Terminal Descent

Terminal descent begins with the ignition of the main retro, which develops 15,000-lbf thrust and a specific impulse of 444-lbf-sec/lbm. The engine can be started in weightless conditions by letting some of the cryogenic liquids and bubbles flow through the pumps and the engine developing in the beginning a very low thrust, which is sufficient, however, to settle the propellants in the tanks. If a more rapid start is required, the verniers or the RCS can be started to perform the ullage maneuver. The total required burn time at a full thrust of 15,000 lbf is approximately 160–170 sec, depending on the type of mission. The gimballing of the engine provides pitch and yaw control of the spacecraft. The roll control during the retro fire is provided by four-RCS engines (which once started continue their operation until the touch-down).

During the retro descent, the Earth-based controller is monitoring the TV and may decide to redesignate the landing site and use the open-loop mode of the terminal guidance. The TV camera on the LLV looks via a mirror (which has a proper tilt for a given mission) through the exhaust plume of the retro. If the picture is not too clear the retro thrust should be reduced to the idle mode that is about 500-lbf thrust, and consequently the plume is very small. The maneuver can be repeated as the engine has a capability of three restarts.

When the main engine burns out and is jettisoned together with the propellant tanks, all the 16 vernier engines are ignited and take over the final descent. As the propellant tanks are pressurized to 295 psia, the thrust level of these engines will be 130–140 lbf depending on the pulse mode or steady-state mode of operation. The verniers are assisted by the RCS thrusters which are using the same propellants from the common storage tanks.

About 100 ft above the lunar terrain the spacecraft hovers for 20 sec with the aid of RCS and eight-vernier engines, some in a pulsed mode of operation because the LLV's lunar weight is now less than 1000 lbf. During the hover the Earth-based controller has time to reposition the spacecraft for a better landing and avoidance of boulders. It has to be borne in mind that there is a time delay of 6 sec from the TV looking at the terrain and the command reaching the spacecraft (the controller's time for viewing the picture and making his decision is included).

Landing

The descent from hover to a soft touch-down is performed under retro power of the vernier and RCS engines. The landing on slopes of up to 35° requires operating the verniers (on the appropriate sides of the LLV), or the use of small stabilization solid motors firing upward at the time of touch-down.†

Considerations were given to the survivability of the spacecraft for 90-Earth days on the moon, and a possibility for the spacecraft to lift and translate by one km to provide a closer surface rendezvous with the astronauts (such maneuver is referred to as "hop"). Cooling requirements during lunar days, and the weight of additional propellant for the hop are problems which require further investigation.

References

[1] Hendel, F. J., "Attitude Control with Hydrogen Microthrusters," AIAA Paper 70-613, San Diego, Calif., 1970.

[2] "Lunar Logistics Vehicle," Final Rept., Sept. 1969, NASA-ASEE Summer Faculty Institute, Houston, Texas.

[3] Hendel, F. J., "Lunar Logistics Vehicle—Summary," Sept. 1969, NASA-ASEE Summer Faculty Institute, Houston, Texas.

† Investigations and simulated tests pertaining to stabilization and attitude control of LIV were performed in 1970 by the following senior students (now graduate aeronautical engineers from California State Polytechnic College): G. A. Matack, J. W. Morris, H. L. Weaver, and J. S. Claes. Claes successfully proved the stabilization technique for lunar landing on slopes.

Supply and Resupply of Stations in Synchronous Orbit

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Current use and plans for use of a synchronous Earth orbit for communication satellites, astronomical telescopes (both manned and unmanned), and manned space stations suggest that future traffic to the 24-hr equatorial orbit is likely to be extensive. Although some preliminary analysis of how this traffic might be implemented has been made by NASA and others, no results, to the best knowledge of this writer, have been published in forms available to the general reader. The present study concerns transfer from a low parking orbit to synchronous orbit, using single- and two-stage systems and reusable and nonreusable stages, all with chemical propulsion. Methods of disposing of nonrecovered states are also discussed. Two primary transfer modes are considered: the use of one-way vehicles that are then discarded, and the reuse of a logistics vehicle that delivers a payload to synchronous orbit and returns another one to the low Earth orbit. The specific impulse of chemical stages is taken as 450 sec.

Analysis

In order to calculate a consistent set of velocity increments to serve as a basis for the purpose of this study, several assumptions or ground rules are made, viz.: 1) The circular Earth parking orbit has an altitude of 150 naut miles and an inclination of 28.45° equal to the latitude of the Atlantic Missile Range. 2) The synchronous orbit has a radius of 6.6107 Earth equatorial radii, corresponding to a sidereal period of 23 hr 56 min 4.09054 sec. 3) A Hohmann elliptical transfer trajectory coplanar with the circular parking orbit is used to ascend to and descend from synchronous altitude. 4) Capture into the synchronous orbit or departure from it is accomplished by combining the 28.45° plane-change requirement simultaneously with the requirement for matching the velocity of the synchronous orbit or of the return trajectory. 5) Gravity losses incurred during the foregoing maneuvers are negligible, as Fig. 1 shows they would be if initial thrust/weight ratios are at least as large as 0.2.

On the basis of the foregoing assumptions, the required velocity increments are $\Delta V_1 = 2.433$ km/sec (7980 fps) for injection into Hohmann transfer from parking orbit or vice versa, and $\Delta V_2 = 1.830$ km/sec (6005 fps) for acquisition of synchronous orbit from Hohmann transfer or vice versa. [If the synchronous orbit were to be acquired by performing the velocity-matching and plane-change maneuvers sequentially rather than simultaneously, ΔV_2 would be nearly 2.258 km/sec (7409 fps)].

One-way trips are considered here to be accomplished with either single- or two-stage propulsion systems with no recovery. In the case of two stages, the stage velocity increments can be apportioned in several ways depending upon requirements. For example, it might be advantageous to have the gross mass of each of the two stages the same from consideration of packaging or handling operations. On the basis of flight operations, the use of the first stage to inject into the transfer ellipse and the use of the second stage to acquire the synchronous orbit may be appropriate. Another alternative is to divide the total velocity increment between the two stages in such a way that maximum over-all performance is achieved.[1]

For the round-trip mission described earlier, a ***single-stage*** propulsion system would require a total of three restarts.

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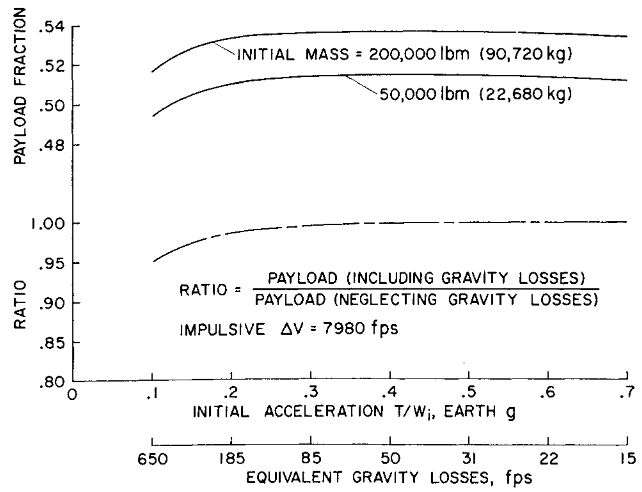

Fig. 1 Effects of initial acceleration on performance of chemical stage during injection from 150 naut miles parking orbit into Hohmann transfer trajectory to synchronous altitude.

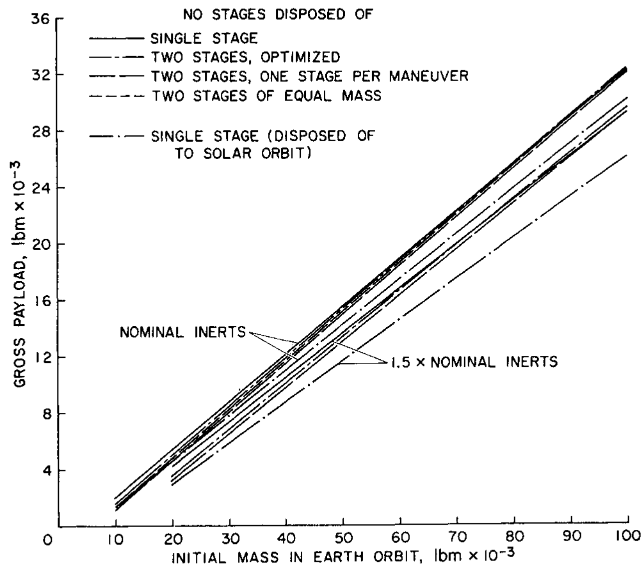

Fig. 2 Performance of single- and two-stage systems in one-way trips.

There are several possible methods of using *two stages* in round-trip missions. The two considered here are: 1) return of both stages to the low Earth orbit for later reuse, and 2) use of the first stage to acquire the synchronous orbit and retention of it there while the second stage exchanges its payload for a similar one and returns to the low Earth orbit. The mission profile assumed for the first two-stage case is as follows. After reaching the apogee of the Hohmann transfer ellipse, the first stage separates from the second stage, coasts back to parking orbit altitude, and restarts to acquire the parking orbit. The second stage ignites to acquire the synchronous orbit, exchanges its payload with a similar one already there, restarts to inject into a return Hohmann trajectory and restarts once more to capture into the parking orbit.

Disposal of Stages

The problem of disposal of nonrecovered stages or stages whose serviceable lifetimes are expiring will be an important one. Reusable stages could conceivably be rebuilt or refurbished in low parking orbit or returned to ground stations for salvage of parts, but such procedures may not be economically attractive. Let us examine some alternative methods of avoiding the littering of near-Earth space.

Obviously, if a single stage is to be disposed of after fulfilling its one-way payload delivery, there will be penalties in the form of reduced payloads and/or larger initial masses. For example, changing the orbit from the circular synchronous orbit to one that enters the atmosphere at an appropriate angle to insure burnup of the spent stage requires a velocity change of approximately 1.49 km/sec (4890 fps), whereas to acquire parabolic speed from synchronous orbit, about 1.27 km/sec (4178 fps) is needed. Hence, disposal of the one-way stage by sending it to escape into a solar orbit appears to entail less propellants than disposal by burnup in the atmosphere. Since for small hyperbolic excess speeds, ΔV requirements are not much greater than for parabolic speed, and since the addition of a separate small payload to the one-way stage might be practical, the possibility is examined of using the stage not only for its chief purpose of placing a payload into synchronous orbit but also to place a small "escape" payload into an interplanetary trajectory as a flyby or orbiter of some planet or into a lunar trajectory.

Disposal of a resuable single stage after it has reached its expected lifetime necessarily would take place after it had returned its payload to the low Earth parking orbit. It might be used, as its last flight, to inject a flyby or orbiter vehicle on a suitable interplanetary or lunar mission if refurbishment should prove impractical. More interesting is the possibility of using the first stage of a two-stage system to serve in its last flight as the space propulsion system for sending payloads on escape trajectories (or to the Moon) and thus dispose of itself while performing a useful mission. The case in which the first stage injects the second stage into the Hohmann transfer orbit and continues in that orbit is particularly attractive, since the ΔV required to attain parabolic speed near the periapsis of the transfer orbit is less than one-third (0.77 km/sec) of the ΔV required to capture into the low parking orbit (2.433 km/sec). With no penalties in terms of payload delivered into synchronous orbit or of increased requirements in initial mass, this first stage could inject itself and modest payloads into lunar or interplanetary trajectories and eliminate the disposal problem. (Only the second stage, which is required to return a payload to the parking orbit, would need to be disposed of.) With additional propellants, the first stage could accommodate larger escape payloads or higher hyperbolic speeds. To examine this possibility quantitatively, payloads delivered to synchronous orbit for given escape payloads are calculated.

Results

Figure 2 indicates that the use of two stages for one-way trips is unwarranted for the range of payloads considered.

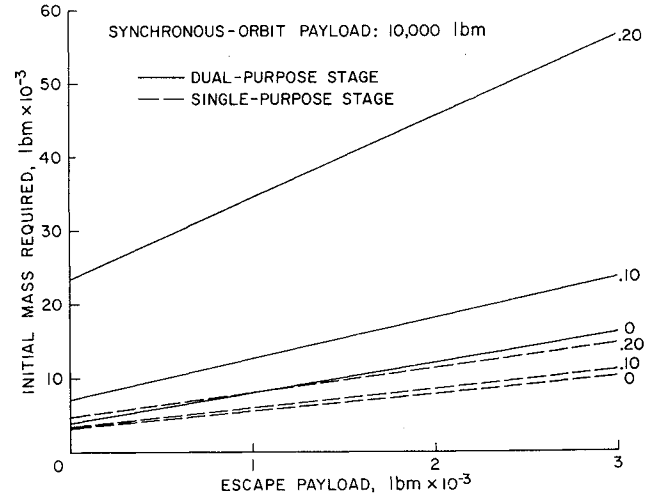

Fig. 3 Comparison of initial mass requirements for 10,000-lbm payload injected to escape by synchronous-orbit stage and by separate stage in parking orbit, one-way trips.

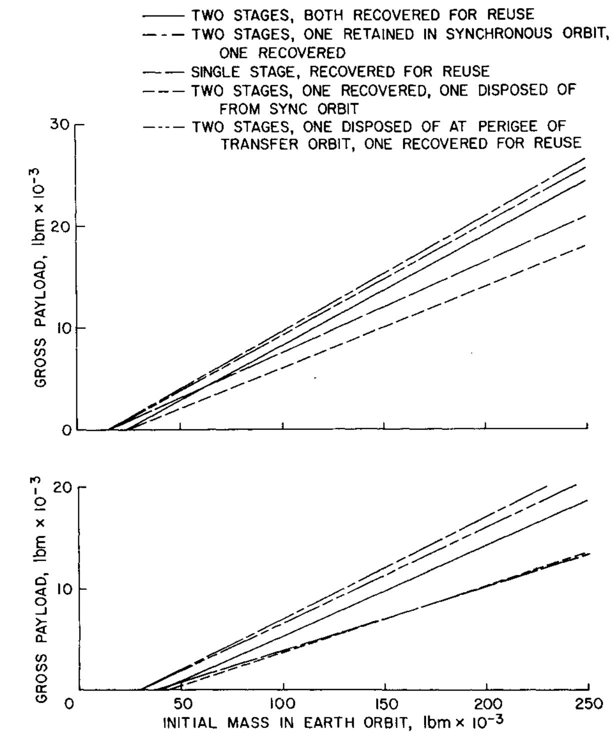

Fig. 4 Performance of single- and two-stage systems in round trips; (top) nominal inerts; (bottom) 1.5 nominal inerts.

The effects of a 50% increase in stage inerts range from about 10% to 50% decreases in payload for initial masses in Earth orbit between 10,000 and 100,000 lbm. The actual increase in the stage mass (chiefly more propellants) required for disposition ranges from about 2000 lbm for small payloads to 5000 lbm for payloads as large as 20,000 lbm, if nominal inert masses are assumed.

Figure 3 compares the performance of a stage that delivers a payload into synchronous orbit and subsequently injects an additional small payload into an interplanetary trajectory, with the performance of a stage that directly injects a small payload to escape from a low parking orbit. Despite the lower inert fraction of the larger dual-purpose stage, the additional mass required initially by the dual-purpose stage for an escape payload is greater than that required by a single-purpose stage starting from a low parking orbit, particularly at larger hyperbolic excess speeds. If the stage that delivers a payload into synchronous orbit were not to be disposed of, there appears to be no performance advantage in using it to send additional payloads to the Moon or to other planets.

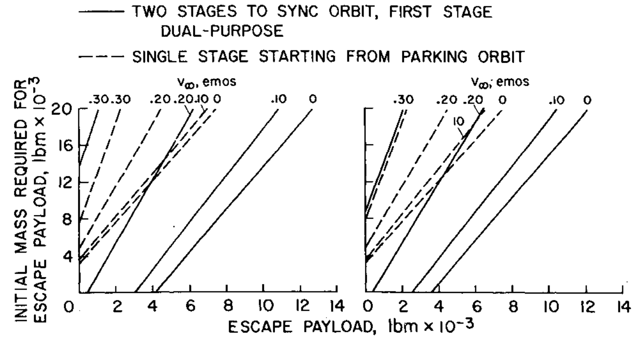

Fig. 5 Comparison of initial mass requirements for payloads injected to escape by one of two stages used to deliver and return synchronous-orbit payload and by single-stage starting from parking orbit; (left) synchronous-orbit payload, 20,000 lbm; (right) synchronous-orbit payload, 10,000 lbm.

On the other hand, if the stage were to be disposed of in any case, the additional mass required would be about 3000 lbm (for zero payload and for $V_\infty = 0$), so that for small escape payloads and for V_∞ not much larger than 0.10 emos (3 km/sec), the net increase in initial mass requirements may not significantly exceed those required for direct injection from low parking orbit. When the costs of acquiring the single-purpose stage and of placing it into the low parking orbit are compared with the costs associated with increasing the size of the synchronous-orbit stage to accommodate an escape payload, the use of the synchronous-orbit stage to send payloads on interplanetary missions may well prove more cost-effective over a useful range of escape payloads and energies than the use of a separate stage. Such cost comparisons are beyond the scope of the present study.

Figure 4 shows that the use of two stages, both recoverable, gives better performance for payloads larger than about 4000 lbm than a single reusable stage. For payloads between 10,000 and 20,000 lbm, about 15% more payload is delivered and returned by the two stages if nominal inerts are assumed. Should inert masses turn out to be 50% larger than the assumed nominal values (bottom part of Fig. 4), the advantage of two stages over a single stage increases. If the first of two stages were to be used to place the second stage and the payload into synchronous orbit and remain there (uppermost curve), up to 10% more payload could be delivered than with two stages, both of which are recovered in the parking orbit. Such use of two stages gives essentially maximum performance. If this first stage is disposed of into solar orbit from synchronous orbit, the performance decreases markedly, as shown by the dotted curve. A more efficient method of disposal of the first stage is represented by the second highest curve in the figure. In this method, as in the use of two recovered stages, the first stage injects the second stage and payload into the Hohmann transfer trajectory to synchronous altitude and continues its flight back to the parking orbit; but instead of retro-thrusting into the parking orbit, it thrusts during perigee passage to attain parabolic speed.

The additional use of the first stage just discussed to inject mission payloads into lunar or interplanetary trajectories is compared in Fig. 5 with the use of a separate stage starting from a low parking orbit for the same purpose. With no additional mass requirements beyond those required for delivering and returning synchronous-orbit payloads, the two stages could also send small to moderate unmanned payloads on flyby or orbiter missions to the Moon or to Mars and Venus. For hyperbolic excess speeds less than 0.20 emos, the escape payloads possible with no additional initial mass increases slightly with an increase in the mass of the synchronous-orbit payload (SOP). The advantage, in terms of initial mass requirements, of the two stages over the single stage starting from the parking orbit is about 11,000 lbm for $V_\infty = 0$, decreasing to about 6500 lbm at $V_\infty = 0.20$ emos, and vanishing at about $V_\infty = 0.30$ emos for a synchronous-orbit payload of 10,000 lbm and at a somewhat smaller V_∞ for larger SOP. For $V_\infty = 0.30$ emos or higher, the situation is analogous to that discussed previously in the case of the use of dual- and single-purpose stages in one-way supply missions; i.e., cost-effectiveness may be an important criterion as to wheter the dual-purpose stages can be advantageously used for high-energy interplanetary missions. Other considerations, such as compatibility in scheduling of interplanetary missions and of resupply trips to synchronous orbit, adjusting the plane of the departure orbit to meet mission requirements, etc., would of course be factors in determining whether or not dual-purpose use of stages is advantageous in a given application.

Reference

[1] Dugan, D. W., "The Role of Staged Space Propulsion Systems in Interplanetary Missions," TN D-5593, Dec. 1969, NASA.