couple probes, if speed of response is not of prime importance. If, for instance, the governing conduction temperature is monitored, then errors due to nonuniform flow around the probe body can be reduced by a data reduction scheme accounting for conduction effects.

References

[1] Hill, F. K., "Turbulent Boundary Layer Measurements at Mach Numbers from 8 to 10," *The Physics of Fluids*, Vol. 2, No. 6, Nov.–Dec. 1959.

[2] Jones, R. A. and Feller, W. V., "Preliminary Surveys of the Wall Boundary Layer in a Mach 6 Axisymmetric Tunnel," TN D-5620, Feb. 1970, NASA.

[3] Goldstein, D. L., and Scherrer, R., "Design and Calibration of a Total Temperature Probe for Use at Supersonic Speeds," TN 1885, 1949, NACA.

[4] Winkler, E. M., "Design and Calibration of Stagnation Temperature Probes for use at High Supersonic Speeds and Elevated Temperatures," *Journal of Applied Physics*, Vol. 25, No. 2, Feb. 1954, pp. 231–232.

[5] Yanta, W. J., "A Hot-Wire Stagnation Temperature Probe," TR 68-60, June 1968, Naval Ordnance Lab, White Oaks, Md.

[6] Softley, E. J., "Use of a Pulse Heated Fine Wire Probe for the Measurement of Total Temperatures in Shock Driven Facilities," TIS R 68SD2, Feb. 1968, General Electric Co.

[7] Wagner, R. D., Jr. et al., "Influence of Measured Freestream Disturbances on Hypersonic Boundary-Layer Transition," *AIAA Journal*, Vol. 8, No. 9, Sept. 1970, pp. 1664–1670.

[8] Fischer, M. C., Maddalon, D. V., and Weinstein, L. M., "Boundary Layer Surveys on a Nozzle Wall at $M_\infty = 20$ Including Hot Wire Fluctuation Measurements," AIAA Paper 70-746, Los Angeles, Calif., 1970.

[9] Ames Research Center Staff, "Equations, Tables, and Charts for Compressible Flow," Rept. 1135, 1953, NACA.

[10] Maye, J. P., "Error Due to Thermal Conduction Between the Sensing Wire and Its Supports When Measuring Temperatures with a Wire Anemometer Used as a Resistance Thermometer," Information 9, Feb. 1970, DISA, Elektronik A/S, Herlev, Denmark.

[11] Kreith, F., *Principles of Heat Transfer*, 2nd ed., International Textbook, Scranton, Pa., 1965.

[12] Adcock, J. B., and Peterson, J. B., Jr., "Experimental Investigation of a Turbulent Boundary Layer at Mach 6, High Reynolds Numbers, and Zero Heat Transfer," TN D-2907, July 1965, NASA.

Unmanned Lunar Logistics Vehicle May Support the Astronauts

Frank J. Hendel*
California State Polytechnic College,
San Luis Obispo, Calif.

For a program of extended lunar exploration under consideration is the addition of an unmanned lunar logistics vehicle (LLV) which will deliver a 2500-lb payload anywhere on the moon's visible face. The spacecraft will be launched by the Titan IIID/Centaur, using at Cape Kennedy the same launch facility as the project Viking.[1]

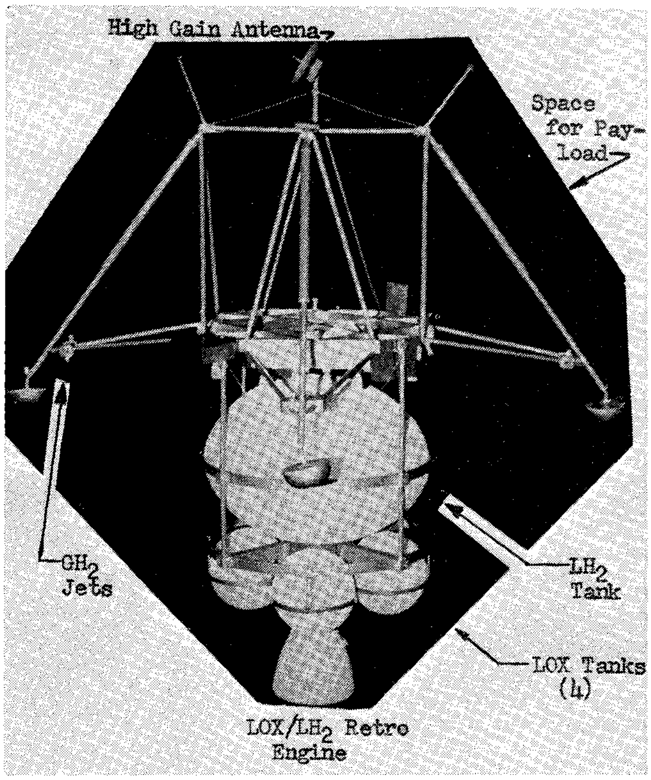

Fig. 1 Model of the lunar logistics vehicle.

It is anticipated that many LLV missions would be in support of manned explorations, and that the payload would consist of equipment that the astronauts will need on the lunar surface but cannot carry with them. Such payload may include a lunar flying unit, science payloads of various types e.g., "Apollo Lunar Science Experiment" (weighing up to 500 lb), a lunar expandable shelter (weighing about 1500 lb), various tools and supplies needed by the astronauts for an extended stay time on the moon, and, if required, a larger lunar roving vehicle. Missions not supporting man would carry experimental packages and, perhaps, devices for lunar mobility of the experiments.

The spacecraft is to soft-land at any selected site on the visible lunar face of the moon with a 3-sigma landing dispersion of one km. Cislunar flight times from 60 to 120 hr are contemplated.

Figure 1 shows the structure and the equipment which weighs approximately 9000 lbs when fully loaded with propellants, fuel cell reactants, and helium gas. The dimension of the spacecraft in the stowed condition is 150 in. in diam. × 210-in. long.

The lower part of the spacecraft forms the main retro propulsion unit and consists of a large liquid hydrogen tank (254 ft^3), four liquid oxygen tanks (each 18 ft^3), and the Pratt and Whitney RL 10A-3-3 engine developing a thrust of 15,000 lbf. This part of the structure is jettisoned after the main retro burns out. The empty main retro equipment and its supporting structure has an approximate mass of about 1000 lb; the tanks contain 5000 lb of LOX and 1000 lbs of LH_2.

Immediately above the LH_2 tank is the main lander structural platform. Mounted on the top of the platform is the payload and below the platform most of the lander subsystems.

Presented as Paper 70-613 at the AIAA 6th Propulsion Joint Specialist Conference, San Diego, Calif., June 15–19, 1970; submitted September 21, 1970; revision received October 9, 1970. Preliminary design of the vehicle[2,3] was performed by F. O. Calvert, J. H. Cole, F. M. Davidson, I. Eisenstein, C. M. Epps, G. E. Gless, G. F. Hauck, F. J. Hendel, B. Koo, S. J. Kozak, T. M. Perkins, A. J. Perna, G. Pincus, C. G. Richards, J. E. Robertshaw, J. E. Rozenberg, F. R. Swenson and J. M. Wampler under direction of S. L. Dickerson, C. J. Huang, D. B. Mackay, and A. N. Paul, all working under NASA Grant NSR 44-005-059 (1969 NASA-ASEE Institute in Houston, Texas). Support of R. Abel, B. Redd, R. Bristow, N. Chaffee, B. D. Kendrick, M. Lauston, P. G. Thomas, C. Vaughen, and others of NASA-MSC is gratefully acknowledged.

* Associate Professor, Aeronautical Engineering. Associate Fellow AIAA.

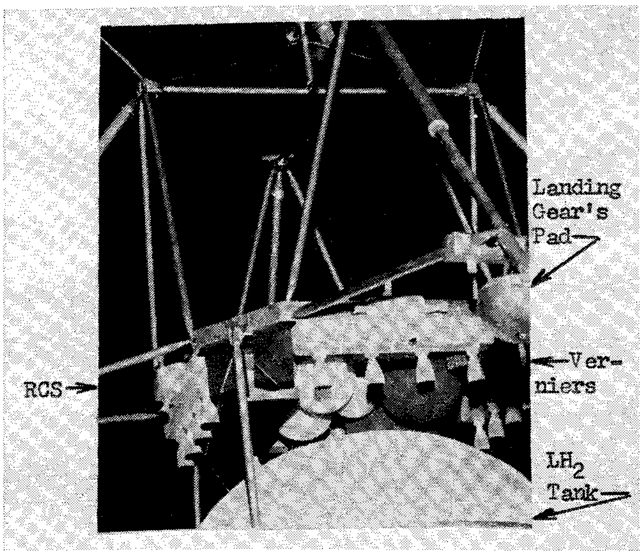

Fig. 2 Details of Vernier and RCS engines.

Around the periphery of the platform are four housings (Fig. 2) each with a cluster of four-vernier engines. The engines are the Marquardt R-4D of 130–140 lbf thrust. In addition, there are 12 reaction control engines (RCS), the Marquardt R-1E each of 22-lbf thrust. The RCS engines are used in couples to provide rotation about the three axes and in double pairs to provide translations.

Located between all the verniers and the RCS engines are the following pieces of equipment: 1) two-spherical tanks with expulsion bladders and hypergolic propellants (nitrogen tetroxide and monomethyl hydrazine) required for the verniers and RCS; 2) one helium spherical tank to pressurize the aforementioned propellants and to supply gas to the pneumatically operated valves of the main retro; 3) two-spherical tanks with oxygen and hydrogen (under supercritical conditions) required to supply reactants to the fuel cells; 4) two Allis Chalmers fuel cells, 5) two electronic equipment boxes; 6) one compartment with LM type inertial measuring unit (IMU) and flight computer for guidance and navigation; 7) one-LM type landing radar antenna protruding on one side to serve as an altimeter and doppler velocity measurement (for approaching to the lunar surface); and 8) two-Surveyor batteries (260 lbs). The fuel cells are rated at 200 w of continuous load or 400 w of pulsed load. The batteries are rated to deliver a total of 9 kw-hr or 300 amp-hr at 30 v.

Protruding on four sides of the LLV is the landing gear with four legs, each provided with crushable struts and pad. In the stowed condition within the shroud the legs are folded down. Located on each leg is one small solid motor to stabilize the craft on landing.

Each leg is also provided with the attitude control jets which use gaseous hydrogen from the large LH_2 tank (Fig. 3). Hydrogen comes from this tank either as boil-off or as liquid which is then evaporated in a small accumulator located near the top of the LH_2 tank. Two legs on the opposite side have four gas jets and two others have only two gas jets. A total of 12 gas jets provide pitch, yaw, and roll control of the spacecraft during coasting in a fassion similar to the RCS engines. Each gas jet develops a thrust of 0.2 lbf.[1]

Above the main lander structural platform there is a space reserved for the payload. Above the payload compartment there are two sensors attached to horizontal struts; these are the sun sensor and the Canopus star tracker. Still higher protruding on each side are two omnidirectional antennas to be used for radio communication when the spacecraft is still close to the Earth. At the very top is a high-gain S-band antenna for communication with the Deep Space Instrumentation Facilities (DSIF). This antenna and its counterweight are gimballed by an electric drive. The communication system is compatible with the Manned Space Flight Network and with the Deep Space Network.

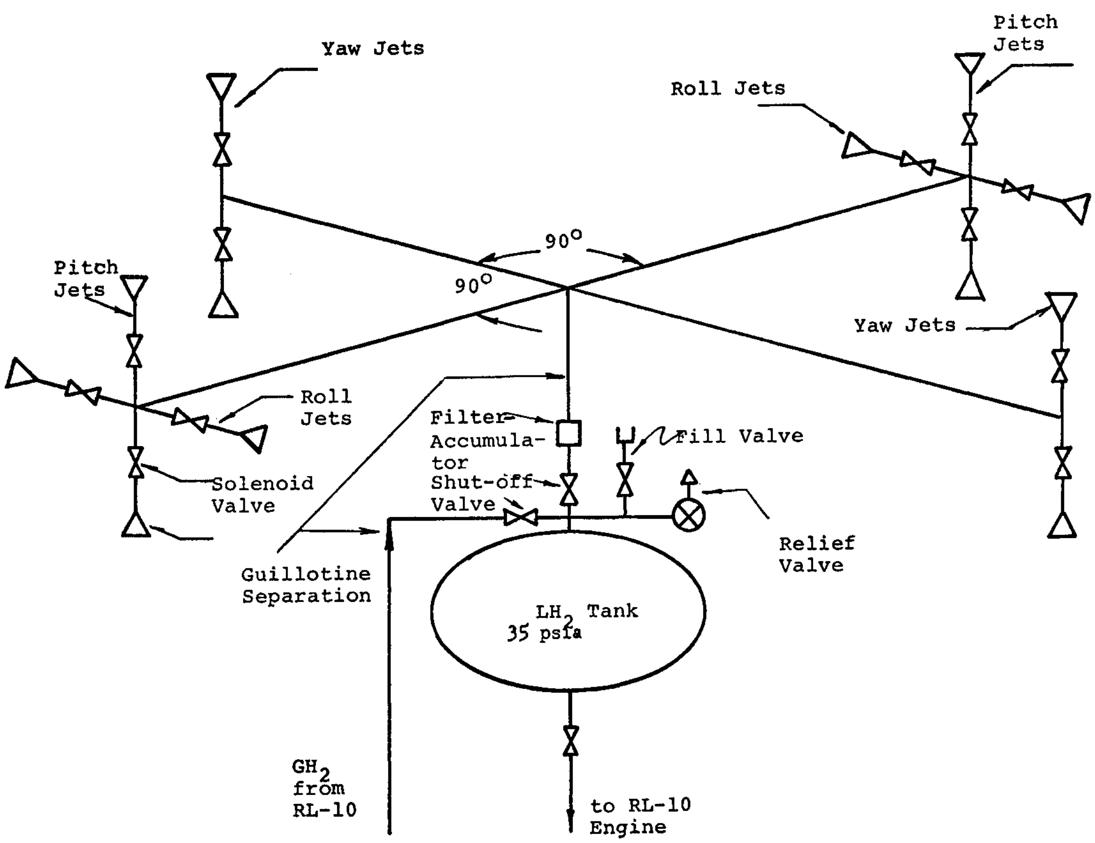

Fig. 3 Attitude control system of the lunar logistics Vehicle.

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

Duane W. Dugan*
NASA Advanced Concepts and Missions Division, OART, Moffett Field, Calif.

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.

Received November 2, 1970; revision received December 21, 1970.

* Research Scientist. Member AIAA.