan characteristics and some solid-core nuclear-propulsion-stage characteristics. The results of this study indicate that the use of several engines per stage to increase the thrust level was not as effective in increasing the performance as a change from a single escape burn to a multiple thrusting arc, multiorbit-injection technique. However, this study was restricted to a tandem stage arrangement for the orbital-launch configuration. In some studies,(2,3) a parallel stage arrangement has been proposed that would be a better configuration for the orbital-launch stages because this arrangement increases the thrust-to-weight ratio for the escape maneuver. The comparison between the performance of the parallel and tandem arrangements for the orbital-launch stages(2,3) is based on a single injection burn only (direct injection).

Performance is not the only important factor in the design of an orbital-launch stage. The elimination of major technical developments and the associated savings in cost are also important considerations when the configuration of an interplanetary vehicle is being determined. Parallel-staging development may be a significant cost factor in the orbital-launch-stage development. This study was made to determine whether a performance justification exists for development of parallel staging for orbital-launch stages.

Analysis

One objective of configuring orbital-launch stages in parallel is to increase the thrust-to-weight ratio. When the thrust-to-weight ratio is increased, greater efficiency is possible because of the increased acceleration, which reduces the thrusting arc when the escape maneuver consists of a single burn. However, when the proper injection technique (a multiburn escape maneuver) is chosen, a vehicle that has one low-thrust engine per stage would perform better than a vehicle that has either two such engines or one higher thrust engine per stage.(1) The multiorbit injection, which was used for this study, is a generalized escape maneuver that includes the direct injection as a special case. Multiorbit injection consists of a series of thrusting arcs near perigee separated by elliptical orbits (Fig. 1). The number and size of the orbits and the steering direction can be varied to optimize the performance of a specific orbital-launch configuration. The method used to analyze vehicle performance with multiorbit injection is outlined in Reference 4. The period of the final intermediate orbit was constrained arbitrarily to a maximum of 48 hours.

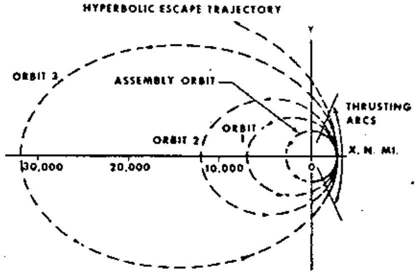

Figure 1. Multiorbit-injection profile for three orbits to escape.

Vehicle performance will be presented in terms of a velocity change defined as the difference between the initial perigee velocity and an equivalent final perigee velocity. Equivalent perigee velocity is the perigee velocity of an orbit that has the same perigee radius as the initial orbit and the same hyperbolic excess velocity as the final orbit. The equivalent perigee velocity is derived from Equation (1).

$$V_\infty^2 = V_p^2 - \frac{2\mu}{r_p} \tag{1}$$

where V_∞ is hyperbolic excess velocity, V_p is perigee velocity, r_p is perigee radius, and μ is the gravitational constant of the Earth. Equivalent perigee velocity is given by Equation (2).

$$V_{p,e} = \sqrt{V_{p,f}^2 - \frac{2\mu}{r_{p,f}} + \frac{2\mu}{r_{p,i}}} \tag{2}$$

where the subscripts e, i, and f indicate equivalent, initial, and final, respectively.

The spacecraft was assumed to have been assembled in low elliptical Earth orbit from six modules, each module approximately equal to the weight of one payload of an Earth-launch vehicle. Two configurations of interplanetary spacecraft were studied. In each configuration, three of the modules were assumed to be NERVA I class nuclear-rocket-engine-powered *orbital-launch stages (NOLS). Each NOLS was assumed* to have a weight equal to 20 percent of fuel weight for tank structure and 27.5 percent of the engine thrust for engine, thrust structure, and interstage weights. For the first configuration, the three NOLS were arranged in parallel; for the second configuration, the NOLS were arranged in tandem, as indicated schematically in Figure 2. The module weights that were considered were 300 000, 375 000, 425 000, and 550 000 pounds. *The NOLS empty weights* were 67 187, 79 687, 88 020, and 108 854 pounds, respectively.

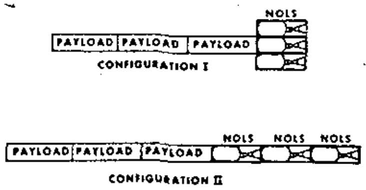

Figure 2. Vehicle configurations.

For the parallel configuration, two operational techniques were considered. The first technique was simultaneous use of all three NOLS (parallel staged). The second technique was to use the two outer NOLS until propellant depletion, stage those NOLS, and then use the third stage (2/3 staged).

The initial thrust-to-weight ratios for the vehicles described previously are given in Table 1.

TABLE 1. INITIAL THRUST-TO-WEIGHT RATIOS

Weight per stage, lb	Tandem staged	2/3 staged	Parallel staged
300 000	0.0416	0.0833	0.1250
375 000	.0333	.0667	.1000
425 000	.0295	.0589	.0882
550 000	.0237	.0455	.0681

Engine startup-ramp, shutdown-ramp, and cooldown propellant requirements were deleted at the end of each engine operation. Ramp and cooldown propellant *requirements were assumed to be 1500 pounds per engine* start plus 145 pounds per minute of engine operation time. These figures, which are based on unpublished data, are simplified approximations to the propellant requirement. The thrust that was produced during startup, shutdown, and cooldown was neglected. No limit was placed on total operation time per engine.

Results and Discussion

The performance results of this study are presented in Figures 3 to 6. The ΔV shown in the *figures is the velocity difference defined* previously. The characteristic velocities (V_c) associated with each of the cases considered are included on the ordinates of the graphs to show the propulsion and cooldown ΔV losses. An advantage of tandem staging is that the tandem-staged vehicle has a higher characteristic velocity than either of the characteristic velocities of the parallel-staged vehicle. The empty weight of the stages must be carried to higher velocities (e.g., all three stages are carried to escape if all three are used simultaneously); therefore, more propellant is required to reach those velocities. Before being used, the cooldown propellant also must be carried to higher velocities with the stage. The data presented in Figures 3 to 6 indicate that, for all *cases, the tandem configuration will perform better* than the parallel configuration if enough thrusting arcs are used. For the 300 000-pound-stage case (Fig. 3), performance advantages [illegible]dicated of 200 fps over the 2/3-staged vehicle and 600 fps over the parallel-staged vehicle. Performance of the 300 000-pound tandem-staged vehicle is only 300 fps lower than the characteristic velocity of the 300 000-pound parallel-staged vehicle. A technique probably could not be devised to limit propulsive and cooldown losses of the parallel-staged vehicle to less than 300 fps to make such a vehicle competitive with the tandem-staged vehicle. As the weight per stage increases, the performance advantages of the tandem configuration decrease to only 100 fps for the 550 000-pound-stage case. The decrease in advantage can be attributed to the increased significance of propulsive losses. The losses are more significant because of the low acceleration that results from low thrust-to-weight ratio (e.g., initial ratio of 0.024 for the 550 000-pound tandem configuration).

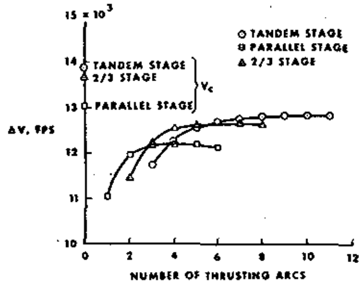

Figure 3. Performance of vehicle with 300 000 pounds per stage.

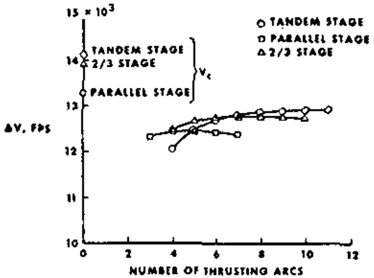

Figure 4. Performance of vehicle with 375 000 pounds per stage.

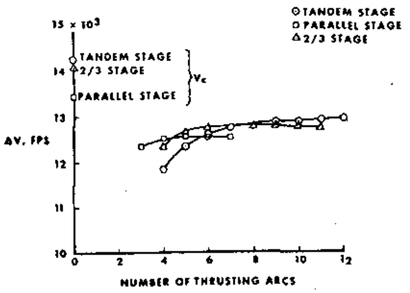

Figure 5. Performance of vehicle with 425 000 pounds per stage.

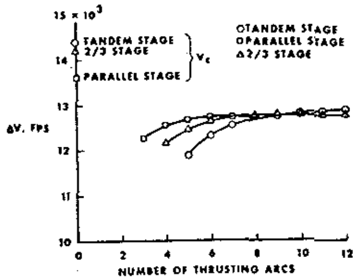

Figure 6. Performance of vehicle with 550 000 pounds per stage.

For each of the stage weights considered, performance of the tandem configuration is greater than 12 800 fps. This Earth-departure ΔV is sufficient to launch the spacecraft on many interplanetary missions. A summary of Earth-departure ΔV requirements for 34 interplanetary missions and the performance capability of the tandem configuration are given in Figure 7. Twenty-nine of the missions are within the capability of the tandem configuration. Four of the five remaining missions are flyby missions and are no longer under serious consideration. Only one of the 30 candidate missions that are shown is not within the capability of the tandem-configured vehicle.

A polar plot of the thrusting arcs of a multi-orbit injection is given in Figure 8. The illustrated injection is the eight-thrusting-arc injection for the 300 000-pound-stage tandem-configuration case. In this injection, the first seven thrusting arcs occur within approximately ±32° of the perigee. These arcs were shortened from the

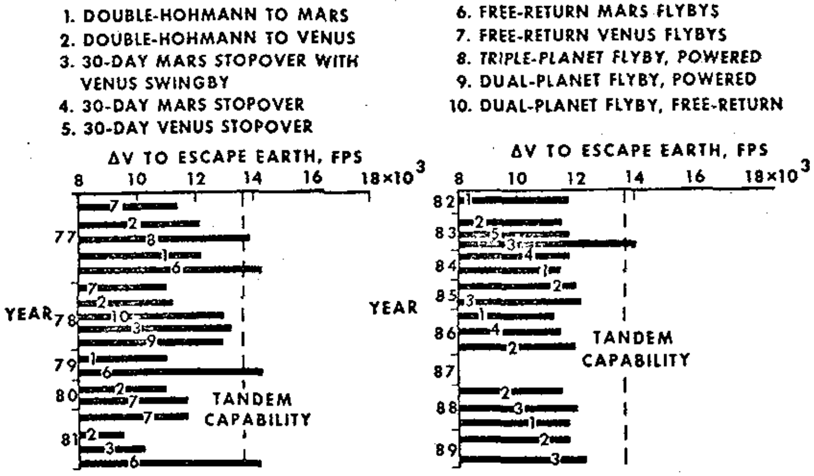

Figure 7. Orbital-launch velocity requirements for 34 interplanetary missions (departure from 100- by 400-nautical mile orbit).

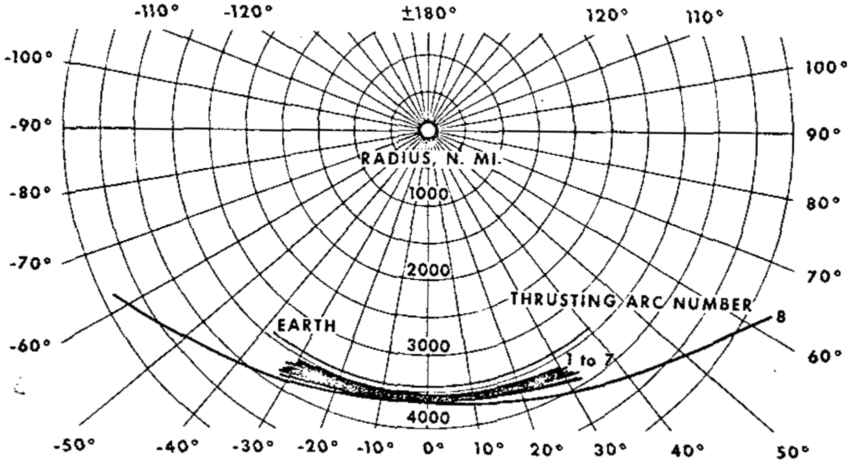

Figure 8. Thrusting arcs for eight-burn escape maneuver, 300 000 pounds per stage, tandem configuration.

optimum for the total injection to constrain the last intermediate orbit period to 48 hours. The last thrusting arc is larger than the first seven so that the highest possible final velocity can be reached from the constrained final intermediate orbit. Perigee was raised approximately 200 nautical miles during the injection. Apogee of the last intermediate orbit had an altitude of 65 190 nautical miles. At final thrust termination, the vehicle had a hyperbolic excess velocity of 14 270 fps.

Because injection time can be a factor when the injection profile is being determined, times for the various injections for each case are presented in Figures 9 to 12. The times that are shown are measured from semilatus rectum passage on the descending leg of the initial orbit immediately before the first thrusting period to final thrust termination. Injection time increases rapidly with an increasing number of thrusting arcs until the 48-hour-period constraint is reached, then it increases slowly. The longest injection time encountered was 106 hours for the 550 000-pound, 2/3-staged vehicle when the 12-thrusting-arc injection was performed. An injection time of 4 days or less does not seem to be an unreasonable addition to the flight time. Interplanetary missions that are being discussed now will be as long as 1100 days, with several months in Earth orbit preceding the missions for launch and assembly of the vehicle.

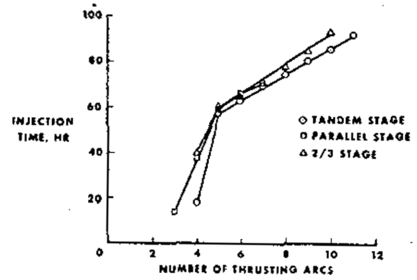

Figure 10. Injection time for vehicle with 375 000 pounds per stage.

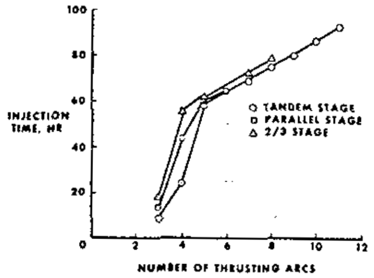

Figure 9. Injection time for vehicle with 300 000 pounds per stage.

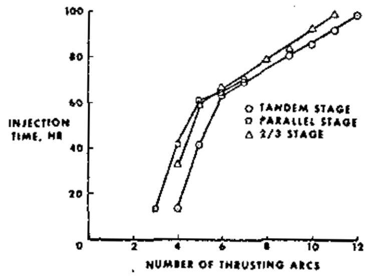

Figure 11. Injection time for vehicle with 425 000 pounds per stage.

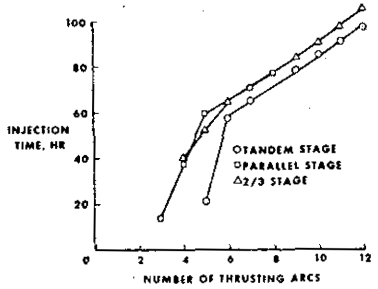

Figure 12. Injection time for vehicle with 550 000 pounds per stage.

The multiorbit-injection technique will require the spacecraft to pass several times through the Van Allen radiation belt. The requirement for crew protection from Van Allen radiation is the equivalent of 5 grams of aluminum per square centimeter of shield. The requirement for crew protection from solar flares is the equivalent of 10 grams of aluminum per square centimeter of shield. Because protection from solar flares will be provided for manned interplanetary missions, adequate shielding from Van Allen radiation certainly will be provided.

Recent studies indicate that a reasonable estimate for the cooldown requirement is a linear function of engine operation time. However, the study reported in this paper assumed no startup and shutdown ramps, which do occur and require some propellant. While the performance gained from the startup and shutdown ramps is assumed to be small enough to be neglected, an amount of propellant corresponding to that required for startup and shutdown should be added to the cooldown propellant. This procedure will give a total propellant cost for starting and operating the engine for a given length of time. The simplified ramp and cooldown propellant requirement that is assumed certainly is not completely representative of the actual requirement of the nuclear engine now being developed. The requirement actually is a complex function of the engine startup and shutdown ramps and the interval between engine uses, as well as of engine full-power operation time. However, the assumed requirement is accurate enough for the purposes of this study. A requirement was used so that some optimum number of thrusting arcs would result as the cost of repeated starts becomes larger than the reductio [illegible] propulsive losses gained by more thrusting arcs. Any other reasonable requirement assumed would not alter the relative performance of the configuration but would change only the magnitude of the performance of both. Incorporation of the thrust caused by cooldown likewise would not affect the relative performance but would alter only the magnitude of the performance of the configurations. Exclusion of the performance gain from the startup and shutdown ramps is justified by the low thrust-to-weight ratio in these operating periods and by the relatively short time involved. The exclusion of the effect of cooldown thrust can be justified because the extremely low thrust-to-weight ratio, the lowered specific impulse, and the high altitude at which cooldown is performed contributed to extremely high propulsive losses for the cooldown portion of the injection.

Recent nuclear-flight-stage studies indicate that the assumed empty stage weights may be low. For example, the 375 000-pound gross weight NOLS may have an empty weight of almost 100 000 pounds. The added weight will penalize the parallel-configured vehicle more than the tandem-configured vehicle. Again, the reason is that more weight must be carried to higher velocities.

Conclusions

The results of the performance study indicate that unless some strict constraint is placed on injection time, no performance advantage exists to staging nuclear-rocket-engine-powered orbital launch stages (NOLS) in parallel for the Earth-escape maneuver onto interplanetary trajectories. If the three NOLS are assembled in parallel, no performance justification exists, at least for the smaller stage sizes, for simultaneous use of all three NOLS because the 2/3-staging technique results in a performance gain. This study dealt only with the use of three NOLS for Earth departure; however, if two NOLS are used, preliminary studies indicate that a vehicle with the two arranged in tandem will outperform a vehicle with the two in parallel.

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