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by Donald R. Bellman and Gene J. Matranga

Flight Research Center
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SUMMARY

This paper presents the significant technical details and research capabilities of a free-flight lunar-landing simulator as they existed at the time of the initial flights of the vehicle. The lunar-landing research vehicle (LLRV) consists of a pyramid-shaped structural frame with four truss-type legs. A pilot's platform extends forward between two legs, and an electronics platform is similarly located, extending rearward. A jet engine is mounted vertically in a gimbal ring at the center of the vehicle. During a lunar-landing simulation, the jet engine remains essentially vertical, regardless of the attitude of the vehicle, and the jet thrust supports five-sixths of the vehicle's weight. The remaining one-sixth of the weight is supported by hydrogen-peroxide lift rockets which are mounted on the main frame and tilt with the vehicle. Thus, attitudes and accelerations are similar to those that will be experienced on the moon where gravity is one-sixth that on the earth. The pilot controls the descent by means of a manual lift-rocket throttle and the vehicle attitude by means of 16 attitude rockets and a complex electronic control system. The electronics give great versatility to the controls, which makes it possible to simulate a wide variety of nonaerodynamic vehicles. Suitable displays provide the pilot with vehicle attitude, altitude, velocities, and acceleration, in addition to pertinent information on the propulsion system.

The LLRV is instrumented for research purposes. The data obtained are converted to digital form and transmitted to a ground tape recorder by means of an 80-channel pulse-code-modulation type (PCM) telemetry system. Each channel can be read every 0.005 second, if desired.

Many safety features are incorporated in the vehicle. The attitude-rocket system and the associated electronics are dualized. The jet engine has sufficient thrust to support the entire weight of the vehicle. Should it fail, a combination of normal lift rockets, emergency lift rockets, and a drogue parachute can be used to effect a landing. Furthermore, the pilot's ejection seat is capable of providing a safe recovery at zero velocity and zero altitude.

INTRODUCTION

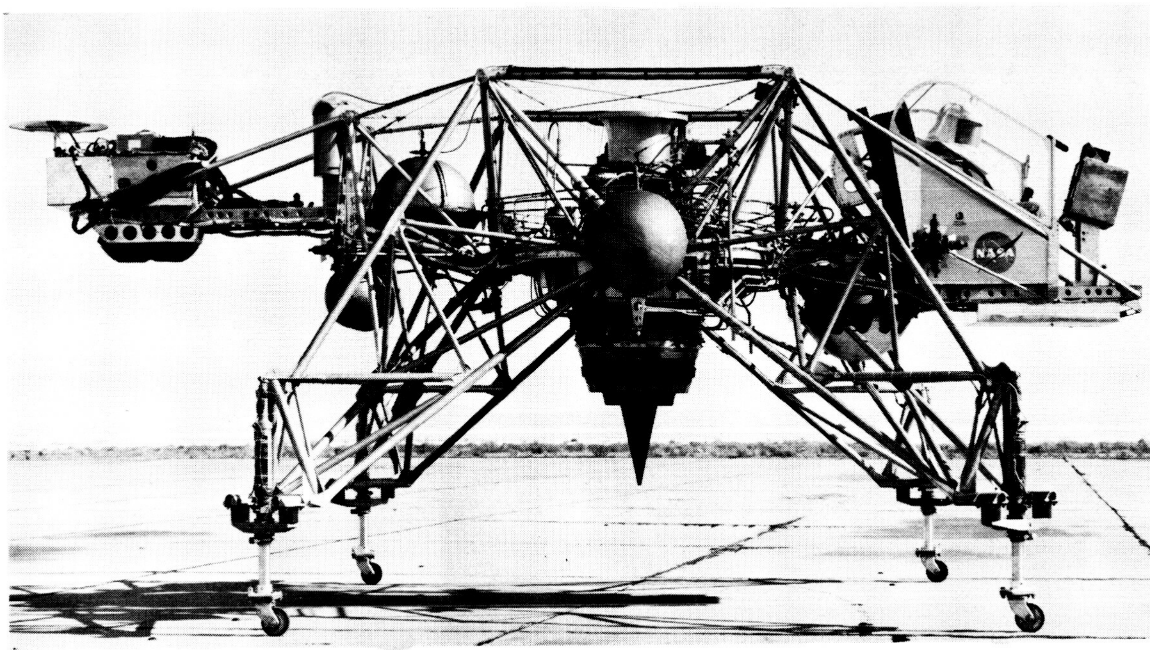
The successful achievement of this country's goal of landing a man on the surface of the moon and returning him safely to earth will require much research in many different fields. One of these areas is control of the vehicle by the human crew. It is expected that a major portion of the terminal descent and landing maneuver and the docking maneuver on return to orbit will be controlled by the crew. Such control will logically add to the mission reliability; the crew will be able to assess the suitability of the surface for landing and apply their control capabilities to provide a flexibility that would be impossible to achieve with an automatic system. Considerable study is needed to determine the capabilities of such a crew, the displays required, and the control authority necessary to make the most effective use of the human occupants. As a means of conducting such research, the NASA Flight Research Center, Edwards, Calif., obtained a specially developed free-flight vehicle, designated the lunar-landing research vehicle. Figures 1(a) and 1(b) are photographs of the vehicle on the ground and in flight, respectively.

On the earth, jet-supported vehicles, such as helicopters, can be maneuvered horizontally by tilting the lift-thrust vector a few degrees, thus creating a horizontal force component without significantly changing the vertical component. On the moon, however, because of the reduced gravity, the lift-thrust vector is only one-sixth as great as on the earth; hence, to achieve the same horizontal acceleration the vehicle would have to be tilted about six times as much. Thus, what would be a rather simple maneuver on earth could be difficult and hazardous on the moon.

This lunar effect is achieved on the LLRV by supporting the vehicle with two separate propulsion devices. Five-sixths of the vehicle weight is supported by a jet engine mounted vertically in a gimbal at the center of gravity of the vehicle. A system of gyros and hydraulic servos keeps the jet engine essentially vertical with respect to the earth, regardless of the vehicle's attitude. One-sixth of the vehicle weight is supported by a pair of rockets mounted on the main frame of the vehicle, one on either side of the jet engine. All horizontal maneuvering and braking is accomplished through the use of these rockets combined with the tilting of the vehicle under the pilot's control. A pilot flying the vehicle has essentially the same angular motions and visual cues that he would have if he were in a lunar gravitational field. Insofar as practical, automatic devices on the vehicle compensate for aerodynamic forces and moments, thus, simulating the lack of significant atmospheric effects on the moon.

Figure 2 shows how the LLRV will be used to simulate the final portion of a lunar-landing trajectory. The vehicle has the capability of duplicating the descent of the lunar surface from an altitude of about 1500 feet, including such maneuvers as hovering and horizontal translation. A lunar takeoff can also be simulated.

Two lunar-landing vehicles were delivered by the manufacturer to the Flight Research Center in the spring of 1964. After delivery, six months was devoted to checking systems and installing research instrumentation. During this period, many minor changes were made in the systems and the configuration. This paper presents the significant technical details and research capabilities of the vehicle for the configuration existing at the time of the first flight on October 30, 1964.



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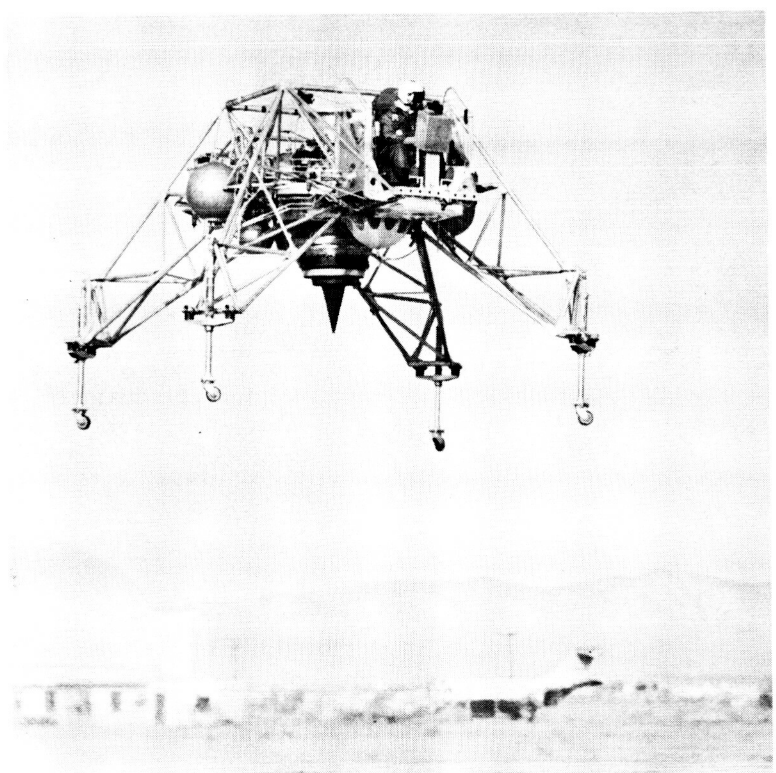
(a) On the ground.



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(b) In flight.

Figure 1.— The lunar-landing research vehicle.

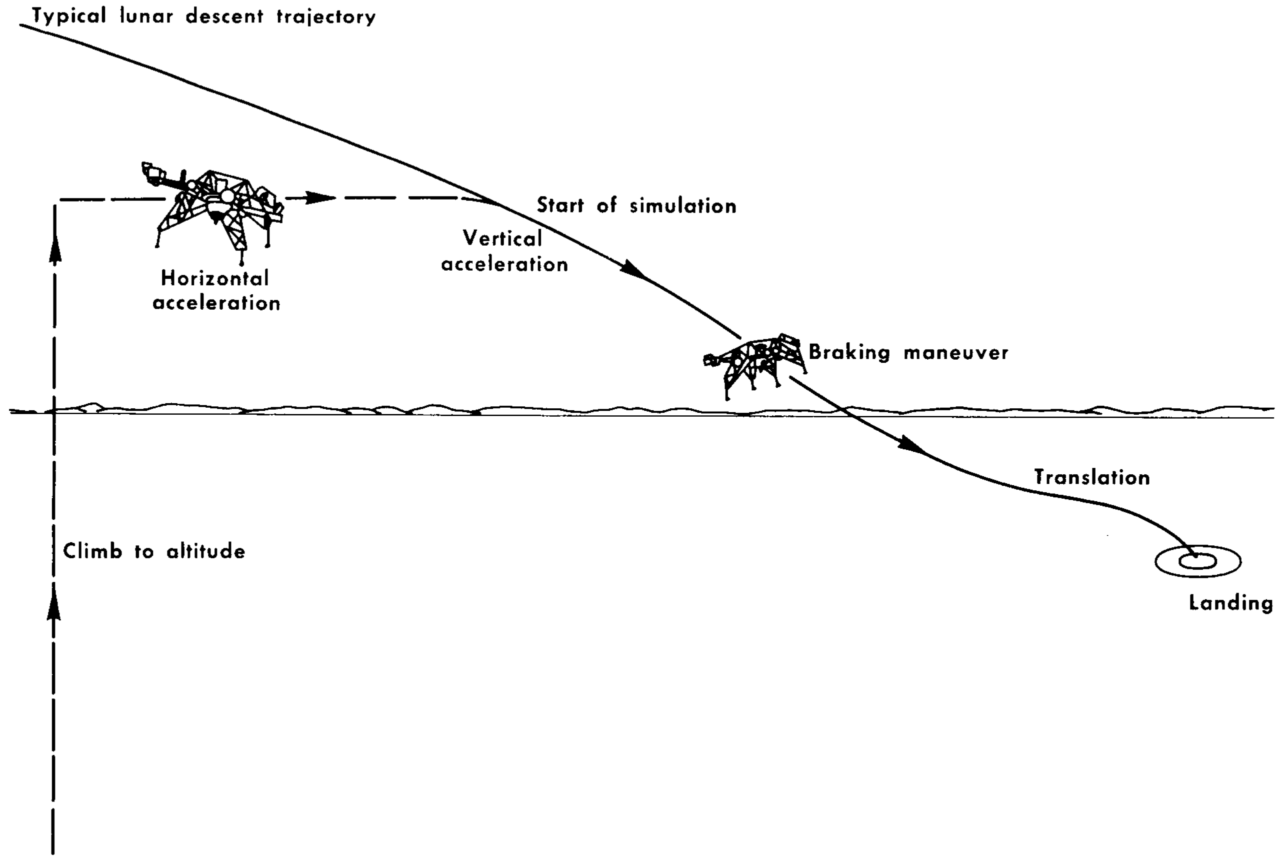


Figure 2.– Use of the LLRV to simulate the terminal portion of a lunar landing.

The notation used in this paper is defined in appendix A. Physical quantities are given, where applicable, throughout the paper in both the U.S. Customary Units and in the International System of Units (SI). Factors relating the two systems are given in reference 1.

CONFIGURATION DEVELOPMENT

In 1961, a committee of five engineers at the Flight Research Center made a study of various simulator techniques applicable to lunar-landing problems. The simulators considered ranged from fixed-base types to free-flight aircraft. The group concluded that a free-flight simulator, which would give a better degree of simulation than any fixed-base or moving-base type, was feasible. It was also concluded that the free-flight simulator should be designed for a minimum of aerodynamic effects in keeping with the lunar conditions, and that some means should be devised to compensate for the excessive gravitational force of the earth as compared to that of the moon.

Shortly after the conclusion of the study, Bell Aerosystems Co. independently proposed a free-flight lunar-landing simulator in which a gimbaled jet engine was used to compensate for the difference between earth and lunar gravitational forces, and rockets provided vehicle lift and control forces equivalent to those required on an actual lunar-landing vehicle. Because this vehicle was very similar to the concept set forth by the study committee, a contract was given to Bell to make a preliminary design of the simulator. Later, a contract was placed with Bell to construct two of these vehicles.

A detailed presentation of the results of the design study is given in reference 2, and more general aspects are given in references 3 and 4. To provide aerodynamic data for design purposes, tests were made on a 0.3-scale model in the 17-foot section of the Langley 7- by 10-foot wind tunnel. Reference 5 comments on these tests and gives additional information on the preliminary design. As the design progressed toward the actual construction of the vehicle, a major reconfiguring was necessary to obtain proper balance (ref. 6). The pilot, instead of being positioned above the jet engine, was placed lower and ahead of the engine. The weight of the pilot's platform was balanced by an instrument platform at the rear. The new design, which permitted a significant decrease in the overall size and weight of the vehicle, was the configuration constructed.

GENERAL FEATURES

The LLRV is a single-place, vertically rising, free-flight vehicle. It has no aerodynamic surfaces and is supported by a combination of jet and rocket engines. The vehicle consists of a lightweight structural frame with four rigid legs extending from a center body (see two-view drawing in fig. 3). A jet engine is mounted essentially vertically in a gimbal ring at the center of the frame. Eight lift-rocket engines are mounted on the frame. A spherical jet-fuel tank is mounted on the vehicle centerline ahead of the jet engine, and

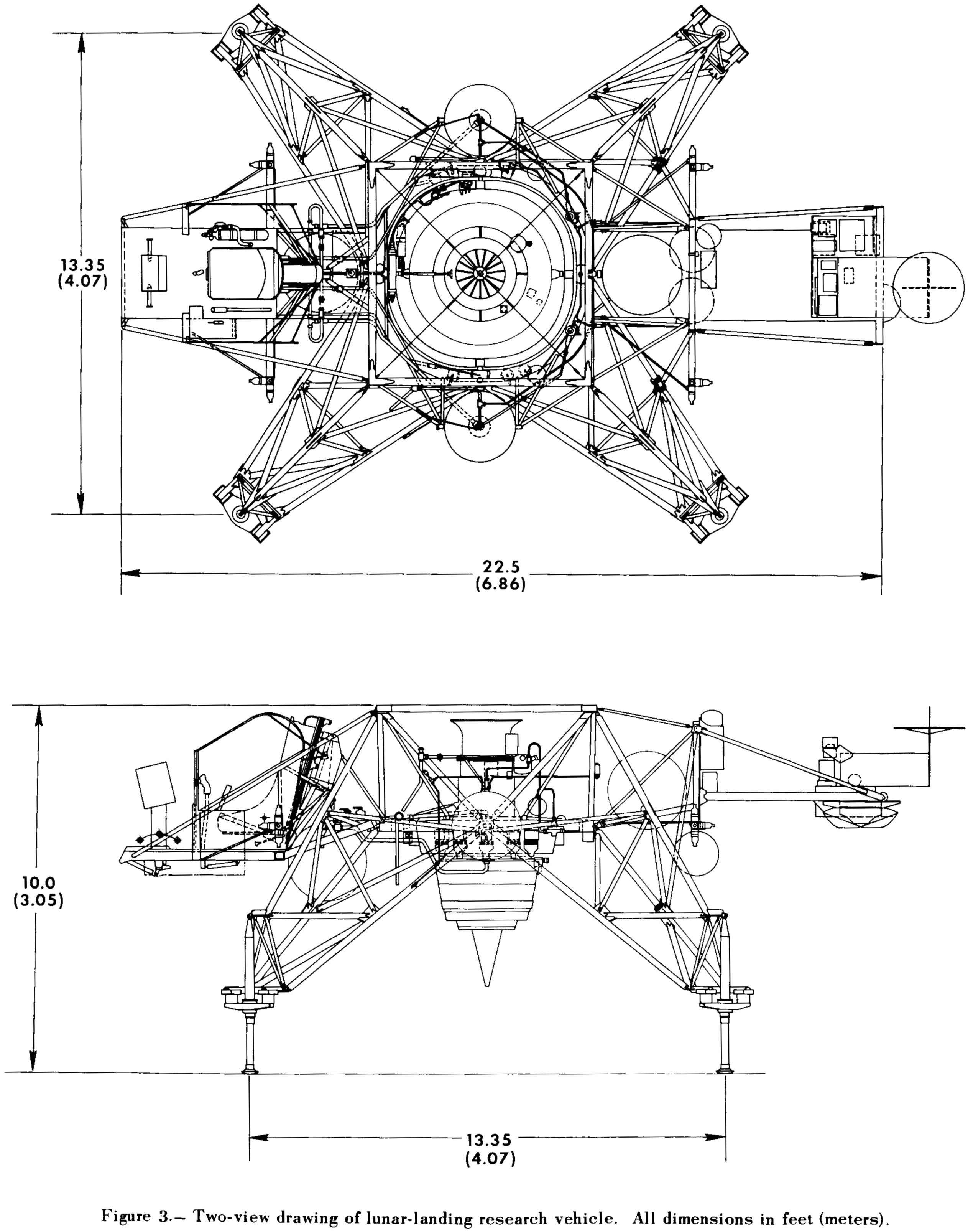
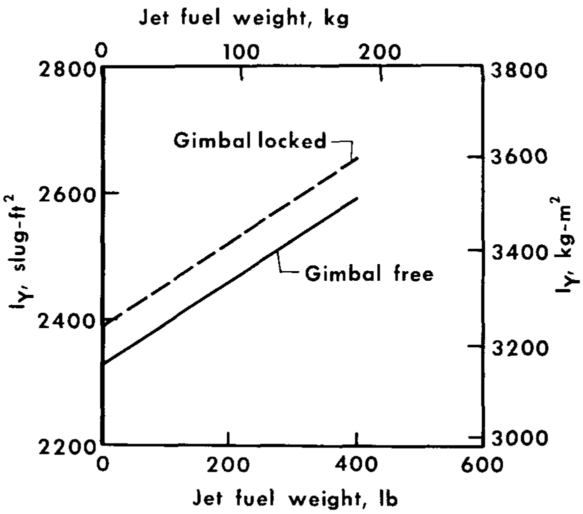


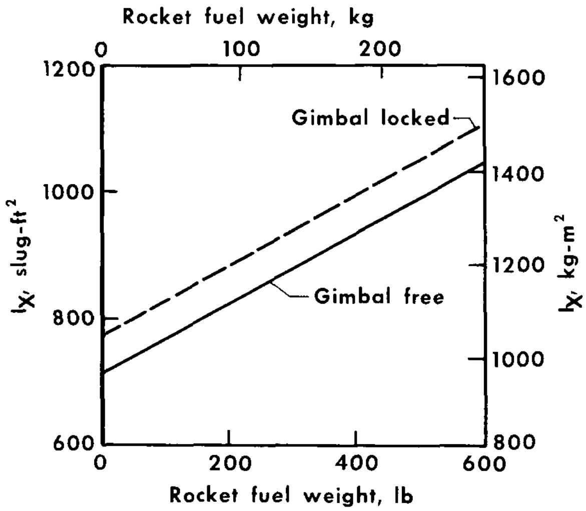
Figure 3.– Two-view drawing of lunar-landing research vehicle. All dimensions in feet (meters).

another tank is mounted behind the engine. A pair of spherical rocket-propellant tanks is located on the lateral axis of the vehicle, one on each side of the jet engine. High-pressure helium is stored in a pair of spherical tanks located aft and below the rear jet-fuel tank. A pilot's platform extends forward from the main frame, and an electronics-equipment platform extends to the rear. Attitude is controlled by means of 16 rockets mounted in clusters of four at each side of the pilot's and the electronics-equipment platforms.

Table I presents the general physical characteristics of the vehicle. The variations of vehicle moments of inertia with fuel weight are presented in figures 4(a) to 4(c).



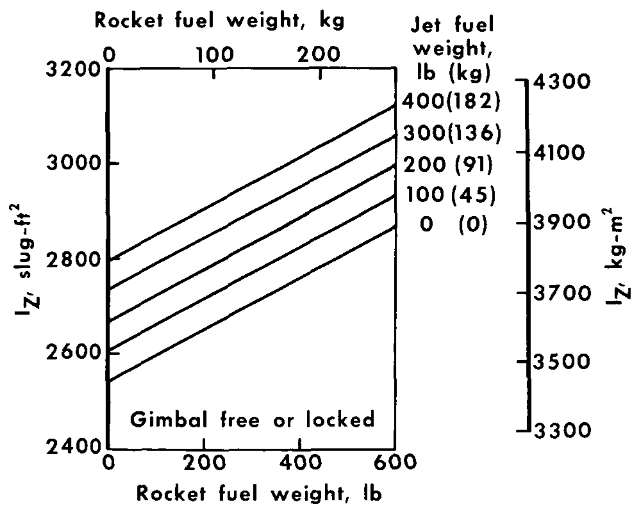
(a) Pitch.



(b) Roll.

COCKPIT

The cockpit of the LLRV is a platform about 6 feet (2 meters) above the ground extending forward from the main frame and between the two forward legs of the vehicle. The cockpit is completely open on the front, top, and back. An aluminum and Plexiglas screen on each side protects the pilot from the exhaust of the forward attitude-control rockets. The sole occupant is the pilot. His equipment consists of an instrument panel, a side console, control sticks, an ejection seat, and breathing oxygen.

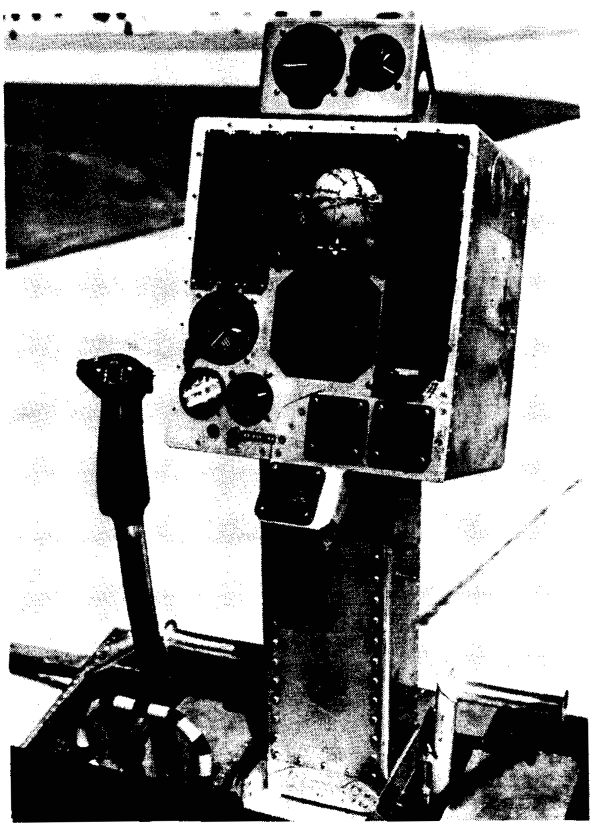


(c) Yaw.

Figure 4.– LLRV moment-of-inertia variations with fuel weight for a 175-pound pilot.

Instrument Panel

Flight instruments, operational instruments, and a bank of annunciator lights are displayed to the pilot on the instrument panel (figs. 5(a) and 5(b)). Vertical-scale instruments present normal acceleration (calibrated in lunar g), vertical velocity, and altitude, and a three-axis ball displays attitudes and vernier indications of forward and side velocity on crosspointers. Below the attitude ball is a horizontal-velocity indicator, which shows longitudinal and lateral velocity. The flight instruments are augmented by a pressure altimeter, a climb and dive indicator, and a clock. For the jet engine, there is a gas-generator tachometer and an exhaust-gas-temperature indicator. In the rocket system, there are pressure indicators for the helium source gas, hydrogen-peroxide tank, and lift-rocket chamber. Annunciator lights show the status of oil pressure, jet-fuel level, local vertical mode, stabilization mode, engine maximum tilt, emergency gimbal locked, hydraulic pressure, automatic-throttle operation, jet-engine thrust, hydrogen-peroxide-fuel level, stuck-rocket valve, source pressure, automatic pilot, hydrogen-peroxide fuel unbalance, ac and dc power, and the Doppler radar.



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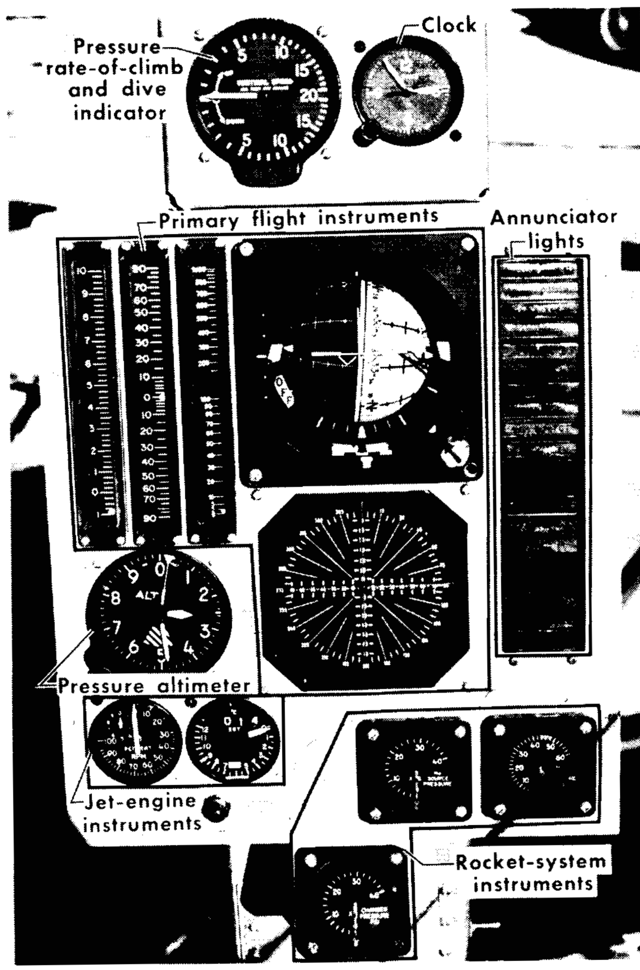
(a) General view.



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(b) Close-up.

Figure 5.– Pilot's instrument panel.

The vertical-scale instruments and the horizontal-velocity indicator were designed and fabricated at the Flight Research Center for the LLRV. The accelerometer (fig. 6) is actuated by signals from the basic vehicle accelerometers. Full-scale indications range linearly from -1 to 10 lunar g, and the instrument is readable to 0.2 lunar g. The vertical-velocity indicator (fig. 7) is operated by tachometer-generator signals of the radar altimeter. Scales range over ±90 ft/sec and are nonlinear in order to be readable to 2 ft/sec near touchdown. The height indicator (fig. 8) is also driven by the radar altimeter. Maximum height indication is 1000 feet and proceeds nonlinearly to zero altitude, at which the instrument is readable to 2 feet of altitude. The nonlinearity of the scale is quite apparent to the pilot at altitudes below 200 feet. The horizontal-velocity indicator (fig. 9) derives its signals from a Doppler radar unit. With full-scale readings of ±70 ft/sec, it is readable to 5 ft/sec. For readings of less than 10 ft/sec, the crosspointers on the three-axis ball are used. The horizontal needle indicates

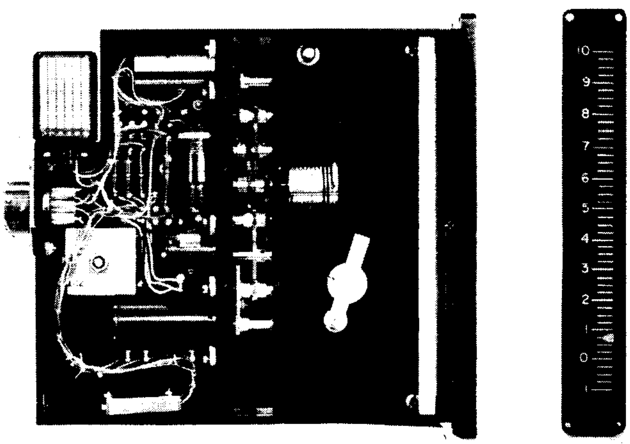

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Figure 6.– Normal-acceleration indicator.

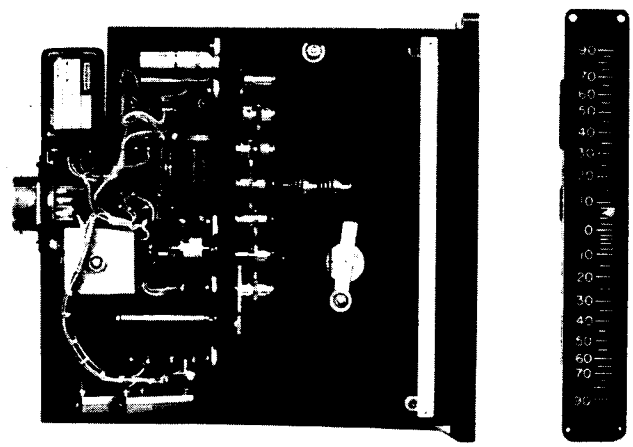

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Figure 7.– Vertical-velocity indicator.

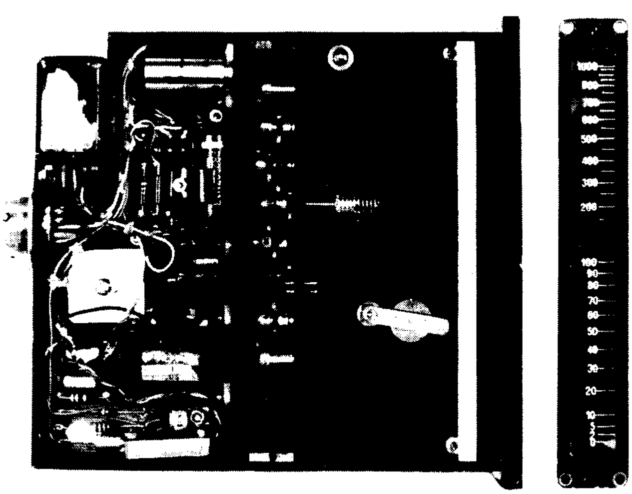

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Figure 8.– Radar-altimeter height indicator.

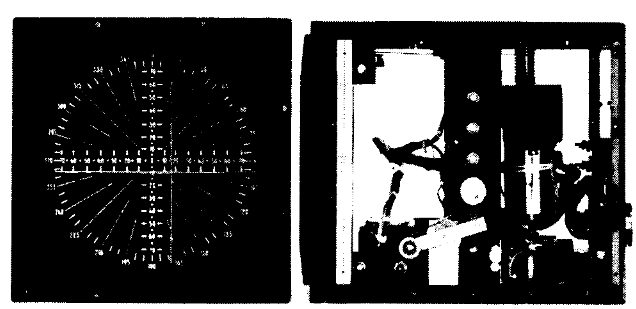

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Figure 9.– Doppler radar velocity indicator.

longitudinal velocity, with up being forward. The vertical needle indicates lateral velocity, with right being velocity to the right. Intersection of the two needles in any quadrant of the indicator is a vector-sum indication of velocity magnitude and direction relative to the vehicle body-axis system.

Side Console

A side console with operational switches, circuit breakers, a radio, and the jet throttle is at the pilot's left (fig. 10). All the equipment on the console is within easy reach of the pilot in flight, even with the shoulder harness secured. Operational switches for the electrical distribution system and protecting circuit breakers, rocket-system selection and operation switches, a jet-engine ignition switch, and electronic selection and operation switches are installed on the console. From this console, the pilot can adjust the control-stick and rudder-pedal sensitivity and set the initial heading for yaw-angle command operation. The switch for deploying the drogue parachute to reduce terminal velocity during emergency recovery from altitude is also on the console. In the rear portion of the console is a four-channel, lightweight, UHF radio transmitter and receiver for communications between the pilot and the ground.

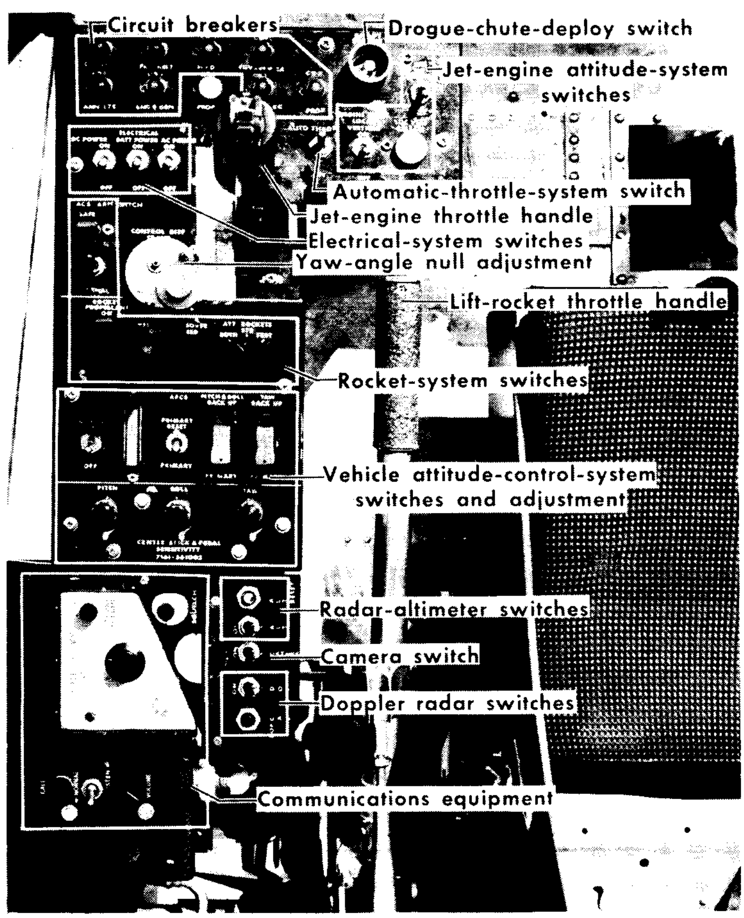

Figure 10.– Pilot's side console.

Controllers

The pilot controls the LLRV with three control sticks and a set of foot pedals. The jet-engine throttle (fig. 11), as mentioned previously, is mounted conventionally on the side console and has a maximum travel of 48° (0.84 rad). Once the throttle is advanced, a detent prevents power from being reduced below idle (48-percent rpm) without releasing the detent, thus, preventing inadvertent shutdown in flight. A closed, temperature-compensated hydraulic system links the jet throttle with the jet-engine master fuel-control valve. Also, a "fly-by-wire" arrangement links the throttle handle with an automatic-throttle actuator, which gives the pilot an electric backup to the hydraulic system. A magnetic clutch in the automatic throttle allows the pilot to override the automatic throttle whenever it is in operation. On the top of the throttle grip is a two-position switch for activating or deactivating the "fly-by-wire" throttle operation.

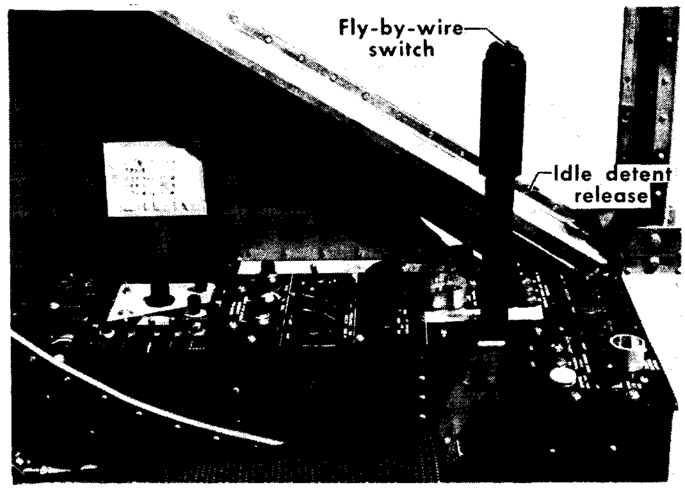

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Figure 11.– Jet-engine control handle.

Between the side console and the pilot's seat is a collective-type stick for lift-rocket operation (fig. 10). This stick is mechanically linked to the rocket-throttle valves, as can be seen in figure 12. The first 17.5° (0.306 rad) of stick travel control the normal lift rockets, and the next 17.5° (0.306 rad) control the emergency lift rockets. Stick forces increase abruptly during the second 17.5° (0.306 rad) of travel to preclude inadvertent use of the emergency lift rockets. The lift-rocket stick has no force gradient; however, stick friction

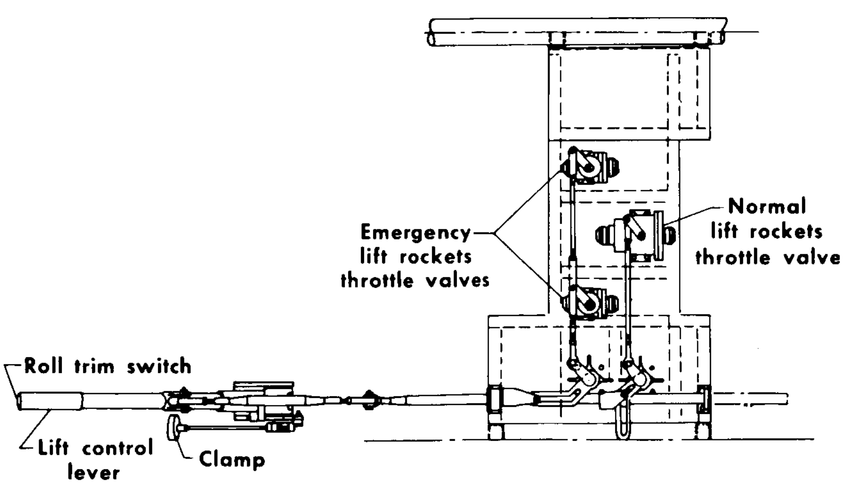


Figure 12.– Throttle layout of lift rockets.

is adjustable from the cockpit through the clamp, and, ideally, the lever will remain in any position, hands off, without excessive breakout forces. A three-position switch, spring-loaded to the middle or neutral position, is on the end of the lift-stick hand grip and allows the pilot to draw hydrogen peroxide from only one of the two tanks during lift-rocket operation for lateral center-of-gravity control.

A conventional center stick is fitted with synchro pickoffs which provide proportional electrical signals to command motion in the pitch and roll directions through the electric attitude-control system. Stick forces may be varied by the use of different bungees. Simulation studies on electronic computers indicate that the pilot will prefer low forces. Plots of the stick forces used during initial flights in the LLRV are shown in figures 13 and 14. A radio microphone button and an emergency gimbal-lock switch are available to the pilot on the stick grip (see fig. 5(a)).

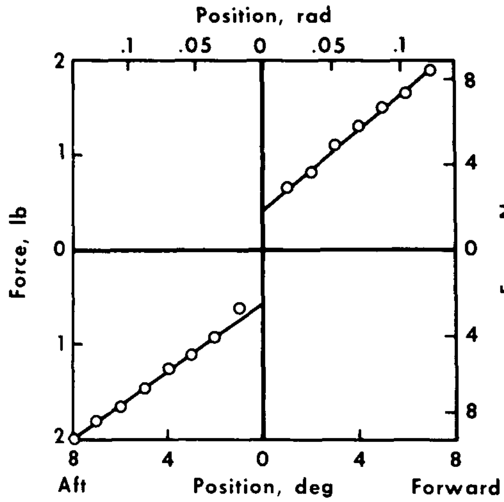


Figure 13.– Longitudinal stick force as a function of stick position.

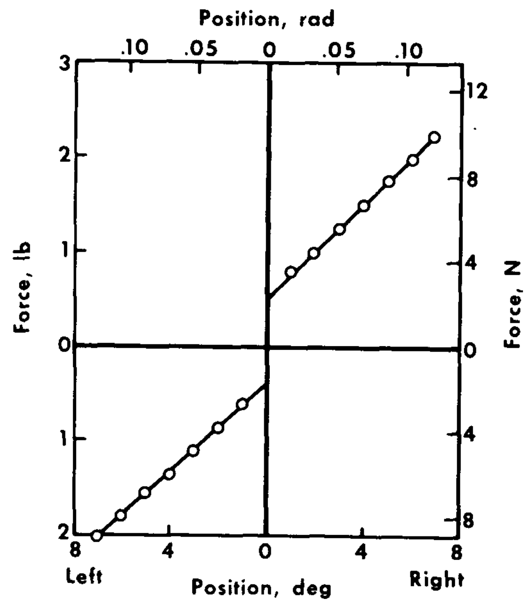


Figure 14.– Lateral stick force as a function of stick position.

Pedals for the control of yaw provide proportional electrical signals to the vehicle primary electronics and on-off signals to the backup electronics. As with the center stick, variable-bungee provisions are available. A plot of the pedal forces is shown in figure 15.

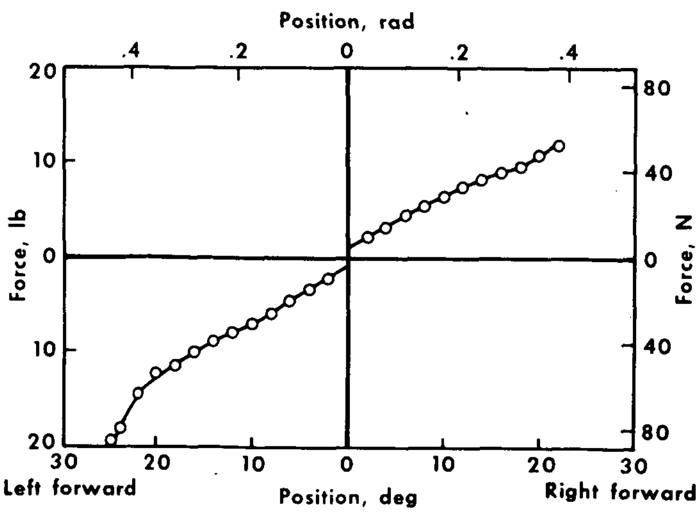


Figure 15.– Pedal force as a function of pedal position.

Seat

The Weber ejection seat (fig. 16) is a lightweight, rocket-propelled unit which has been demonstrated to have ground-level and zero-velocity ejection capability. A three-point release, integrated parachute restraint harness is provided on the seat. With this seat, the pilot has the option of leaving the parachute in the seat bucket when egressing on the ground, of ejecting by using the rocket capability, or of bailing out over the side without the use of the rocket. To initiate the rocket sequence, the pilot pulls a single D-ring handle on the front of the seat; to bail out without using the seat he uses a "ditching" handle on the right side of the seat; and to make the normal ground egress he unsnaps the seat belt and shoulder restraints. The seat does not have an inertia reel, so the pilot is restricted in his shoulder movement throughout the flight. This feature necessitated extra care in positioning the controls and switches.

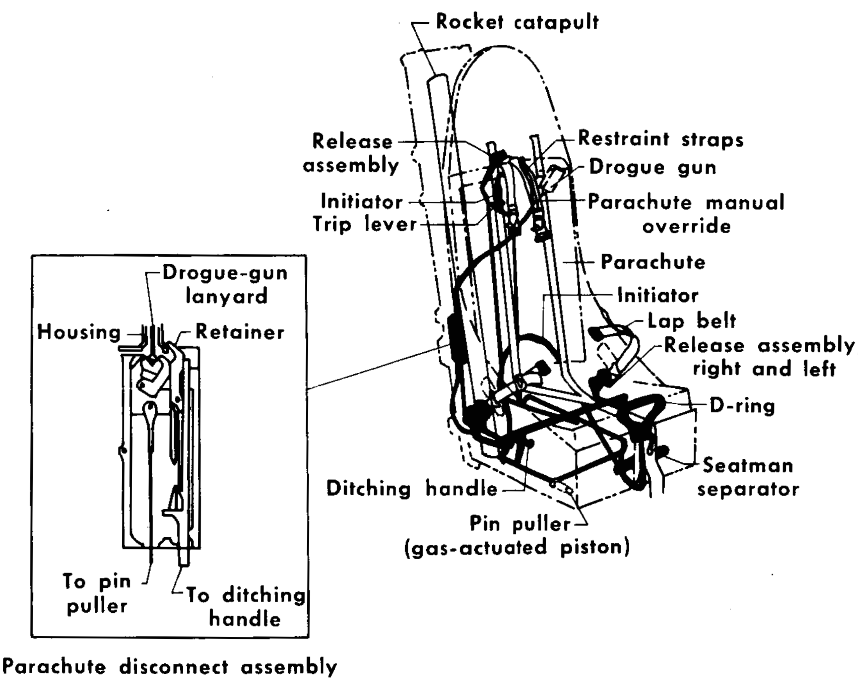


Figure 16.– Schematic drawing of ejection seat.

Breathing Oxygen

The pilot is provided with breathing oxygen during flight to avoid the possibility of inhaling fumes from the jet engine or rocket exhaust. The oxygen system, on the pilot's compartment floor to the right of the seat, consists of a supply tank, shutoff valve, demand-type regulator, pressure gage, and hose which connects to a T-block on the pilot's harness. When fully charged, the tank contains breathing oxygen at 1800 psi (1240 N/cm^2) and is designed for 15 minutes of use at demand flow. The pressure gage is integrally mounted on the front of the tank and is visible to the pilot in flight.

Radars

As indicated previously, the radar units on the LLRV provide the pilot and the research instrumentation with the primary on-board altitude and velocity information. All principal equipment for the radars, such as the antennae, power supplies, transmitters, and receivers, are on the instrument platform at the aft end of the vehicle.

The Doppler radar, which provides longitudinal- and lateral-velocity information, is basically a Ryan AN/APN-97A unit. The unit operates at a frequency of 13,300 Mc with 1 watt of power. Tests of the radar in a helicopter have indicated that, although the system operates as designed above normal terrain, performance over any relatively smooth surface deteriorates rapidly with altitude. Therefore, additional signal preamplification was incorporated into the system to keep velocity errors to less than 5 ft/sec (1.5 m/sec) over smooth surfaces.

The radar altimeter is a Radcom Model 1180 unit. Operating up to altitudes of 3000 feet (900 meters), it has an accuracy of ±5 percent above 40 feet (12 meters) and ±2 feet (±0.6 meter) at altitudes of less than 40 feet (12 meters). Its rf frequency is 4300 Mc at 0.2 watt of power. A tachometer-generator signal from the altimeter provides vertical-velocity information. Helicopter tests of this unit have proved that it performs as designed.

ELECTRONIC-CONTROL SYSTEMS

The major portion of pilot control of the LLRV is achieved through one of four electronic systems on the vehicle. The four systems are: the vehicle attitude-control system, jet-engine attitude-control system, jet-engine automatic-throttle system, and thrust/weight computer.

Vehicle Attitude-Control System

The vehicle attitude-control system is a redundant, "fly-by-wire" system which accomplishes all vehicle attitude changes. On-off rockets displaced from the vehicle center of gravity operate in pairs to produce pure force couples in the pitch, roll, and yaw planes.

Feedback of vehicle rate and attitude-angle signals allows the system to be mechanized for rate and angle command, as well as for acceleration command. Variable-control sensitivity, threshold, and hysteresis features are available. Figure 17 is a block diagram of the system and is typical of each of the three motion planes.

Sixteen attitude-control rockets are used. The rockets are divided into two separate systems, designated as the standard and the test sets. During flight the pilot may select either or both systems. Attitude-rocket layout, nomenclature, and possible combinations in the redundant modes to achieve the desired motion are shown in figure 18. Each rocket is adjustable on the ground

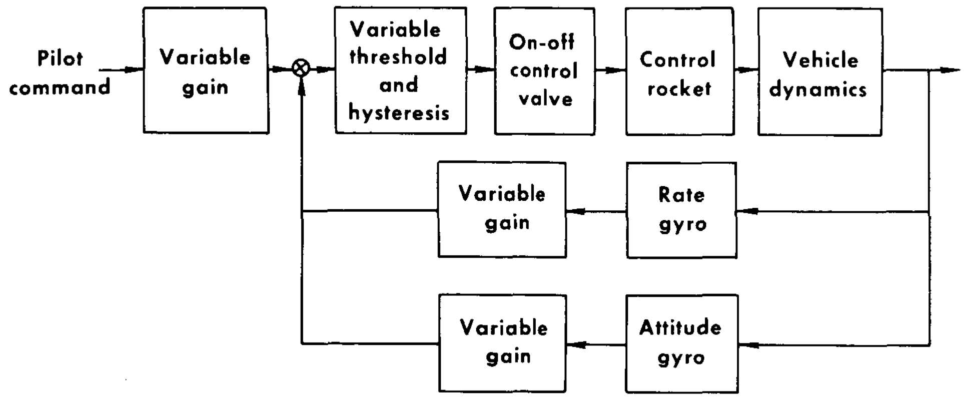
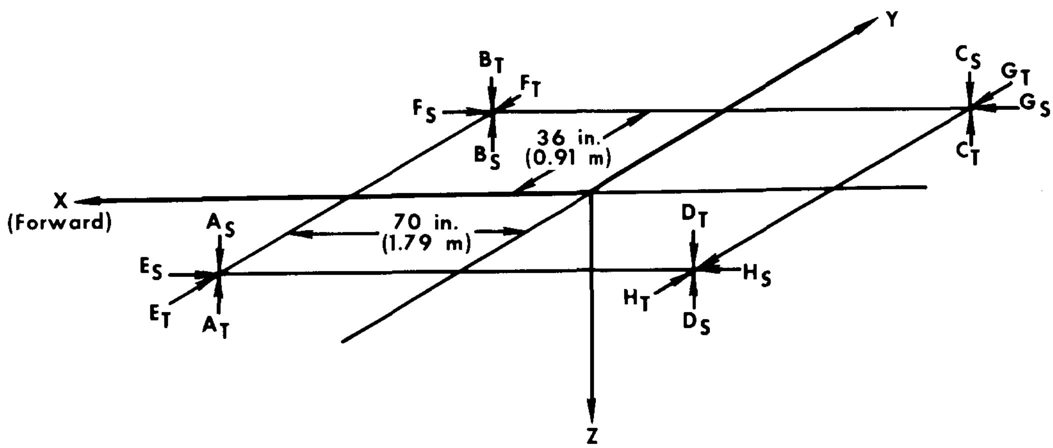


Figure 17.– Block diagram of vehicle attitude-control system.



Rockets with subscript S denote standard rockets fire.
Rockets with subscript T denote test rockets fire.
For dual-system operation, both standard and test rockets fire.

Rocket		A_S	A_T	B_S	B_T	C_S	C_T	D_S	D_T	E_S	E_T	F_S	F_T	G_S	G_T	H_S	H_T
Pitch	Up		✓	✓		✓			✓								
	Down	✓			✓		✓	✓									
Roll	Right		✓		✓	✓		✓									
	Left	✓		✓			✓		✓								
Yaw	Right										✓	✓			✓	✓	
	Left									✓			✓	✓			✓
Pitch up and roll right			✓			✓											
Pitch up and roll left				✓					✓								
Pitch down and roll right					✓			✓									
Pitch down and roll left		✓					✓										

Figure 18.– Attitude-control-rocket firing logic.

from thrusts of 18 pounds (80 newtons) to 90 pounds (400 newtons). The pitch rockets and the test yaw rockets are positioned 70 inches (1.79 meters) from the axis of rotation. The roll rockets and the standard yaw rockets are 36 inches (0.91 meter) from the axis of rotation. The standard rockets are adjusted to produce acceptable control sensitivity for the pilot. The test rockets are adjustable through the thrust range to investigate a representative interval of control authorities. To produce maximum control authorities, the two subsystems are actuated simultaneously.

By dividing the moments produced with the rockets by the appropriate inertias (see fig. 4), the control-authority levels shown in figure 19 are obtained. This figure shows that, by adjusting rocket thrust and vehicle weight, control effectiveness can be varied from 5 deg/sec^2 (0.087 rad/sec^2) to 57 deg/sec^2 (1.0 rad/sec^2) in pitch, from 7 deg/sec^2 (0.12 rad/sec^2) to 96 deg/sec^2 (1.67 rad/sec^2) in roll, and from 2 deg/sec^2 (0.035 rad/sec^2) to 40 deg/sec^2 (0.70 rad/sec^2) in yaw. The steady-state angular-rate limitation of 40 deg/sec (0.70 rad/sec) in pitch, roll, and yaw is a system limitation imposed by rate-gyro characteristics. These values cover the major portion of those considered suitable in the studies of references 7 and 8 as well as the higher values rated best for vehicles of the vertical takeoff and landing (VTOL) type (ref. 9).

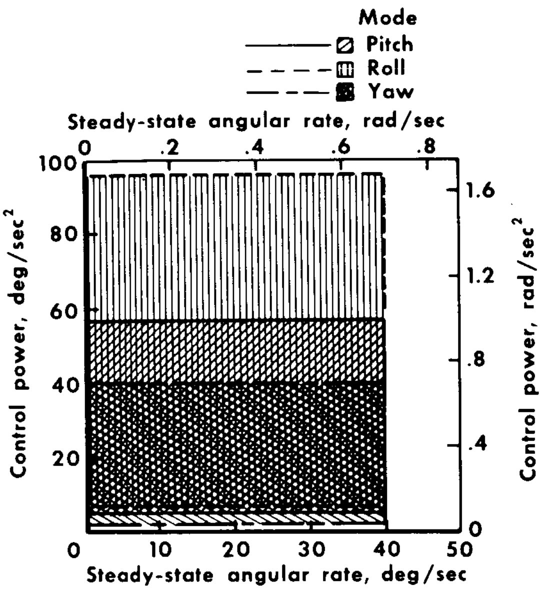


Figure 19.— Control-authority variability.

The major sensing elements used in the control system are indicated by asterisks in table II. These sensors are used by both the control system and research-instrumentation system. Of these elements, only the pitch and roll rate gyros and the stick-position potentiometers are redundant. For failures in the yaw mode, microswitches on the pedals provide a backup, on-off command signal.

Under normal conditions, the pilot flies the vehicle in the primary mode of the system electronics. In this mode, the system determines which attitude rockets should be fired as a result of pilot control motions and/or vehicle motions. It should be noted that any uncommanded angular motion, caused by factors such as aerodynamics or fuel unbalance, that is sensed by the vehicle is automatically canceled. The manner in which the vehicle is controlled is determined by control sensitivity, threshold, hysteresis, and feedback-gain setting. Of these quantities, only control sensitivity can be varied in flight. By proper adjustment of the feedback gains, the pilot can be provided with attitude command, rate command, or on-off acceleration command.

The system is also equipped with a backup mode in which the pilot commands pitch and roll rate and yaw on-off acceleration. Control sensitivity, threshold, hysteresis, and feedback-gain settings are all adjusted on the ground for the desired response under all anticipated flight conditions. This mode is used only if the primary mode fails, since it has no monitor or failure warning features.

A monitor subsystem parallels and checks the operation of the primary mode. When inconsistencies are detected in either the pitch and roll or the yaw mode, control is switched to the proper backup mode within 0.3 second. A warning light tells the pilot whenever a particular attitude rocket is being used excessively, which is indicative of a stuck valve. The pilot must manually select the standard or the test set of rockets to isolate the malfunctioning rocket.

In the primary mode, a detection subsystem is provided to sense excessive angular rates which would result primarily from hardover failures of the flight-control sensors. When an angular rate exceeds a ground-adjusted threshold level, the system switches control to the backup mode. In the roll plane, rocket-on time is compared to total time, averaged over a 4-second interval, to inform the pilot when disturbing moments are exceeding a ground-adjusted threshold level. When the preset level is exceeded, a warning light is illuminated and the pilot must determine if the cause is aerodynamic, the result of fuel unbalance, or the result of a stuck rocket. If the cause is aerodynamic, he will reduce speed to lessen the aerodynamic moments imposed on the vehicle. If it is caused by fuel unbalance, he will use rocket fuel from the heavy tank only until the situation is corrected. If the cause is a stuck rocket, he will switch to the proper set of rockets, as indicated previously.

Jet-Engine Attitude-Control System

The jet-engine attitude-control system operates a hydraulic servo-driven dual-gimbal arrangement which positions the jet engine in the proper direction, relative to the vehicle. The system can operate in two pilot-selected modes and in two other modes that may be selected by the pilot as well as automatically determined. The modes of operation are the gimbal-locked mode, local vertical mode, engine-centered mode, and the stabilization mode. These modes of operation exercise a priority over each other in the following manner: The gimbal-locked mode, when either pilot-selected or automatically selected, will override all other modes. When the vehicle is on the ground or when the jet engine is deflected more than 15° (0.26 rad) from the vertical, the local vertical mode is automatically actuated and overrides all modes except the gimbal-locked mode. To change from the local vertical mode to the engine-centered mode or jet-stabilization mode, the local vertical release button on the console must be depressed. Actuation of the jet-stabilization-mode switch will override the engine-centered mode. If no mode has been selected, the vehicle will operate in the engine-centered mode.

Gimbal-locked mode.- The gimbal-locked mode provides the pilot with an independent valve and pressure source to aline the engine with the vehicle Z-body axis. The mode can be selected manually. It will also be actuated whenever the jet engine exceeds an angle of 15° (0.26 rad) from the vertical and is not commanded to the local vertical mode by the time 0.5 second has elapsed. The hydraulic-system schematic on page 26 shows additional details of the mechanization of the gimbal lock.

Local vertical mode.– In the local vertical mode (fig. 20(a)) the jet engine is always alined with the vertical with respect to the earth as sensed by vertical gyros, regardless of vehicle attitude. When the vehicle is on the ground, microswitches on the vehicle shock struts make contact and dictate that the local vertical mode is in operation, unless the gimbal lock is on. This feature is incorporated primarily to reduce to a minimum landing tipover moments resulting from an engine "hard-over" command. In flight, the pilot can elect to fly with the engine in the local vertical mode. Also, whenever the jet engine deviates more than 15° (0.26 rad) from vertical, this mode is automatically established in order to prevent damage from improper lubrication of the bearings.

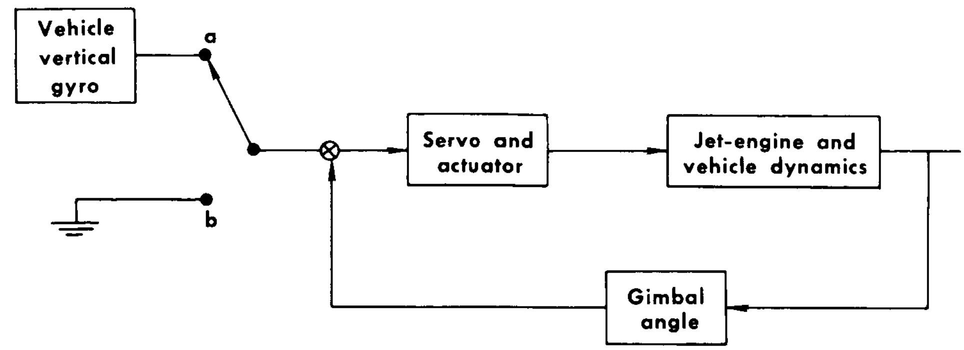
Engine-centered mode.– The engine-centered mode (fig. 20(a)) is a pilot-selected mode that alines the jet engine with the vehicle Z-body axis. This mode senses the gimbal angles and commands zero angular displacement relative to the vehicle frame, thus making the LLRV, in effect, a conventional VTOL vehicle. This mode could be used for takeoff and climb to a lunar-simulation starting point in flight.

Stabilization mode.– The stabilization mode is the mode used during lunar simulation. As shown in figure 20(b), this mode utilizes a command system which senses acceleration in the horizontal plane (both longitudinal and lateral), compares it with the commanded lunar value resulting from lift-rocket operation, and tilts the jet engine as required to null the summed signals. This mode thus serves to develop the proper linear lunar acceleration in the horizontal plane. Characteristics of the sensing elements are given in table II.

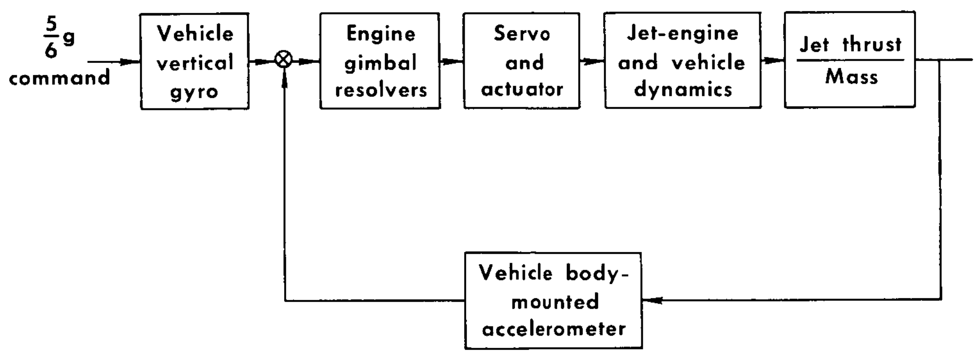
Jet-Engine Automatic-Throttle System

The jet-engine automatic-throttle system (fig. 21) performs the same function in the vertical plane as the jet-engine attitude system does in the horizontal plane for the lunar-simulation mode. As shown in the figure, in this instance, the jet-engine throttle modulates thrust to provide the proper vertical acceleration for lunar simulation. By actuating both the jet-stabilization-system switch on the left console and the lift rockets to initiate the weighing operation (described in the following section), the pilot commands that the jet engine automatically support five-sixths of the vehicle weight so that he can then concentrate on lift-rocket operation.

High- and low-thrust limit switches are incorporated in the circuitry to preclude the possibility of the automatic system commanding thrust above or below preset values. When these limit switches are actuated, no further change in jet-engine thrust will occur.



(a) Local vertical mode (switch is in position a) engine centered mode (switch is in position b).



(b) Stabilization mode (lunar simulation).

Figure 20.— Block diagram of the jet engine attitude-control system (typical for pitch and roll axes).

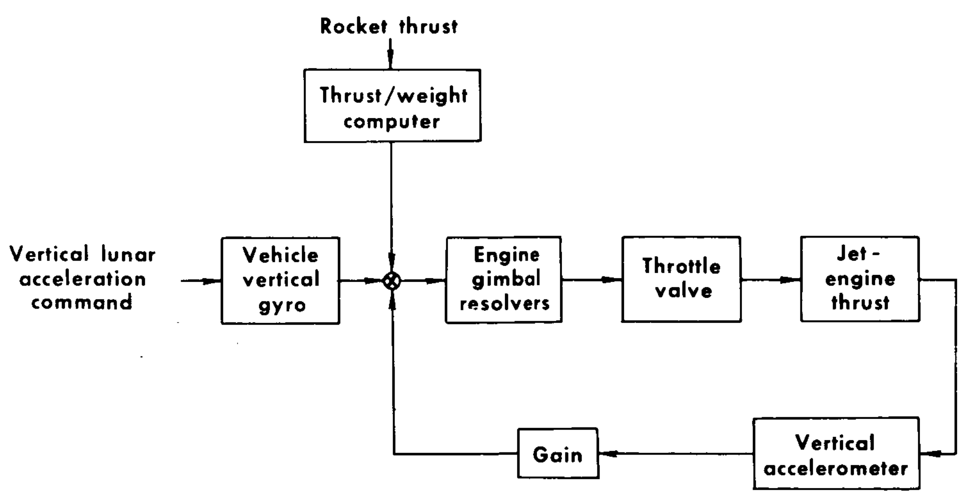


Figure 21.— Block diagram of automatic-throttle control system.

Thrust/Weight Computer

The thrust/weight computer (fig. 22) is essentially a timer to compute weight change during lunar simulation. The greatest complication to the system is an in-flight weighing procedure which determines an initial weight for the computer.

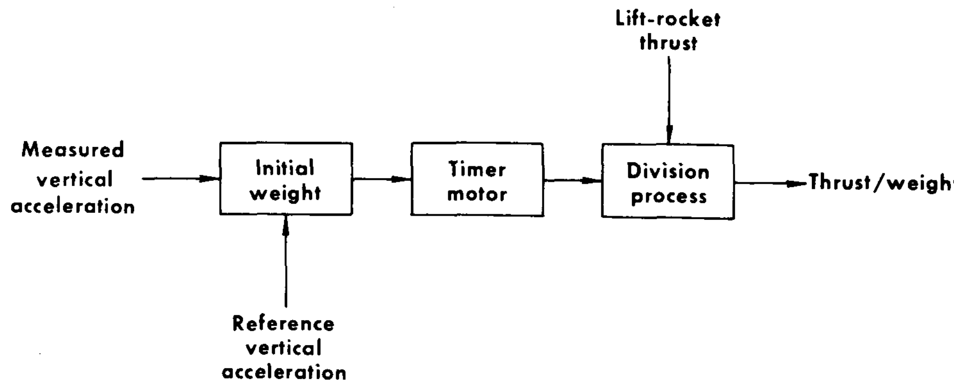


Figure 22.– Block diagram of thrust and weight computer.

A microswitch on the lift-rocket throttle is preset to open at a given low-thrust level. When the microswitch is tripped, vertical acceleration is measured, giving an initial weight, and a relay starts the timer which runs for 150 seconds. Once the timer is actuated, the computer continuously determines the vehicle weight on the basis that the jet engine uses 33 pounds (15 kilograms) of jet fuel each minute and that the lift rockets use 300 pounds (136 kilograms) of hydrogen peroxide each minute. Since, in a lunar simulation, the jet engine supports five-sixths of the vehicle weight and since fuel consumption is relatively constant, little error can result. The lift rockets are assumed to support one-sixth of the vehicle weight (just enough thrust to hover).

VEHICLE ELECTRICAL SYSTEM

Direct-current and alternating-current electrical power are required on the LLRV. Power sources, power distribution, and equipment operating from each power source are shown in figure 23.

The primary source of power for the entire vehicle is a 165-ampere, 28-volt generator driven by the jet engine. As can be seen in the figure, under normal conditions all dc equipment is operated by the generator. This dc power source also operates two ac inverters: an inverter for normal ac power, and an inverter for emergency ac power. The normal ac inverter is rated for 115-volt, three-phase, 400-cps, 1800-volt-ampere operation and powers the primary flight control system, the research instrumentation, and the pilot's flight-display instruments and sensors (including the Doppler radar and radar altimeter). The emergency ac inverter produces 115-volt, three-phase, 400-cps power for a maximum of 100-volt-ampere output and powers the backup flight-

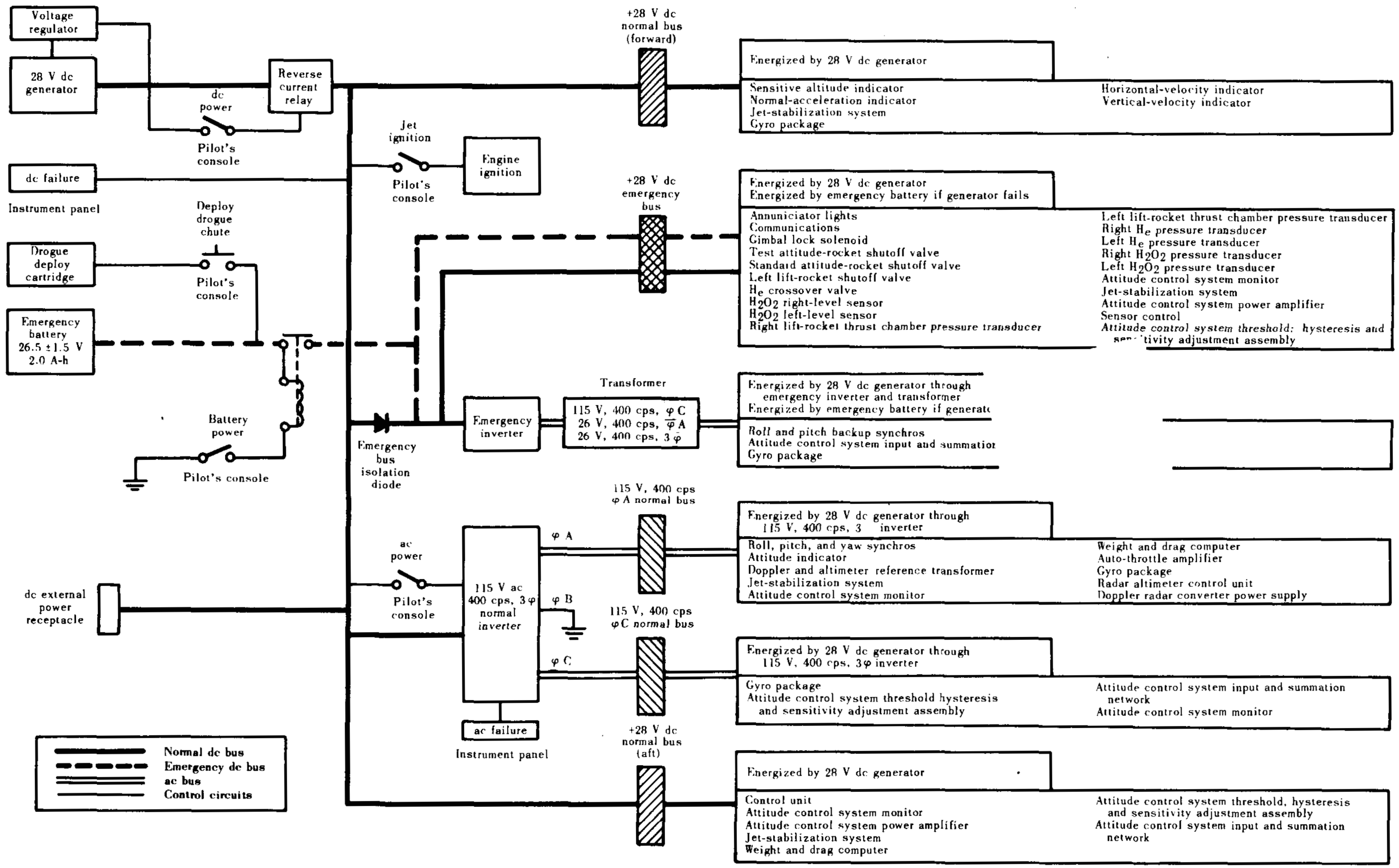


Figure 23.– Power-distribution diagram.

control system. For both inverters, voltages should be within 2 volts and frequency within 5 cps of the rated values.

Should the dc generator fail, a relay closes and dc power is supplied by a 26-volt, 2-ampere-hour battery. The battery also drives the emergency ac inverter. A diode insures that the battery powers only the emergency electrical system. Quantities that are essential for flight operation are on the emergency electrical system, such as rocket-system valves, hydraulic servo actuation, backup electronics, and drogue-parachute initiator (see fig. 23).

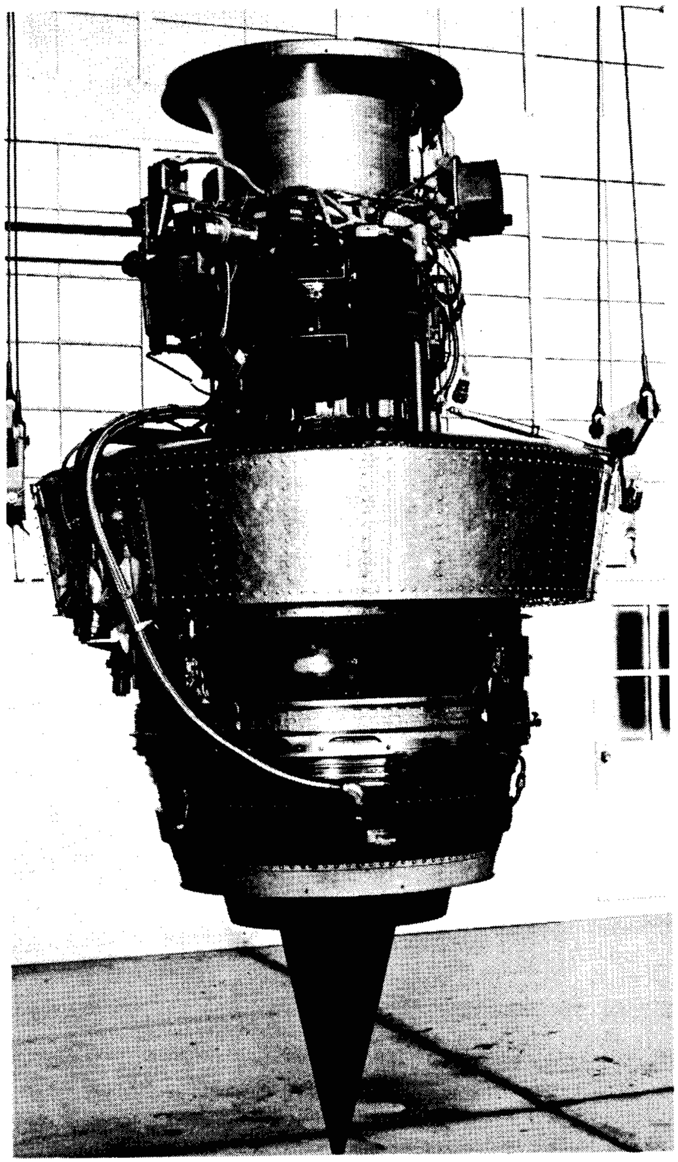
JET-ENGINE SYSTEM

The principal propulsion system of the LLRV is a General Electric CF700-2V turbojet engine mounted in a gimbal ring at the center of gravity of the vehicle. This engine provides thrust to lift the vehicle to altitude and to establish velocity conditions for a maneuver. During a lunar-simulation maneuver, the jet engine provides thrust to compensate for the excessive earth gravitational force. This thrust is adjusted in magnitude to compensate for vertical aerodynamic drag, and the thrust vector is tilted slightly from true vertical to compensate for horizontal drag.

The CF700 engine is essentially a nonafterburning J85 military jet engine, to which an aft fan is attached. The addition of the fan increases the maximum thrust of the basic engine from 2800 pounds (12,450 newtons) to 4200 pounds (18,680 newtons) and decreases the specific fuel consumption from about 1.0 lb/hr-lb (0.028 g/sec-N) to less than 0.7 lb/hr-lb (0.020 g/sec-N). However, the airflow through the engine is increased by a factor of three, and the engine weight is increased from 325 pounds (147 kilograms) to 629 pounds (285 kilograms). The fan is not connected mechanically to the gas-generator rotor but is driven by its own turbine formed by the root section of the fan blades (bluckets). General views of the engine are shown in figures 24(a) and 24(b) and a cutaway view in figure 25. Physical characteristics of the engine are presented in table III, and a detailed description is given in reference 10.

The lubrication system of the -2V version of the CF700 engine was specially modified for the vertical operation required by the LLRV. The engine may be operated indefinitely when vertical or inclined up to 10° (0.15 rad) from the vertical, operated up to 60 seconds when inclined up to 15° (0.22 rad) from the vertical, operated in any other position and at zero or negative accelerations for periods up to 10 seconds.

Three of the -2V versions have been constructed for the LLRV program. Figure 26 shows performance data obtained during the final acceptance tests of these engines. The engines have an extremely low value of gyroscopic moment, since the fan rotates in the opposite direction from that of the gas-generator rotor. The fan has the greater moment, so the net effect is in the direction dictated by the fan rotation.



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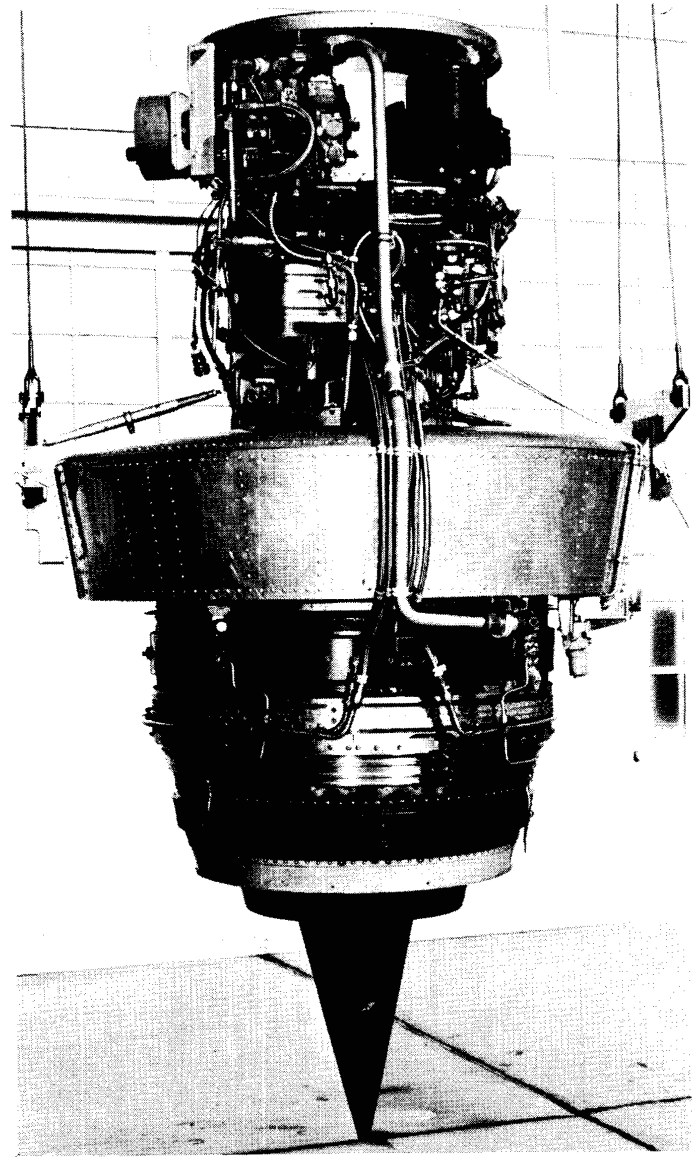
(a) Right side.



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(b) Left side.

Figure 24.– Photograph of CF700 turbofan engine showing accessories installed.

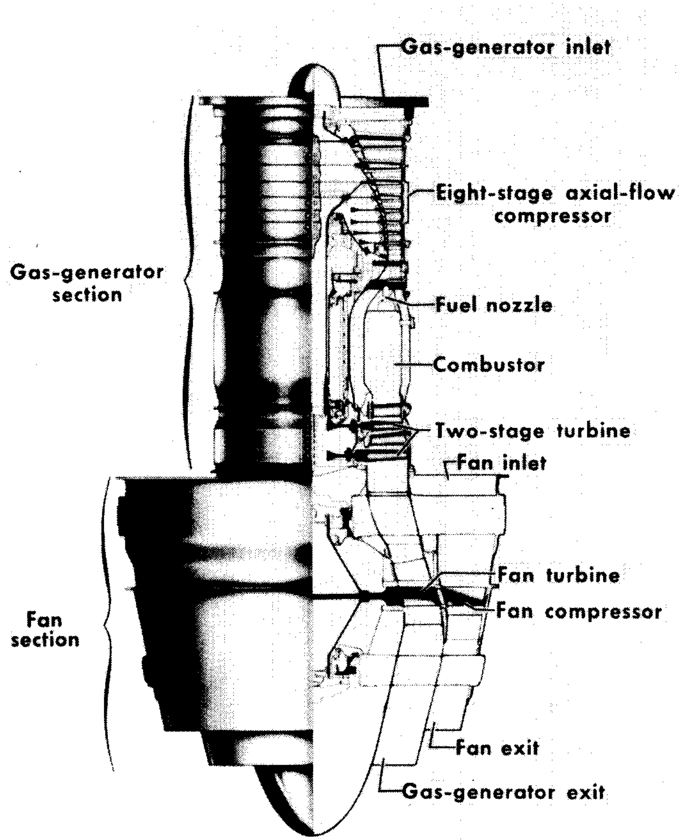


Figure 25.– Cutaway view of CF700-2V turbofan engine.

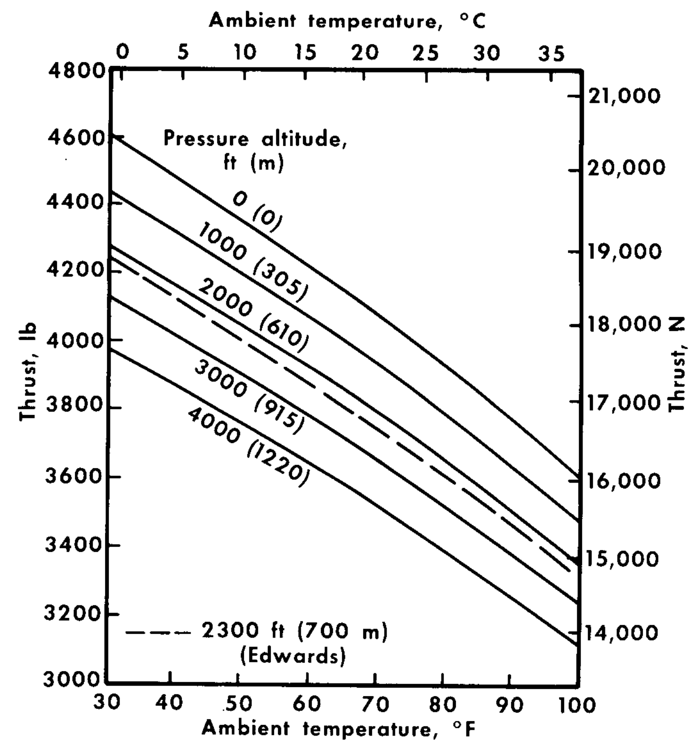


Figure 26.– Variation in thrust of the CF700-2V jet engine with ambient temperature and altitude.

Mounted on the turbojet engine, and connected by means of a mechanical drive, is a 5-kVa, 28-volt, dc electrical generator which provides the primary electrical power source for the vehicle. A 3-gal/min (190-cm^3/sec), 3000-psi (2070-N/cm^2) hydraulic pump is similarly driven by the engine accessory drive to provide hydraulic power to the engine gimbal-actuation and locking system.

The engine is designed to use either JP4 or JP5 fuel, having a lower heating value of at least 18,400 Btu/lb (42,700 J/g). Engine starting is accomplished by compressed air, injected through an integrally mounted impingement air horn. Starting air requirements are 100 lb/min (0.75 kg/sec) at 43 psi (30 N/cm^2) and 360° F (456° K) for a minimum period of 20 seconds for each start.

The jet-engine fuel system is shown schematically in figure 27. Fuel is supplied from two spherical tanks mounted on the main structure of the vehicle ahead of and behind the engine. A Sterer flow proportioner insures that equal amounts of fuel are withdrawn from the two tanks. Before the engine is started, the tanks are pressurized to 40 psig (27.6 N/cm^2) by gaseous nitrogen from a ground supply. During flight, the pressure is maintained at about 22 psig (15 N/cm^2) by bleed air from the fifth compressor stage. There is no fuel-rate meter, but two sensors in each tank give the pilot a caution light when about 200 pounds (90 kilograms) of fuel remain and a low-level warning light when about 80 pounds (36 kilograms) of fuel remain. A self-contained hydraulic system links the pilot's throttle lever with the fuel control on the jet engine. An electrical servo control unit is also mounted on the jet engine to adjust the throttle when the automatic-throttle control system is in operation. A slip clutch permits the pilot to override the electrical servo in an emergency.

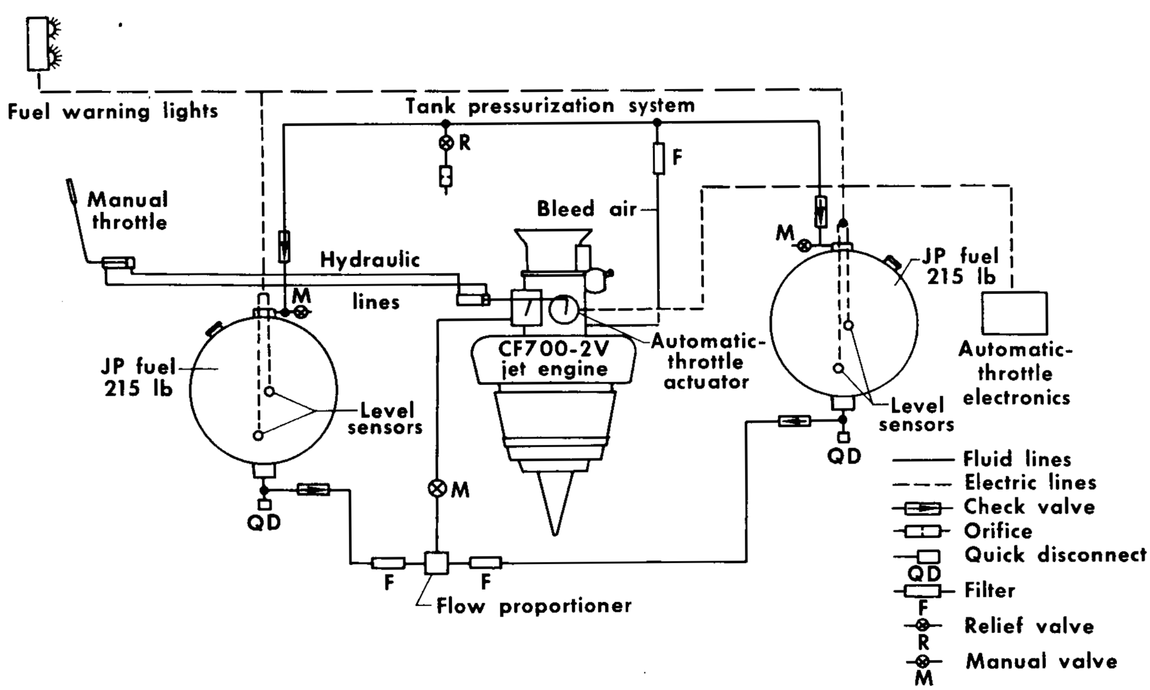


Figure 27.— Jet-engine fuel system.

The attitude of the jet engine with respect to the vehicle frame is adjusted and maintained by means of hydraulic actuators in accordance with electrical signals from the automatic control system. The hydraulic system is shown schematically in figure 28. The hydraulic pressure is maintained at

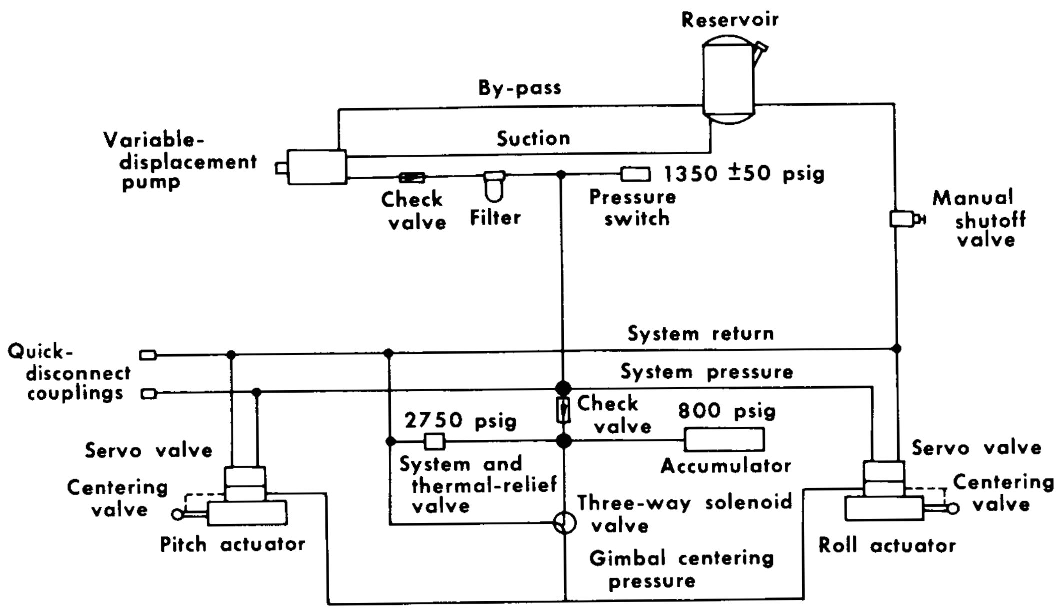


Figure 28.– Hydraulic-system schematic. System pressure, 2400 psi (1655 N/cm^2); hydraulic fluid, MIL-H-5606; pump displacement, 0.10 $in.^3$/revolution; pump speed, 7088 rpm maximum; 15-micron full-flow filtration; relief-valve cracking pressure, 2750 psi (18.96 N/cm^2); flow 2.0 gal/min (0.126 liter/sec) at 7088 rpm.

2400 psig (1655 N/cm^2) by means of a pump mounted on the jet engine. There are also provisions for connecting an auxiliary hydraulic source for ground operations when the jet engine is not operating. Both the pitch and roll actuators have an automatic-centering feature. Each actuator has a third hydraulic connection in addition to the normal supply and return ports. When pressure is applied to this third port, the actuator moves to and remains at its mid position, regardless of the primary-system pressure or electrical signals. Pressure to this centering port is provided by an accumulator; thus, the pilot always has the capability of centering the engine, in the event of a hydraulic- or electronic-system malfunction, by moving a thumb switch on the attitude control stick.

ROCKET SYSTEM

The general arrangement of the rocket system for the LLRV is shown schematically in figure 29. The system consists of sets of 500-pound-thrust (2224 newtons) lift rockets and sets of 90-pound-thrust (400 newtons) attitude-control rockets. Cutaway views of the rockets are shown in figure 30. Both types of rockets use the monopropellant, 90-percent hydrogen peroxide from a common supply system. The propellant is fed by pressurizing the main-propellant tanks with helium. Physical characteristics of the rocket system are presented in table IV.

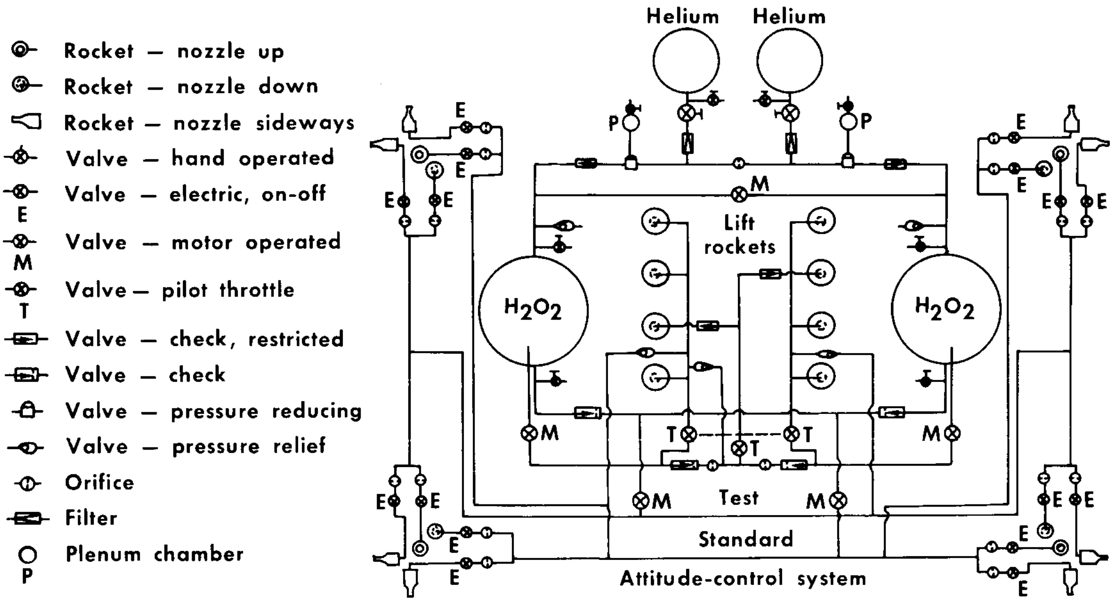


Figure 29.— Schematic drawing of rocket system.

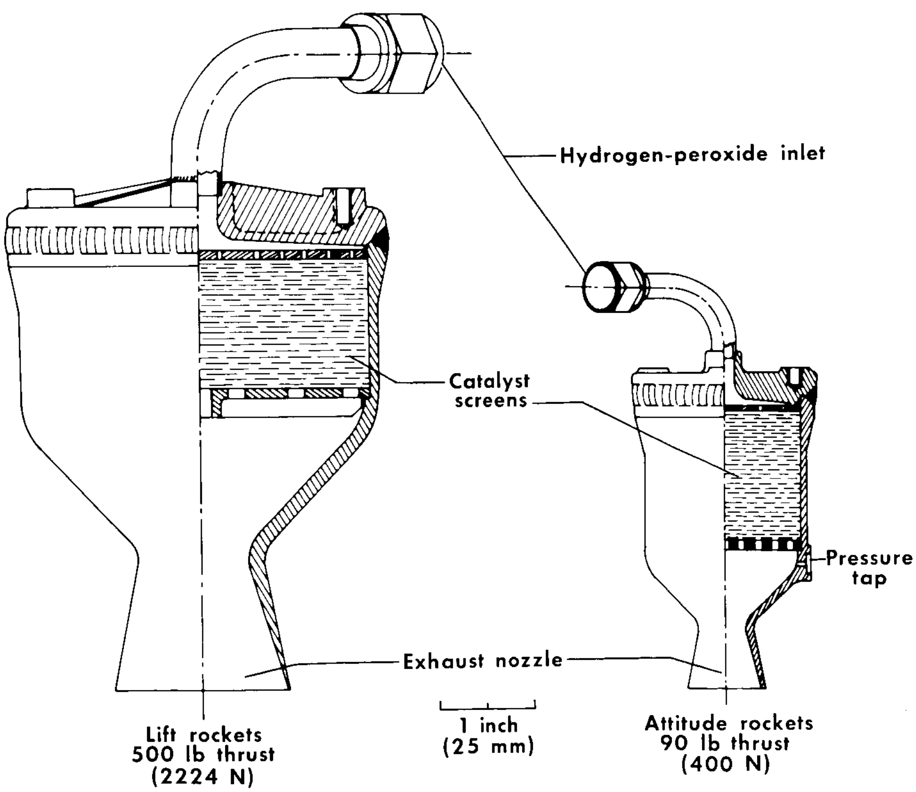


Figure 30.— Cutaway view of lift and attitude rockets.

Lift Rockets

Both vehicles carry two normal and six emergency lift rockets, each with a maximum thrust of 500 pounds (2224 newtons). The thrust of each rocket may be varied at the discretion of the pilot to a minimum of about 100 pounds (454 newtons). The rockets are rigidly mounted on the fixed-frame member that forms the outer portion of the jet-engine gimbal ring. They are always operated in pairs spaced about the center of gravity so that there will be no objectionable moments. Control is by means of a collective-type stick, described previously (page 11). It is expected that the emergency rockets will be used only if the jet engine should fail. Performance data from a typical lift rocket are shown in figure 31. The maximum life of the catalyst beds is estimated at about 2 hours; up to 5-percent loss in performance and efficiency can be expected during this period.

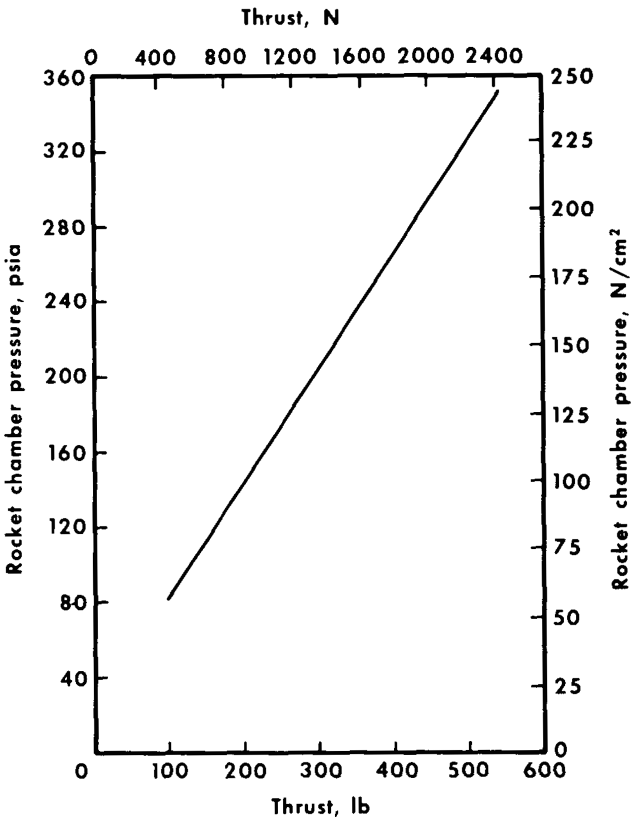


Figure 31.– Variation of lift rocket thrust with chamber pressure.

Attitude Rockets

The 16 attitude rockets on each LLRV are mounted in clusters of four on the ends of struts projecting from each side of the cockpit and instrument platforms. The clusters are located farther from the pitch axis than from the roll axis in order to compensate for the unequal moments of inertia about the two axes. Each attitude rocket is controlled by an individual, on-off type solenoid valve, and the thrust level may be preset before flight to any desired value between 18 pounds (80 newtons) and 90 pounds (400 newtons). Only eight rockets are required to produce plus and minus moments about all three vehicle axes. The 16 rockets are divided into two separate systems, designated the standard and the test sets, and the pilot during flight may select either or both systems. Since the rocket thrust levels cannot be adjusted in flight, the standard set will be kept at thrust levels giving good handling qualities, and the test set adjusted in accordance with the research program. Figure 32 presents a typical performance curve for the attitude rockets. The expected operational life of the rockets is 2 hours, the same as that of the lift rockets. A performance deterioration of up to 5 percent may take place during this time.

Propellant System

The components of the propellant feed system were shown in figure 29 (page 27). Except for the rockets, all metal in contact with hydrogen peroxide is stainless steel. Experience with both the X-15 (ref. 11) and the Mercury programs showed that there would be fewer operational problems with this metal than with other commonly used metals.

The hydrogen-peroxide fuel is stored in two 23.35-inch-diameter (59.3 centimeters) spherical tanks made from AM350-type stainless steel. The maximum fuel load at takeoff is 628 pounds (285 kilograms); combined, the two tanks hold slightly more fuel to allow for preflight checks of the rocket system. A set of orifices and restricted backflow check valves are designed to equalize the flows from the two tanks and, thus, maintain the lateral balance of the vehicle. The pilot can adjust any unbalance by momentarily closing the outlet valve on either tank by means of a switch on the lift-rocket control stick.

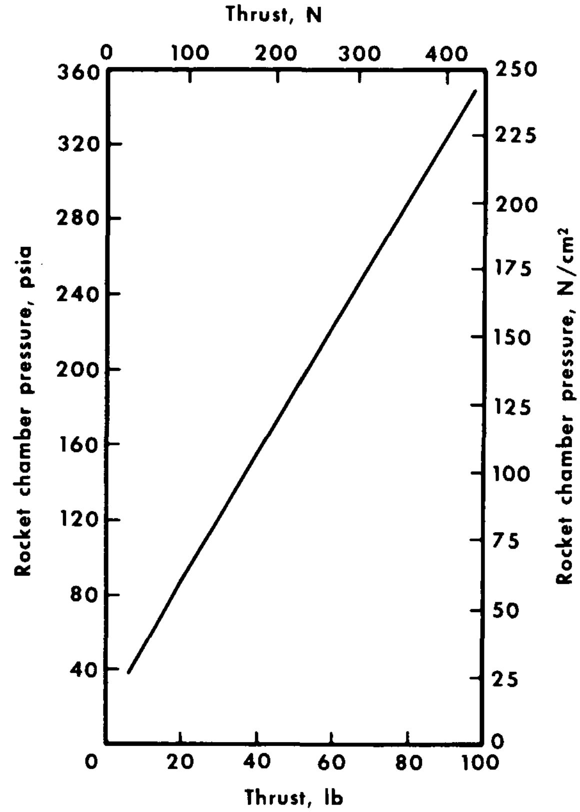


Figure 32.– Variation of attitude-rocket thrust with chamber pressure.

Helium pressurizing gas is stored in two 15-inch-diameter (38 centimeters) titanium spheres at an initial pressure of 3750 psig (2588 N/cm^2). Pressure regulators reduce the tank pressure to the required tank-pressurization level of 488 psia (337 N/cm^2). The two essentially separate pressurizing systems are connected through an orifice upstream of the regulators and a motorized valve downstream. These connections insure equal flow from the helium tanks and equal pressurization of the hydrogen-peroxide tanks. This latter condition is necessary to insure equal propellant flows from the two tanks. In the event of a major pressure leak, the pilot can close the motorized valve and effect an emergency landing using the jet engine without losing attitude control and without intolerable lateral unbalance. The 0.025-inch-diameter (0.63 millimeter) orifice upstream of the regulators is too small to permit a disastrous loss of pressure on the "good" side of the system, even if the pressure on the other side drops to nearly zero.

Other valves in the propellant lines, in addition to the previously mentioned four electrically operated valves controllable by the pilot, are the rocket control valves, electric solenoid valves for the attitude rockets, and mechanical ball valves for the lift rockets. There is a possibility that hydrogen peroxide trapped between the two sets of valves will decompose and build up excessive pressure. To prevent damage from such an occurrence, three relief valves, set to open at 650 psi (448 N/cm^2) have been installed as shown in figure 29. Hydrogen peroxide from these valves passes through the emergency lift rockets so that it will be decomposed before being discharged from the vehicle. Similarly, a relief valve set to open at 560 psi (386 N/cm^2) is located on the gas side of each propellant tank.

Under conditions of high rocket thrust, the pressure losses in the rocket system are considerable. These losses result from the sizes of the line and components, which were selected to keep the system weight to a minimum. Table V shows the pressure, computed by the vehicle manufacturer, in various parts of the system for three operational conditions. It should be noted that at the high flows of the emergency mode it is not possible to obtain 500 pounds (2224 newtons) of thrust from the lift rockets. Maximum total thrust of the lift engines is 3752 pounds (16,700 newtons) rather than 4000 pounds (17,800 newtons). Under these flow conditions, the maximum thrust of the attitude rockets drops about 7 pounds (31 newtons).

STRUCTURES

The structural frame of the LLRV serves to position and carry loads between the various vehicle components. Because of the limited thrust of the jet engine, which must lift the entire vehicle, the frame was made as small and as lightweight as practical. There were three basic considerations for the design of the frame: the jet engine in its gimbal ring must be held vertically so that the jet exit is about two diameters above the ground to avoid significant back pressures; the landing pads must have sufficient spread to give adequate landing stability and to clear the jet exhaust; and the vehicle components must be distributed so that the center of gravity is close to the intersection of the gimbal axes for all flight conditions. Weight considerations dictated the selection of four legs in preference to three or five. The pilot was centered between two legs in order to keep all the legs similar in design.

The structural frame of the LLRV consists of six basic sections: landing-leg structure, center-body structure, cockpit-support structure, aft-equipment structure, engine-mount assembly, and gimbal-ring assembly. The sections are constructed primarily of 2024- and 6061-type aluminum alloy. Shock-absorbing struts on the four legs are each capable of withstanding a load of 2260 pounds (10,000 newtons) and each have a stroke of 14 inches (35 centimeters). These struts will make it possible to touch down with sinking speeds up to 10 ft/sec (3.05 m/sec), if both the vehicle and the landing surface are level and there is no side velocity. Table VI shows the strut loads and vehicle load factors for various combinations of critical conditions. It is assumed that the landing surface is concrete and that the coefficient of friction between the landing pad and the concrete is infinite. During a lunar-simulation landing, jet-engine thrust will be five-sixths of the vehicle weight. To be conservative, the structural-design analysis assumed a jet thrust equal to only two-thirds of the vehicle weight.

Two design weight configurations were used in the structural design and analysis: 3400 pounds (1542 kilograms) and 2600 pounds (1179 kilograms). At the time the analysis was made, these values were considered to be the maximum and the minimum landing weight, respectively. The 3400-pound weight (1542 kilograms) represents a vehicle that has consumed some fuel in taking off and maneuvering into position to land. It is assumed that normal landings will be made after appreciable time in flight. Landings after flights of short duration, with a vehicle weight greater than 3400 pounds (1542 kilograms), must be made at less than design velocities. The 2600-pound weight (1179 kilograms) represents a vehicle with some equipment off-loaded and most of the fuel consumed. The maximum weight of 3400 pounds (1542 kilograms) is used to establish the design ground reactions based on the conditions listed in table VI. These design ground reactions are critical for the landing gear, legs, and portions of the center-body structure. To take advantage of this strength at the lighter vehicle weight (which will exist for most landings), the same design ground reactions are used for the 2600-pound vehicle (1179 kilograms) to determine the design load factors for the portions of the structure that are critical for inertia landings. These higher load factors for the 2600-pound weight (1179 kilograms) are also listed in table VI.

Additional load factors are applied to special portions of the vehicle. To insure that the fuel tanks will remain with the airframe in a crash severe enough to collapse the leg structure, the tank mounts are designed for a vertical limit load factor of 8.5. Similarly, the seat-support structure is designed for a limit load factor of 13.33 applied in any direction between vertical and 20° from vertical. Design ultimate loads for the airframe are 1.5 times greater than the limit loads.

The rubber mounts, which react to the lateral loads of the landing gear, are designed to permit the landing foot to deflect 6.75 inches (17.15 centimeters) to the side when one foot arrests the 3400-pound vehicle (1542 kilograms) moving at the design horizontal velocity of 3 ft/sec (0.91 m/sec). This landing condition results in a 1700-pound lateral load (7560 newtons) at the bottom of the foot. The corresponding vertical load is calculated by making the line of action of the resultant of the lateral and vertical loads pass through the center of gravity. A smaller vertical load would cause the vehicle to overturn. A greater vertical load would be less critical for the leg structure.

There is no structural limit on horizontal velocity at touchdown if the coefficient of friction between the vehicle and the ground is 0.5 or less. Rubber-tired casters, rather than the usual steel pads, were used on initial flights to insure this condition. Even when steel pads are used on future flights, the coefficient of friction will probably be about 0.5; hence, horizontal velocity should present no structural hazard.

The lack of aerodynamic surfaces makes air loads too small to be of structural importance for speeds within the design limits of 60 ft/sec (18.3 m/sec) horizontally and 100 ft/sec (30.5 m/sec) vertically. The deployment of the drogue parachute could apply a load of 2500 pounds (11,100 newtons) to any one bridle.

RESEARCH INSTRUMENTATION

Weight limitations in LLRV design made it necessary to use the lightest possible components in the research instrumentation. It was determined that significant weight could be saved by utilizing, for research purposes, sensors that were to be installed by the manufacturer for system use. Therefore, such items as the gyros and accelerometers discussed previously are used to acquire research information.

During the design phase, the validity of pressure-sensed altitude or velocity data was questioned because of the unpredictable flow characteristics about the vehicle. Thus, it was decided to use a radar altimeter for height and vertical-velocity information and a helicopter Doppler radar for forward and lateral velocity.

Further, in the interest of saving weight, a pulse-code modulation (PCM) telemetry system was selected to relay research data to a ground station in lieu of conventional, on-board NASA film recording. Such a system also has the advantage of providing data in a convenient tape form for automatic data processing. The PCM system transmits 80 channels of data, and data from each channel is recorded 200 times each second. All the research-instrumentation sensors and their ranges are listed in table II. Quantities are accurate to at least 1 percent of full-scale readings, with the exception of altitude and yaw angle. The altimetry system has an accuracy of 5 percent above an altitude of 40 feet (12.2 meters) and an accuracy of 2 feet (0.6 meter) below 40 feet (12.2 meters). The directional gyro has a drift rate of 0.25 deg/min (0.24 rad/sec); thus, accuracy is a function of time.

VEHICLE AERODYNAMICS

The LLRV was designed to have the smallest possible aerodynamic forces and moments to reduce to a minimum the task of the electronic aerodynamic-compensation system. Since the attitude-control system will compensate for all uncommanded moments on the vehicle, the ultimate velocities of the vehicle are established at the point where full control authority is required to maintain vehicle attitude. Therefore, a knowledge of the aerodynamic characteristics is necessary before control power can be set for a desired trajectory. As the design progressed, theoretical estimates of major aerodynamic parameters were made. Contributing factors considered were the structure and the jet engine.

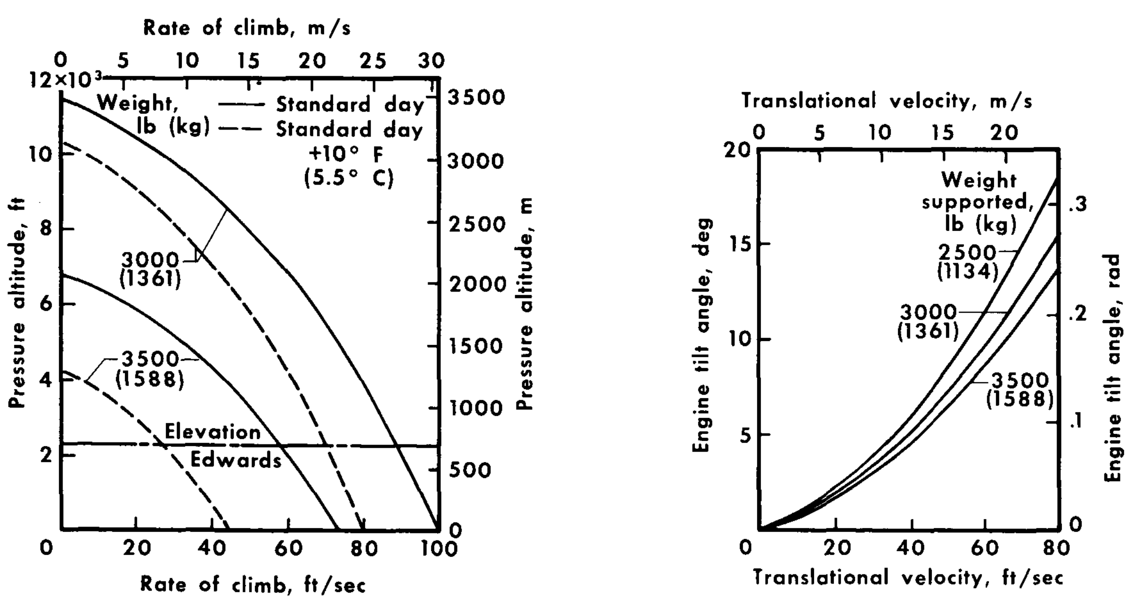
The engine-off forces and moments of the basic structure were estimated by using a cross-flow drag theory. Contributions of the canopy, equipment platform, tanks, and other regular geometric shapes were obtained from Hoerner (ref. 12). Aerodynamic forces on the jet engine were analyzed on the basis of data obtained from reference 13. Corrections were made to account for differences between the configuration in reference 13 and the LLRV. External drag and moments, which, in general, were a small part of the total, were calculated by using a drag coefficient of 1.0 and a center of pressure at the centroid of area. The procedures used to estimate force and moment were verified during wind-tunnel tests on an early configuration.

A 0.3-scale model of an early configuration was tested in the 17-foot section of the Langley 7- by 10-foot wind tunnel (ref. 5). An air-driven motor in the model simulated LLRV jet-engine operation. Test variables included tunnel dynamic pressure (0.27 to 12 lb/ft^2) (12.9 to 574 N/m^2), engine fan speed (static and 22,000 rpm), pitch-gimbal angle (0°, 20°, 40°) (0, 0.35, 0.70 rad), angle of attack (-60° to 180°) (-1.05 to 3.14 rad), and angle of sideslip (0° to 45°) (0 to 0.79 rad). Configuration variables included engine only, complete vehicle, cockpit modifications, payload packages removed, and several types of ground-adjustable control surfaces.

The aerodynamic force and moment characteristics for the vehicle are summarized in appendix B.

DESIGN PERFORMANCE

The LLRV jet engine lifts the vehicle to altitude and establishes velocity conditions at the start of a lunar simulation. As can be seen in figure 26, performance is highly dependent on barometric pressure and ambient air temperature. These factors are reflected in the operating envelopes shown in figure 33. From figure 33(a) it can be seen that even a heavily loaded vehicle can reach an altitude of several thousand feet under standard-day conditions. Figure 33(b) shows the tilt angles required of the vehicle in the engine-centered mode to achieve horizontal velocities. Less than 10° (0.6 rad) of



(a) Vertical flight.

(b) Horizontal flight.

Figure 33.— Estimated operating envelopes.

tilt will produce design velocities of 60 ft/sec (18.3 m/sec) at most weight conditions.

The exact nature of the terminal landing maneuver of the lunar excursion module (LEM) is one of the areas of research proposed for the LLRV. Several maneuvers have been examined (refs. 14 and 15); the nominal maneuver seems to be similar to that depicted in figure 34, which shows a typical simulation using the LLRV. The LLRV takes off with thrust from only the jet engine and climbs to altitude, tips to gain forward velocity, then reduces thrust to achieve vertical velocity, all with the jet-engine-gimbal locked. At the start of lunar simulation, the gimbal is unlocked and the jet engine supports five-sixths of the vehicle weight in addition to canceling all aerodynamic and extraneous forces on the vehicle. LLRV flight duration is limited by fuel capacity. A proposed maneuver such as that shown in figure 34 requires 37 pounds (17 kilograms) of jet fuel and 14 pounds (6 kilograms) of rocket fuel for the attitude rockets to get into position to simulate the lunar landing. The landing requires 55 pounds (25 kilograms) of jet fuel, 450 pounds (204 kilograms) of rocket fuel for the lift rockets, and 22 pounds (10 kilograms) of rocket fuel for the attitude rockets. The vehicle has adequate fuel for 10 minutes of jet flight and 2 minutes of normal lift-rocket operation.

Trajectory position	A	B	C	D	E	F	G
Time, sec	-56	-44	-31	-4	0	37	99
Horizontal velocity, ft/sec (m/s)	0	0	0	60 (18.3)	60 (18.3)	23 (7.0)	0
Vertical velocity, ft/sec (m/s)	0	62 (18.9)	0	0	-20 (-6.1)	-10 (-3.0)	-6.7 (-2.0)
Vehicle pitch angle, deg (rad)	0	0	-10 (-0.174)	-10 (-0.174)	10 (0.174)	9.7 (0.169)	0
Flight phase	Takeoff	Ascent	Horizontal acceleration	Vertical acceleration	Start simulation	Mid simulation	Landing

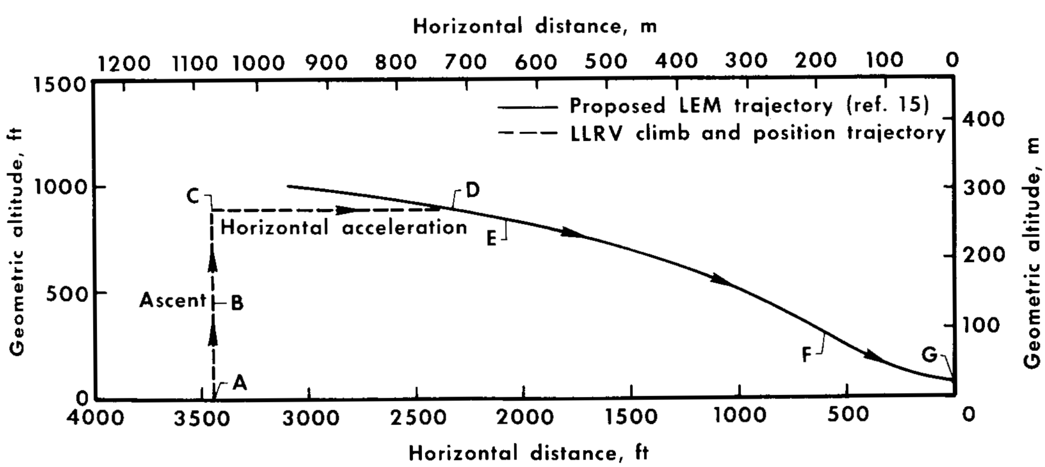


Figure 34.— Typical LLRV trajectory simulating a lunar landing.

EMERGENCY FEATURES AND PROCEDURES

In the design of the LLRV, considerable attention was given to the problem of vehicle and pilot safety. With no lifting surfaces, the power-off condition on the vehicle is critical. In an emergency, however, there are two means of saving the vehicle and three means of saving the pilot.

The vehicle has two propulsion sources; if one fails, the pilot can land by using the other. The jet engine, the primary propulsion unit, has proved to be reliable. At takeoff, the thrust-to-weight ratio is 1.05 or greater. At landing, the ratio could increase to about 1.5, well above the level specified in reference 16. Should the jet engine fail, the pilot would use the lift rockets for recovery. For maximum thrust, he would use the two normal lift rockets and six emergency lift rockets, which combine to produce a thrust-to-weight ratio of 1.39 at landing.

The prime restriction on the rockets for recovery is the fuel limitation. Predictions, verified by wind-tunnel tests, indicated the terminal velocity to be approximately 200 ft/sec (60 m/sec). Therefore, to reduce the energy required to decelerate, a 22-foot-diameter (6.7 meters), flared-skirt drogue parachute is used to lower the terminal velocity to about 85 ft/sec (26 m/sec), from which recovery can be made. To expedite inflation, the parachute is mortar-deployed. A parachute system to recover the entire vehicle was considered but was discarded when the system weight approached 1000 pounds (450 kilograms) (ref. 2).

Figure 35 shows the recovery envelope in terms of vertical velocity and altitude for the vehicle with the drogue parachute extended. The boundary at the lower right indicates the area below which the LLRV does not have sufficient thrust to execute a soft landing. The boundary at the upper left is the fuel-limiting boundary. To the left of this boundary, the vehicle will descend at such a low velocity that all fuel will be expended before touchdown. The pilot can, however, free-fall across this boundary into the safe-recovery area as long as any altitude margin is available. Then, when the combination of altitude and vertical velocity reaches some convenient relationship within the limiting boundaries, the pilot would actuate the lift rockets and modulate lift-rocket thrust to maintain the relationship between altitude and vertical velocity until touchdown.

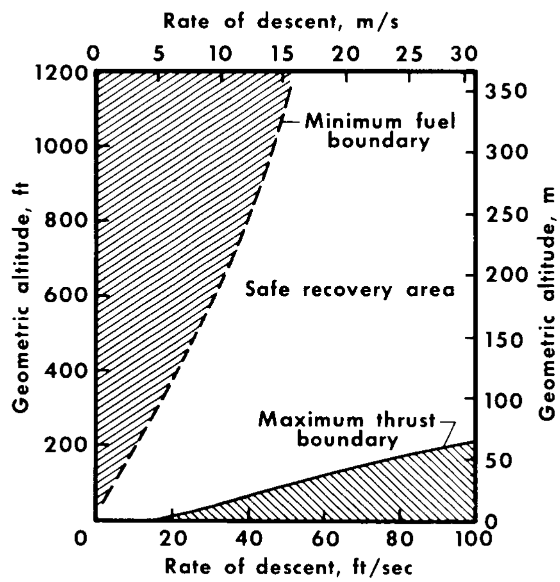


Figure 35.– LLRV recovery boundaries using emergency lift rockets. Initial weight, 3400 lb (1542 kg), drogue chute extended.

If both the jet- and the rocket-propulsion devices fail, the pilot is

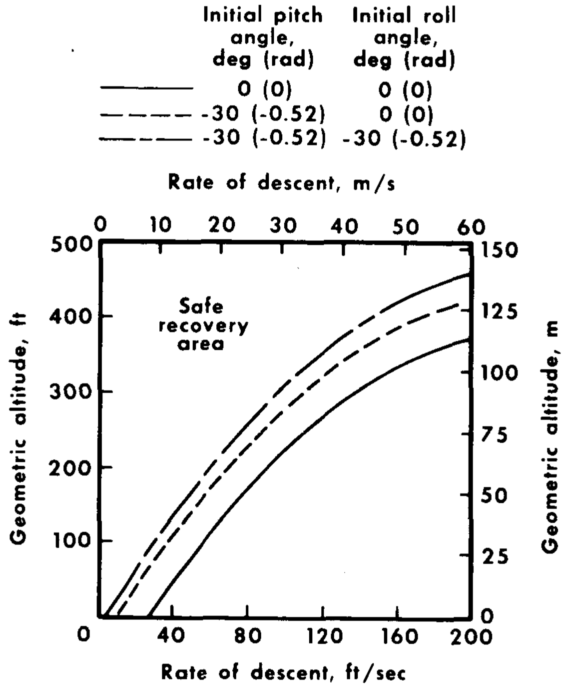


Figure 36.– Ejection-seat recovery boundary.

equipped with a ground-level ejection seat. Recovery curves for the ejection seat are presented in figure 36 for various combinations of vertical velocity and altitude. The drogue-parachute attachments on the four corners of the center frame and riser length are designed so that the pilot can eject from the vehicle with the parachute attached and deployed.

The drogue parachute and the ejection seat were tested to verify satisfactory operation. Tests of the parachute at the Naval Aerospace Recovery Facility at El Centro, Calif., demonstrated satisfactory mortar deployment, riser-load distribution to the vehicle frame, and overall system stability with the parachute inflated. In addition to static loadings and bench tests of all critical components, the seat manufacturer demonstrated the ground-level recovery capability of the seat in three dynamic ground firings. A 75-percentile anthropomorphic dummy simulated pilot weight and center of gravity. Tests were performed from the upright position, rolled 30°, and pitched forward 30°. As can be seen from figure 36, the latter case, as predicted, was marginal. The important factor, however, was that all components functioned properly in all tests.

Flight Research Center,
National Aeronautics and Space Administration,
Edwards, Calif., June 28, 1965.

APPENDIX A

NOTATION

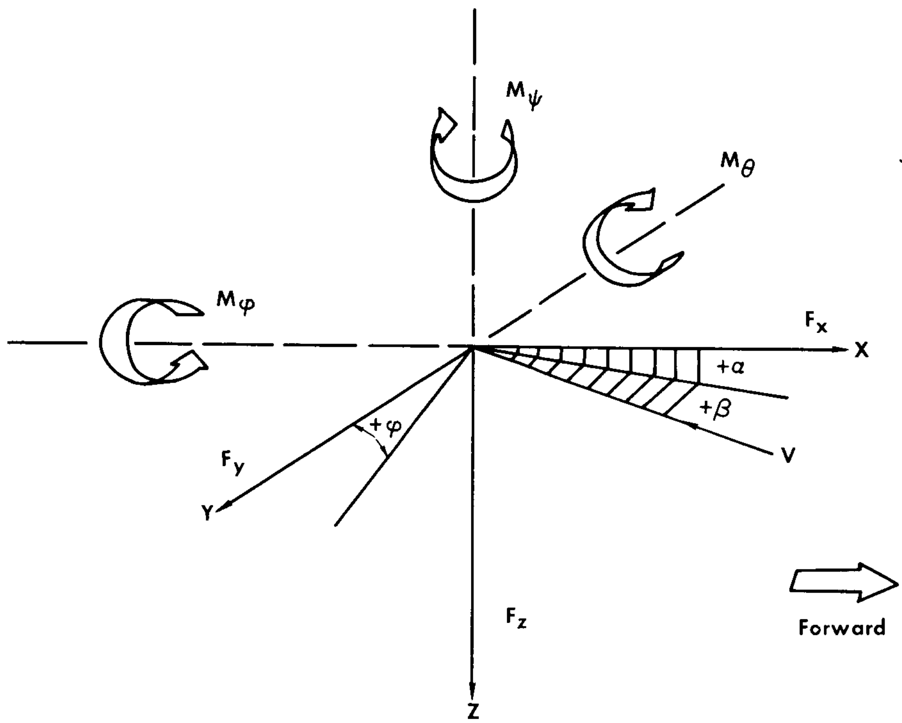
D	jet-engine drag force, pounds (newtons)
F_X	X-body-axis force, pounds (newtons)
F_Y	Y-body-axis force, pounds (newtons)
F_Z	Z-body-axis force, pounds (newtons)
g	acceleration due to gravity, 32.2 feet/second^2 (9.80 meters/second^2)
I_X	moment of inertia about X-axis, slug-foot^2 (kilogram-meters^2)
I_Y	moment of inertia about Y-axis, slug-foot^2 (kilogram-meters^2)
I_Z	moment of inertia about Z-axis, slug-foot^2 (kilogram-meters^2)
M	jet-engine tipover moment, foot-pounds (meter-newtons)
M_θ	pitching moment, foot-pounds (meter-newtons)
M_φ	rolling moment, foot-pounds (meter-newtons)
M_ψ	yawing moment, foot-pounds (meter-newtons)
T	jet-engine thrust, pounds (newtons)
V	total velocity, $\sqrt{V_X{}^2 + V_Y{}^2 + V_Z{}^2} = \sqrt{(\dot{x})^2 + (\dot{y})^2 + (\dot{z})^2}$, feet/second (meters/second)
V_X	velocity along X-body axis, feet/second (meters/second)
V_Y	velocity along Y-body axis, feet/second (meters/second)
V_Z	velocity along Z-body axis, feet/second (meters/second)
W_p	pilot weight, pounds (kilograms)

$\dot{W}$	air mass flow through jet engine, pounds/second (kilograms/second)
X	body axis, longitudinal axis parallel to ground when vehicle is in normal level position
X_{cp}	moment arm of jet-engine aerodynamic forces, feet (meters)
$\dot{x}$	earth-referenced forward velocity, feet/second (meters/second)
Y	body axis, lateral axis parallel to ground when vehicle is in normal level position
$\dot{y}$	earth-referenced side velocity, feet/second (meters/second)
Z	body axis, vertical axis perpendicular to X- and Y-axes
$\dot{z}$	earth-referenced vertical velocity, feet/second (meters/second)
α	angle of attack
β	angle of sideslip
θ	pitch angle
ρ	air density, slugs/foot^3 (kilograms/meter^3)
φ	roll angle, and electrical phase (fig. 23 only)

Subscripts:

g	gimbal relative to body
v	vehicle
ϵ	jet engine
τ	total

The reference frame used for the aerodynamic characteristics presented in this paper is the standard NASA notation shown in the following diagram. In this vehicle body-axis system, the positive sense of the orthogonal axes and force coefficients along the axes is forward for the X-direction, to the pilot's right side for the Y-direction, and down for the Z-direction. Vehicle nose up, right yaw, and right roll are positive.



A nominal altitude of 2500 feet (760 meters) is assumed with the resultant constant density of 0.0022 slug/ft^3 (11.34 kilograms/m^3) for all dimensioned forces and moments presented. No dynamic pressures in excess of 10 lb/ft^2 (479 N/m^2) are anticipated for any LLRV flights.

APPENDIX B

AERODYNAMIC FORCE AND MOMENT CHARACTERISTICS

The LLRV aerodynamic forces and moments were estimated early in the design phase and verified with wind-tunnel tests. The original predictions agreed well with the wind-tunnel tests; thus, the aerodynamic characteristics were reestimated without additional tunnel tests when a major configuration change became necessary.

The estimated engine-off force and moment characteristics are

$$F_{X_V} = -0.0832 V_X \sqrt{V_X^2 + V_Z^2}$$

$$F_{Y_V} = -0.0723 V_Y \sqrt{V_X^2 + V_Y^2}$$

$$F_{Z_V} = -0.0775 V_Z \sqrt{V_X^2 + V_Z^2}$$

$$M_{\theta_V} = -\left[0.0382 - 0.000141(200 - W_p)\right] V_X \sqrt{V_X^2 + V_Z^2}$$

$$M_{\psi_V} = -\left[0.1007 + 0.000124(200 - W_p)\right] V_Y \sqrt{V_X^2 + V_Y^2}$$

$$M_{\varphi_V} = -\left[0.051 - 0.000083(200 - W_p)\right] V_Z \sqrt{V_Y^2 + V_Z^2}$$

The engine drag, which is in the direction of the airflow, and the moment are

$$D = \frac{\dot{W} V \cos \alpha_\epsilon}{g} + 11.7\left(\frac{1}{2}\rho V^2\right)$$

$$M = \frac{\dot{W} V \cos \alpha_\epsilon}{g} X_{cp}$$

where

$$\dot{W} = 24.5 + 36.6\left(\frac{T}{1000}\right) - 3.303\left(\frac{T}{1000}\right)^2$$

$$X_{cp} = 2.2855(1 + 0.511 \sin \alpha_\epsilon)$$

so

$$D = \left[0.757 + 1.106\left(\frac{T}{1000}\right) - 0.1021\left(\frac{T}{1000}\right)^2\right] V \cos \alpha_\epsilon + 0.01287V^2$$

$$M = \left[1.726 + 2.551\left(\frac{T}{1000}\right) - 0.2326\left(\frac{T}{1000}\right)^2\right] (V \cos \alpha_\epsilon + 0.255V \sin 2\alpha_\epsilon)$$

Converting these data to the body axes yields

$$F_{X_\epsilon} = -D \frac{V_X}{V}$$

$$F_{Y_\epsilon} = -D \frac{V_Y}{V}$$

$$F_{Z_\epsilon} = -D \frac{V_Z}{V}$$

$$M_{\theta_\epsilon} = M \cos \varphi_g \left(\frac{V_X}{V} \cos \theta_g - \frac{V_Z}{V} \sin \theta_g\right)$$

$$M_{\psi_\epsilon} = M\left(\frac{V_X}{V} \sin \varphi_g + \frac{V_Y}{V} \sin \theta_g \cos \varphi_g\right)$$

$$M_{\varphi_\epsilon} = -M\left(\frac{V_Y}{V} \cos \theta_g \cos \varphi_g + \frac{V_Z}{V} \sin \varphi_g\right)$$

The $\cos \alpha_\epsilon$ and $\sin 2\alpha_\epsilon$ terms in the drag and moment equations can be expressed as involved transformations through the gimbal rings. However, by using an earth-fixed reference with $\dot{x}$ and $\dot{y}$ horizontal components, the influencing velocity vector becomes $\sqrt{(\dot{x})^2 + (\dot{y})^2}$ and the $\sin \alpha_\epsilon$ and $\cos \alpha_\epsilon$ terms reduce as follows

$$\sin \alpha_\epsilon = \frac{\dot{z}}{V}$$

$$\cos \alpha_\epsilon = \frac{\sqrt{(\dot{x})^2 + (\dot{y})^2}}{V}$$

$$\sin 2\alpha_\epsilon = \frac{2\dot{z} \sqrt{(\dot{x})^2 + (\dot{y})^2}}{V^2}$$

The total aerodynamic characteristics are the summation of the engine-off characteristics and the engine contributions

$$F_{X_\tau} = -V_X\left\{\left[0.757 + 1.106\left(\frac{T}{1000}\right) - 0.1021\left(\frac{T}{1000}\right)^2\right]\frac{\sqrt{(\dot{x})^2 + (\dot{y})^2}}{V} + 0.01287V + 0.0832\sqrt{V_X^{\,2} + V_Z^{\,2}}\right\}$$

$$F_{Y_\tau} = -V_Y\left\{\left[0.757 + 1.106\left(\frac{T}{1000}\right) - 0.1021\left(\frac{T}{1000}\right)^2\right]\frac{\sqrt{(\dot{x})^2 + (\dot{y})^2}}{V} + 0.01287V + 0.0723\sqrt{V_X^{\,2} + V_Y^{\,2}}\right\}$$

$$F_{Z_\tau} = -V_Z\left\{\left[0.757 + 1.106\left(\frac{T}{1000}\right) - 0.1021\left(\frac{T}{1000}\right)^2\right]\frac{\sqrt{(\dot{x})^2 + (\dot{y})^2}}{V} + 0.01287V + 0.0775\sqrt{V_X^{\,2} + V_Z^{\,2}}\right\}$$

$$M_{\theta_\tau} = \left[1.726 + 2.551\left(\frac{T}{1000}\right) - 0.2326\left(\frac{T}{1000}\right)^2\right]\sqrt{(\dot{x})^2 + (\dot{y})^2}\left(1 + 0.51\,\frac{\dot{z}}{V}\right)\left(\cos\varphi_g\right)\left(\frac{V_X}{V}\cos\theta_g\right.$$

$$\left. - \frac{V_Z}{V}\sin\theta_g\right) - \left[0.0382 - 0.000141\left(200 - W_p\right)\right]V_X\sqrt{V_X^{\,2} + V_Z^{\,2}}$$

$$M_{\psi_\tau} = \left[1.726 + 2.551\left(\frac{T}{1000}\right) - 0.2326\left(\frac{T}{1000}\right)^2\right]\sqrt{(\dot{x})^2 + (\dot{y})^2}\left(1 + 0.51\,\frac{\dot{z}}{V}\right)\left(\frac{V_X}{V}\sin\varphi_g\right.$$

$$\left. + \frac{V_Y}{V}\sin\theta_g\cos\varphi_g\right) - \left[0.1007 + 0.000124\left(200 - W_p\right)\right]V_Y\sqrt{V_X^{\,2} + V_Y^{\,2}}$$

$$M_{\varphi_\tau} = -\left[1.726 + 2.551\left(\frac{T}{1000}\right) - 0.2326\left(\frac{T}{1000}\right)^2\right]\sqrt{(\dot{x})^2 + (\dot{y})^2}\left(1 + 0.51\,\frac{\dot{z}}{V}\right)\left(\frac{V_Y}{V}\cos\theta_g\cos\varphi_g\right.$$

$$\left. + \frac{V_Z}{V}\sin\varphi_g\right) - \left[0.051 - 0.000083\left(200 - W_p\right)\right]V_Z\sqrt{V_Y^{\,2} + V_Z^{\,2}}$$

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TABLE I

PHYSICAL CHARACTERISTICS

Dimensions, feet (meters) –	
Overall length	22.50 (6.85)
Overall width	15.08 (4.60)
Overall tread	13.35 (4.07)
Height with footpads attached and struts extended (plus 0.62 ft (0.19 m) if casters are attached; minus 1.17 ft (0.36m) if struts are fully retracted):	
Overall	10.00 (3.05)
Center of gravity	6.50 (1.98)
Cockpit floor	5.83 (1.78)
Weights, pounds (kilograms) –	
Primary structure including engine gimbal ring	500 (227)
Landing gear (plus 38 lb (17.2 kg) when casters replace the pads)	140 (64)
Manual controls	45 (20)
Control avionics and wiring	160 (73)
Jet-engine system including hydraulics	800 (363)
Rocket system	370 (168)
Flight instruments, radar sensors, wiring, and console	115 (52)
Electrical system	115 (52)
Ejection-seat parachute and breathing oxygen	135 (61)
Drogue parachute and attachments	20 (9)
Research instrumentation and telemetry system	105 (48)
Communications	5 (2)
Normal empty weight	2510 (1139)
Useful load at takeoff:	
Pilot	185 (84)
JP4 fuel (430 lb (195 kg) less 6 min of idle at 8 lb/min (3.6 kg/min))	382 (173)
Hydrogen peroxide (672 lb (305 kg) less 60 lb (27.2 kg) for preflight checks)	612 (285)
Helium gas	5 (2)
Engine oil	8 (4)
Total useful load	1192 (548)
Maximum takeoff weight	3702 (1687)

TABLE II

RESEARCH INSTRUMENTATION

Quantity	Type of sensor	Range
Altitude	Radar altimeter	0 to 1000 ft (0 to 305 m)
Velocity –		
Forward	Doppler radar	±60 ft/sec (±18 m/sec)
Side	Doppler radar	±50 ft/sec (±15 m/sec)
Vertical	Radar altimeter	±50 ft/sec (±15 m/sec)
Acceleration –		
*Longitudinal	Linear accelerometers	±1g
*Lateral	Linear accelerometers	±1g
*Vertical	Linear accelerometers	0 to 2g
Vehicle attitude –		
*Pitch	Vertical gyro	±45° (±0.78 rad)
*Roll	Vertical gyro	±45° (±0.78 rad)
*Yaw	Directional gyro	360° (6.28 rad)
Vehicle angular rate –		
*Pitch	Rate gyro	±40 deg/sec (±0.70 rad/sec)
*Roll	Rate gyro	±40 deg/sec (±0.70 rad/sec)
*Yaw	Rate gyro	±40 deg/sec (±0.70 rad/sec)
Vehicle angular acceleration –		
Pitch	Displaced linear accelerometers	±60 deg/sec^2 (±1.05 rad/sec^2)
Roll	Displaced linear accelerometers	±60 deg/sec^2 (±1.05 rad/sec^2)
Yaw	Displaced linear accelerometers	±60 deg/sec^2 (±1.05 rad/sec^2)
Engine attitude –		
*Pitch	Rotary potentiometers	±45° (±0.78 rad)
*Roll	Rotary potentiometers	±45° (±0.78 rad)
Stick position –		
Pitch	Control-position transducer	±10° (±0.17 rad)
Roll	Control-position transducer	±10° (±0.17 rad)
Pedal yaw position	Control-position transducer	±45° (±0.78 rad)
Jet-throttle position	Control-position transducer	0 to 50° (0 to 0.87 rad)
Lift-rocket-throttle position	Control-position transducer	0 to 34° (0 to 0.59 rad)
Demodulator output –		
*Pitch control system	Signal voltage	±2 volts
*Roll control system	Signal voltage	±2 volts
*Yaw control system	Signal voltage	±2 volts
Accelerometer error –		
*Longitudinal control system	Signal voltage	±5 volts
*Lateral control system	Signal voltage	±0.015 volt
*Vertical control system	Signal voltage	±0.015 volt
*Thrust-weight-computer output	Signal voltage	0 to 5 volts
Chamber pressure –		
*Attitude rocket (16)	Potentiometer transducer	0 to 400 psia (0 to 276 N/cm^2)
*Lift rocket (2)	Potentiometer transducer	0 to 750 psia (0 to 517 N/cm^2)
Peroxide-tank pressure (2)	Potentiometer transducer	0 to 500 psia (0 to 345 N/cm^2)
Jet-compressor discharge pressure	Potentiometer transducer	0 to 100 psia (0 to 69 N/cm^2)
Gear –		
Oleo position (4)	Control-position transducer	0 to 14 in. (0 to 0.36 m)
Acceleration	Linear accelerometer	-3g to 10g
Strain gages	Two- and four-arm bridge gages	------------------------------
Warning light –		
Jet-fuel caution	On-off digital voltage monitor	------------------------------
Jet-fuel low	On-off digital voltage monitor	------------------------------
Jet-engine oil pressure	On-off digital voltage monitor	------------------------------
Hydraulic pressure	On-off digital voltage monitor	------------------------------
Peroxide low (2)	On-off digital voltage monitor	------------------------------
Helium pressure	On-off digital voltage monitor	------------------------------
Doppler radar status	On-off digital voltage monitor	------------------------------
On-board-camera status	On-off digital voltage monitor	------------------------------
Attitude-rocket-system selection	On-off digital voltage monitor	------------------------------
Jet-stabilization-system operation	On-off digital voltage monitor	------------------------------
Local-vertical-mode selection	On-off digital voltage monitor	------------------------------
Emergency gimbal lock position	On-off digital voltage monitor	------------------------------
Automatic-throttle selection	On-off digital voltage monitor	------------------------------
Jet-engine maximum tilt	On-off digital voltage monitor	------------------------------
Initial weighing event	On-off digital voltage monitor	------------------------------
Roll-authority warning light	On-off digital voltage monitor	------------------------------
Roll trim-switch position	On-off digital voltage monitor	------------------------------
Backup-electronic-system selection	On-off digital voltage monitor	------------------------------
Malfunction-detector operation –		
Pitch	On-off digital voltage monitor	------------------------------
Roll	On-off digital voltage monitor	------------------------------
Yaw	On-off digital voltage monitor	------------------------------

*Used in control system as well as in research-instrumentation system.

TABLE III

PHYSICAL CHARACTERISTICS OF THE CF700-2V TURBOJET ENGINE

Rated thrust, sea-level static, ICAO standard day, lb (N)	4,200 (18,683)
Rated specific fuel consumption, lb/hr-lb (kg/hr-N)	0.69 (0.0704)
Gas-generator section –	
Number of compressor stages	8
Compression ratio	6.8
Number of turbine stages	2
Maximum speed, rpm (corrected)	16,500
Idle speed, rpm (corrected)	7,920
Direction of rotation (viewed from beneath, looking up)	Clockwise
Polar moment of inertia of rotating parts, slug-ft^2 (kg-m^2)	0.495 (0.0624)
Maximum gas-generator-turbine-outlet temperature, deg F	1,310
Fan section –	
Combined turbine and fan stages	1
Fan compression ratio	1.6
Maximum speed, rpm (corrected)	9,200
Idle speed, rpm	7,920
Direction of rotation (viewed from beneath, looking up)	Counterclockwise
Polar moment of inertia of fan rotor, slug-ft^2 (kg-m^2)	1.197 (0.1508)
Net gyroscopic torque at 1 rad/sec angular velocity and maximum thrust, ft-lb (m-N)	305 (414)
Length, front frame to tip of exhaust cone, in. (m)	63.5 (1.613)
Diameter (maximum), in. (cm) –	
Gas-generator section	17.7 (44.96)
Fan section	34 (86.4)
Weight (excluding non-engine accessories), lb (kg) –	
Dry	629.25 (285.42)
Wet	639.25 (289.96)
Exhaust exit area, in.^{2} (cm^2)	16.0 (103.2)

TABLE IV

ROCKET-SYSTEM CHARACTERISTICS

	Attitude rockets	Lift rockets
Rocket motors (manufacturer's specifications) –		
Thrust range, lb (N)	18 to 90 (80.1 to 400.3)	100 to 500 (444.8 to 2224)
Specific impulse (maximum thrust), sec (N-sec/kg)	129 (1265)	129 (1265)
Specific impulse (minimum thrust), sec (N-sec/kg)	102 (1000)	102 (1000)
Chamber pressure (maximum thrust), psia (N/cm^2)	319 (220)	325 (224)
Injector pressure (maximum thrust), psia (N/cm^2)	387 (267)	395 (272)
Propellant	90-percent H_2O_2	90-percent H_2O_2
Start response (hot), milliseconds	40	55
Service life, hr	2	2
Number required per vehicle:		
Normal use	8	2
Emergency use	8	6
Type of catalyst	Silver and platinum screens	Silver and platinum screens
Chamber weight, lb (kg)	1.91 (0.866)	7.92 (3.59)
Control valve, size and type	1/2 to 3/8 in., solenoid	1 in., manual ball
Nozzle-area expansion ratio	3.00	3.02

Propellant system –	
Propellant tanks:	
Number per vehicle	2
Internal volume of each, ft^3 (m^3)	3.86 (0.1093)
Material	AM350 stainless steel
Working pressure, psig (N/cm^2)	475 (328)
Helium tanks:	
Number per vehicle	2
Internal volume of each, ft^3 (m^3)	1.012 (0.0287)
Material	6AL-4V titanium
Working pressure, psig (N/cm^2)	3750 (2586)
Connecting tubing, material	347 stainless steel

TABLE V

COMPUTED ROCKET-PROPULSION-SYSTEM PRESSURES

Operating conditions	Mode	Normal lift rockets		Emergency lift rockets		Attitude rockets	
		No.	Thrust	No.	Thrust	No.	Thrust
A	Normal	2	500 lb (2224 N)	0	--------	4	83 lb (369 N)
B	Normal	2	250 lb (1112 N)	0	--------	4	83 lb (369 N)
C	Emergency	2	448 lb (1993 N)	6	476 lb (2117 N)	4	81 lb (360 N)

Location	Operating conditions		
	A	B	C
	Pressure, psia (N/cm^2)		
Upstream helium pressure regulator (assumed)	600 (413.7)	600 (413.7)	600 (413.7)
Downstream helium pressure regulator	490 (337.8)	490 (337.8)	490 (337.8)
Propellant tank	488 (336.5)	489 (337.2)	473 (326.1)
Downstream lift-system shutoff valve	484.6 (334.1)	488.15 (336.6)	423.8 (292.2)
Upstream normal lift-throttle valve	443.6 (305.9)	477.9 (329.5)	389.5 (268.6)
Downstream normal lift-throttle valve	411.9 (284.0)	216.1 (149.0)	372.8 (257.0)
Normal lift-rocket inlet	395 (272.3)	211.8 (146.0)	358.5 (247.2)
Normal lift-rocket chamber	325 (224.1)	178 (122.7)	296 (204.1)
Emergency lift-rocket inlet	0 (0)	0 (0)	376 (259.2)
Emergency lift-rocket chamber	0 (0)	0 (0)	314 (216.5)
Downstream attitude-system shutoff valve	460 (317.2)	460 (317.2)	446.2 (307.6)
Upstream attitude-rocket orifice	424 (292.3)	424 (292.3)	412 (284.1)
Attitude-rocket inlet	364 (251.0)	364 (251.0)	355 (244.8)
Attitude-rocket chamber	300 (206.8)	300 (206.8)	293 (202.0)

TABLE VI

STRUCTURAL LANDING CONDITIONS

	Vertical descent					Side drift			
	Four leg			Two leg		One leg		Two leg	
Weight, lb (kg)	3400 (1542)	2600 (1179)	2600 (1179)	3400 (1542)	2600 (1179)	3400 (1542)	2600 (1179	3400 (1542)	2600 (1179)
Lift, lb (N)	2267 (10,084)	1733 (7709)	4000 (17,793)	2267 (10,084)	1733 (7709)	2267 (10,084)	1733 (7709)	2267 (10,084)	1733 (7709)
Velocity, ft/sec (m/sec)									
Vertical	10 (3.05)	10 (3.05)	10 (3.05)	6 (1.83)	6 (1.83)	6 (1.83)	6 (1.83)	6 (1.83)	6 (1.83)
Horizontal	------	------	------	------	------	3.4 (1.04)	3.4 (1.04)	3 (0.92)	3.4 (1.04)
Strut load, lb (N)									
Vertical	2260 (10,053)	2260 (10,053)	1690 (7517)	2260 (10,053)	2260 (10,053)	1260 (5605)	1260 (5605)	1260 (5605)	1260 (5605)
Longitudinal	--------	--------	------	--------	--------	1200 (5338)	1200 (5338)	1200 (5338)	1200 (5338)
Lateral	--------	--------	------	--------	--------	1200 (5338)	1200 (5338)	------	------
Load factor									
Vertical	3.33	4.14	4.14	2.00	2.40	1.04	1.15	1.41	1.64
Horizontal	----	----	----	----	----	.35	.46	.71	.92
Horizontal	----	----	----	----	----	.35	.46	----	----

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