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ATTENUATION OF LANDING IMPACT FOR MANNED SPACECRAFT

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NASA, Manned Spacecraft Center, [redacted]

INTRODUCTION

The landing techniques available for ballistic-reentry manned spacecraft require the use of deployable descent devices such as parachutes, paragliders, rotor systems, and balloons. Of these descent devices, only rotor systems and balloons have the capability to provide essentially zero impact velocity. Unfortunately, neither of these systems has yet reached a state of development sufficiently advanced to warrant their serious consideration for application to manned spacecraft.

The most highly developed descent device is, of course, the parachute system. In order to keep the system weight and volume within reason, parachutes are generally sized to give a vertical impact velocity of 25 to 30 feet per second. Parachutes are also subject to wind drift, which can result in horizontal impact velocities up to 50 feet per second. These impact velocities can produce acceleration levels and onset rates well beyond human tolerance levels unless some method is provided for attenuation of the impact energy.

The paraglider is another promising candidate deserving consideration as a spacecraft descent system. The paraglider will be capable of aircraft-type landings wherein the major part of the vertical velocity is converted to horizontal velocity through a flare maneuver. The flying characteristics of practical paraglider systems are such that vertical impact velocities of 5 to 10 feet per second can be expected, with horizontal velocities of about 100 feet per second. These touchdown velocities require not only initial impact shock attenuation but also a stable landing-gear configuration which permits a long run-out to dissipate the horizontal velocity.

As with all spacecraft systems, weight is a prime consideration in the design of impact attenuators. However, the volume requirement is often of more serious concern since space must be provided to accomodate the full stroke of the attenuation devices. It is difficult to allocate the premium volume required for stroking a device which is used only at the instant of mission termination. This fact

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does not compromise the requirements for impact attenuation, but it does dictate that attenuation systems be designed to operate efficiently and with minimum performance margins and establishes the need for extensive development and qualification testing to demonstrate that the selected systems will meet the performance requirements.

This paper presents an outline of the techniques used in the selection, design, development, and qualification of spacecraft impact attenuation systems. This outline of techniques is followed by a discussion of the attenuation systems used in NASA's current manned space programs, including the basis for selection, development procedures, and some of the development problems encountered.

DEVELOPMENT CONSIDERATIONS

The first step in the evolution of an impact attenuation system is to establish the landing surface and the terminal flight conditions for both normal and emergency mission situations. The landing surface, vertical and horizontal impact velocities, and vehicle landing attitudes must be defined. The desirability of refurbishment and reuse of the spacecraft must also be considered. These criteria are presented in figure 1. Analysis of these factors through parametric studies and tradeoffs leads to the selection of the most promising system concept.

Landing conditions as dictated by the normal levels of human endurance are presented in figure 2. These levels are valid if adequate support and restraint are provided for crew members. Such restraint would include supports and harnesses for the head, torso, and limbs. In order to grasp how these levels are related to energy absorption, consider, as an example, the onset rate of 500g per second at a 15g peak load. This onset rate would require an attenuator stroke of 2.83 inches to dissipate a 10-foot-per-second velocity, 8.4 inches to dissipate 20 feet per second, and 16.4 inches to dissipate 30 feet per second. The shock absorber is assumed 100-percent efficient. To compensate for the usual loss in absorber efficiency, another 25 percent must be added to obtain the actual strokes required. To establish the optimum crew position in the spacecraft, the crew acceleration limits, the magnitude and direction of impact loads and the stroking distance available must be considered.

Scaled models are used extensively during the initial investigation of impact systems. These models are usually geometrically and dynamically scaled. Analytical computer studies are initiated in conjunction with the model tests to verify test results. If good correlation is found between the test data and the computer study, the computer program can be used to extend data to a number of conditions without testing the model over the entire range. Model drop tests should be made at extreme conditions to establish confidence in the analytical program.

Initial full-scale testing is started on vehicles which are usually constructed of boilerplate steel. These vehicles also simulate the actual spacecraft in geometry, weight, inertia, and center-of-gravity location. Computer work continues through this stage of development. If acceptable results are obtained in the model and boilerplate testing, a relatively small number of actual spacecraft qualification tests is required.

MERCURY IMPACT SYSTEM

The well-known Mercury project is of particular interest from the standpoint of impact system development and qualification. The system was selected based on certain nominal criteria and later redesigned to accomodate specific emergency conditions. Logical development testing of the redesigned system disclosed problem areas which required further modification. Many of the techniques now used in the development of spacecraft impact attenuation systems evolved during the Mercury program.

The Mercury system incorporated a single 64-foot-diameter ringsail parachute and was initially designed for a water landing with the vehicle impacting at a nominal pitch attitude of 0°. Tests indicated that the spacecraft and crew would experience an impact load of about 30g, which was within the range of accepted human endurance levels. For additional crew protection during emergency conditions, crushable honeycomb was added to the couch support system. The honeycomb was designed to crush at a 35g level.

Although this configuration was adequate for normal water landing it became apparent that, for some abort situations, the spacecraft could impact on land. Land impact with this system could produce deceleration forces beyond the human tolerance level, thus further impact attenuation was required.

Inadequate space inside the vehicle for the attenuation stroke required in the astronaut couch structure made it necessary to install an airbag impact attenuation system to reduce the landing loads on the entire spacecraft. This system is shown in figure 3. The airbag is formed by a cylindrical fiberglass skirt attached at the heat shield and the bottom of the spacecraft. Following deployment of the landing parachute a release mechanism detached the heat shield from the spacecraft. Gravity extended the heat shield to form the impact skirt beneath the vehicle. Impact loads are controlled by restricting the pressure level within the bag. This restriction is accomplished by exhausting the entrapped air through a series of properly sized orifices during impact.

The development and qualification of this comparatively simple system was more difficult than might be imagined. The airbag performed satisfactorily in reducing vertical landing loads, but it had little tolerance for horizontal velocities induced by wind drift. In early impact tests the airbag was ruptured by high horizontal loads. A series of thin steel straps was added to prevent shearing of the airbag. For water landings, problems were encountered relating to flotation characteristics of the spacecraft. To avoid puncture of the pressure vessel by the detached heat shield a thin layer of honeycomb was added to the exposed face of the pressure vessel. Retention of the heat shield after water impact was required to provide the proper flotation attitude. Prolonged wave action could cause fatigue failure of the fiberglass skirt and the steel straps, resulting in loss of the heat shield. This possibility was eliminated by the addition of twenty-four 1/8-inch stainless steel cables.

Figure 4 shows the maximum accelerations along the X-axis for water impact of a spacecraft without the impact skirt, with zero horizontal velocity and a vertical velocity of -30 feet per second. This curve presents the impact load in g units as a function of spacecraft attitude. Note from this figure that a peak load of approximately 29g occurred at an impact attitude of 0°; however, as the attitude increases, the magnitude of the loads decreases appreciably to a value of less than 10g at an attitude of 30° as a result of the effect of the corner of the heat shield contacting the water first. Presented in figure 5 is an acceleration-time history for the spacecraft impacting on land and water with the skirt extended and retracted. All conditions shown are with the vehicle impacting at a vertical velocity of -30 feet per second, a horizontal velocity of 0, and a pitch attitude of 0°.

The Mercury program was completed in $4\frac{3}{4}$ years, during which time over 300 model and boilerplate impact tests were conducted to develop the landing system. Twenty production spacecraft were tested during this period to qualify the complete Mercury system for extended earth orbital missions.

GEMINI IMPACT SYSTEM

The Gemini program was initiated as an intermediate step between Project Mercury and the Apollo lunar-landing program. The original design requirements for the Gemini landing system included provision for astronaut selection of the landing site and landing under such controlled conditions as to insure the reuseability of the vehicle. To comply with these requirements, it was imperative that the descent system provide some glide capability which would enable the spacecraft to reach a designated prepared surface, such as an airfield, or at least to avoid major obstructions during the final landing approach.

One of the landing systems being developed for Gemini is a new concept in spacecraft design. It includes a deployable flexible wing, known as the paraglider, and an aircraft-type tricycle skid landing gear. After reentry the delta-wing configuration is deployed in the same manner as a parachute. Upon full deployment the vehicle is suspended beneath the wing by steel cables whose lengths are controlled manually by the astronaut to maneuver the vehicle. The nose gear is extended automatically during deployment of the paraglider. For a water landing the rear gear is not deployed until after impact. In this event its only function would be to improve flotation stability. For the normal ground landing the main gear is deployed manually at approximately 5,000 feet above the surface.

The landing sequence is illustrated in figure 6. Just prior to touchdown the astronaut executes a flare maneuver which converts the major portion of the vertical velocity into horizontal velocity. This horizontal velocity will be dissipated in friction forces between the landing surfaces and the landing gear skids. The residual vertical velocity is dissipated through hydraulic shock attenuators located inside the vehicle. These are the only shock absorbers included in the present design. In the event of pad abort the crew will eject from the spacecraft and descend separately on personnel parachutes.

Because of the extensive effort required for development and qualification of the paraglider, the Gemini spacecraft will be recovered by an 84-foot-ringsail parachute for the initial flights. The landing gear will not be used on these missions, which will normally terminate in a water landing. The spacecraft is suspended from the parachute at an attitude of 55°, nose up. As previously mentioned in the discussion of the Mercury program, this attitude gives low accelerations for water impact.

Model and full-scale jig tests of the landing gear have been accomplished. Model tests simulating the parachute landing impact have been accomplished and drop tests of a prototype spacecraft are in progress.

APOLLO COMMAND MODULE IMPACT SYSTEM

The Apollo command module will provide life-support systems for its three astronaut crew members for the long duration lunar missions and will serve as the reentry vehicle for the return flight. The large size of the command module requires a cluster of three 88-foot-diameter ringsail parachutes to provide a rate of descent of about 23 feet per second. At present, both land and water landings are being considered.

Discrete portions of the command module structure are designed for controlled failure during landing, providing limited attenuation of the impact energy. The command module is suspended on the parachute system at an attitude of 30°, which insures that the specially designed structure will always be the point of initial contact. Crushing of this edge will provide 2.5 inches of vertical stroke and 4.5 inches of horizontal stroke. Impact at the 30° attitude also provides some dissipation of energy through rolling of the command module about the point of initial contact.

The Apollo crew is further protected by a system of honeycomb shock struts which support the astronaut couches. This system is shown in figure 7. The strokes available for the crew couch struts are 14 inches in the "eyeballs-in" direction, 14 inches "eyeballs-down," 5 inches "eyeballs-up," and 4.5 inches in the "eyeballs-left" and "eyeballs-right" direction.

Requirements for onset rate and peak deceleration force are met by utilizing the stroke available from command module structural deformation and the operation of the crew shock struts. This impact system is entirely passive, which is a desirable feature. However, extensive testing of representative full-scale structure will be required in order to define concretely the energy dissipation capability available from plastic deformation of the structure.

Model and full-scale boilerplate drop tests are being conducted to determine impact dynamics, including onset rates, maximum accelerations, and vehicle turnover characteristics. Computer studies are also being utilized in this effort. Later in the program full-scale vehicles which incorporate the actual structure of the command module in the planned crushing area will be drop tested.

An alternate command module impact attenuation system was investigated early in the Apollo development program. This system consists of a series of six air-oil shock struts and eight honeycomb shock struts which are brought into operating position by extension of the entire aft heat shield. Drop tests of a 1/4-scale model of this system were conducted at NASA Langley Research Center, and a similar system is being used on the boilerplate vehicles employed in the parachute development program of the Joint Parachute Test Facility, El Centro, Calif.

LUNAR EXCURSION MODULE IMPACT SYSTEM

The lunar excursion module (LEM) will be inserted into a lunar orbit with the command module. The vehicles will separate, and the LEM will shuttle two astronauts to a landing on the lunar surface. The landing system for the LEM poses unique problems. Propulsion systems must be used to decrease the rate of descent since the lack of an atmosphere precludes the use of aerodynamic deceleration devices. The final phase of descent is controlled by a variable thrust propulsion system which provides limited capability for hovering over the landing site or maneuvering to a more desirable touchdown point. The LEM landing sequence is illustrated in figure 8.

The propulsion system is designed to provide impact vertical velocities no greater than 10 feet per second and horizontal velocities no greater than 5 feet per second. The landing gear must accomodate these impact velocities, as well as provide a stable platform for launch of the LEM for the return trip. The present gear configuration is made up of four spider-type legs or struts. In addition, two small secondary struts are attached between the LEM and each main strut. Both the main and secondary struts contain crushable honeycomb for shock attenuation. The honeycomb in the main struts is composed of two sections which are designed to crush at different acceleration levels. In the event that one gear contacts the landing surface before the others, the weaker honeycomb will provide some degree of attenuation without introducing serious pitching moments. To improve stability, each main strut is terminated with a circular pad of aluminum honeycomb.

One of the more serious problems affecting the design of the LEM landing system is the fact that the exact nature of the lunar surface is not known. Hypotheses on the composition vary from a deep dust with grain size of a few microns to frothy volcanic rock. Coefficient of friction may be anything from zero to infinity. Surface slope is known a little more accurately. The slope is thought to be not greater than 5° at the probable landing location. These factors will certainly affect final landing system design, but they cannot be positively determined without preliminary unmanned flights. For this reason, the LEM landing gear must necessarily be designed to accomodate the widest practical range of impact surfaces. Educated guesses have been made about the surface of the moon, and from these an engineering model has been established. It postulates semicontinuous solid rock covered by a 30-centimeter layer of rock froth and a 10-centimeter layer of dust. Average bearing pressure of the dust surface is assumed to be 12 pounds per square inch and its mean density is 3.3 cm^3. Holes and protuberances should not be greater than 10 centimeters.

Evaluation of the landing gear must include studies of stability, ultimate strength, and energy absorption characteristics. Model drop tests will be conducted on a simulated lunar surface. Of particular concern is the fact that the lunar gravity is only 1/6 that of earth. Since gravitational effects are important in determining impact dynamics, tests on earth must consider the reduced lunar gravity. Efforts have been applied to the development of test techniques which mechanically simulate the lunar gravity. Analytical methods are also utilized in correcting for gravitational differences.

ADVANCED IMPACT STUDIES

Future missions in space will pose new problems in providing for safe landing on various surfaces. Development and testing of specific impact systems are becoming more costly and time consuming as the size of the spacecraft increases. New techniques must be developed to eliminate or reduce the requirement for drop testing of full-scale prototype vehicles.

Methods for accurately scaling prototype structure in the plastic, as well as the elastic, region of material deformation are being investigated. Proper scaling of structural characteristics is mandatory in tests of impact systems that depend on programed structural failure for reduction of loads.

A similar problem is involved in the determination of the dynamic effects resulting from deformation of the impact surface. A study of the dynamic response characteristics of various types of soils is in progress.

Incorporation of the results of these studies in model testing and application of refined analytical techniques should alleviate the requirement for full-scale testing.

- I VELOCITY
  - A. HORIZONTAL
  - B. VERTICAL
- II LANDING SURFACE
  - A. TYPE (LAND OR WATER)
  - B. SURFACE CONDITION (MAX SLOPES)
  - C. COEFFICIENT OF FRICTION
- III VEHICLE ATTITUDE
  - A. PITCH
  - B. YAW
  - C. ROLL
- IV WEATHER CONDITIONS
- V OBJECTIVE OF MISSION
  - A. MANNED OR UNMANNED
  - B. REUSE OF VEHICLE

Figure 1.- Basic landing criteria.

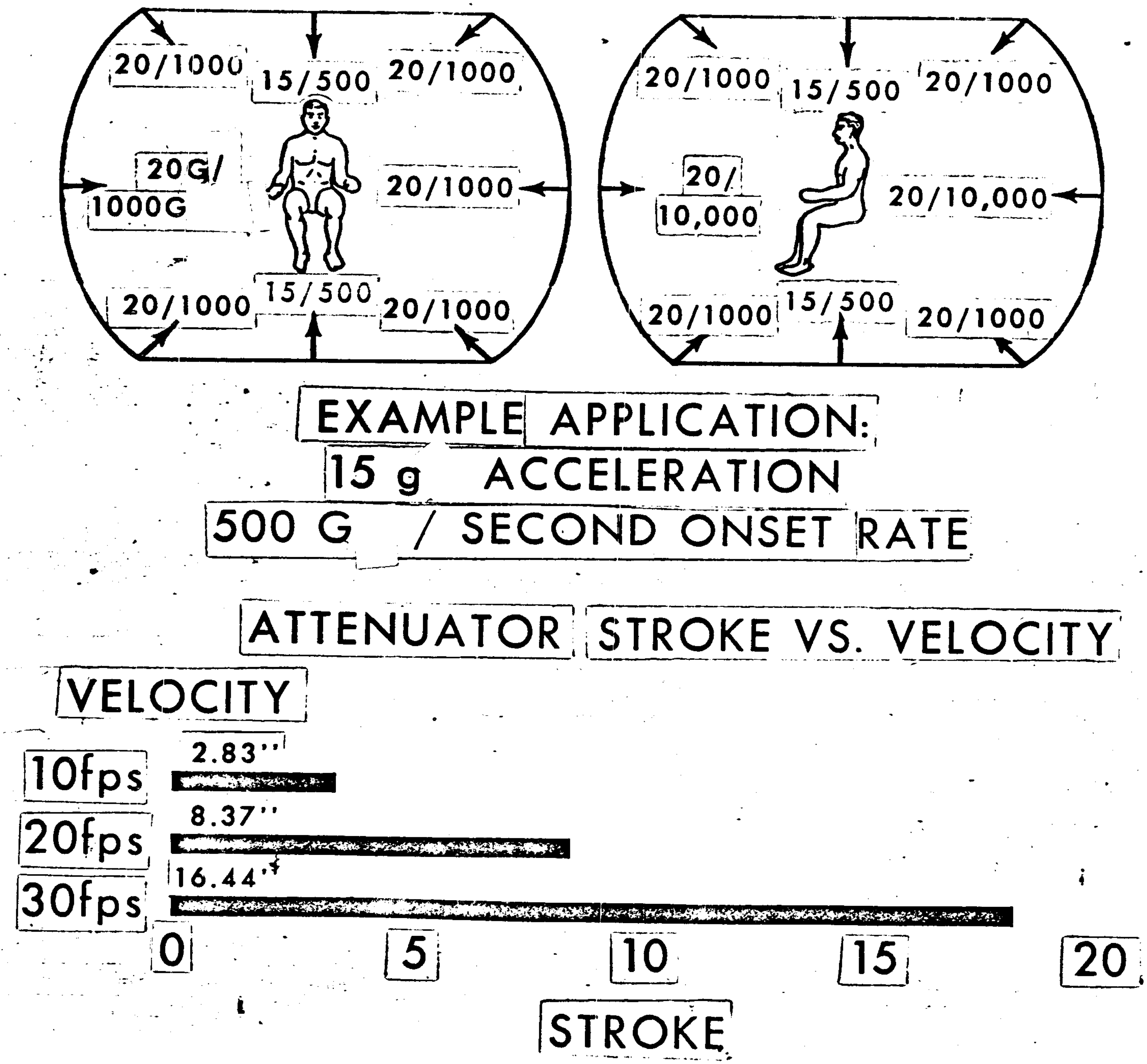

Figure 2 - Crew impact endurance levels.

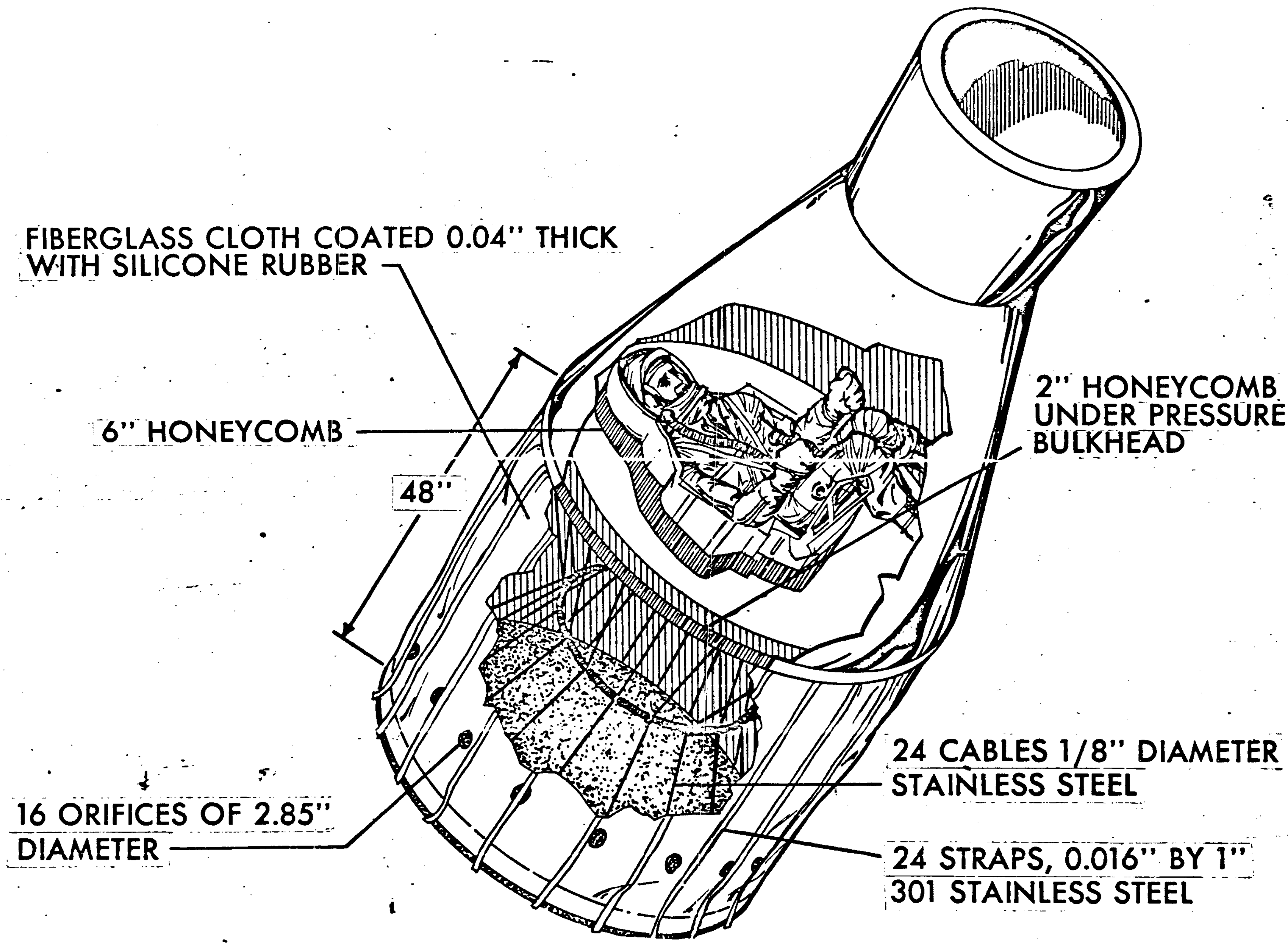

Figure 3.- Mercury impact attenuation system.

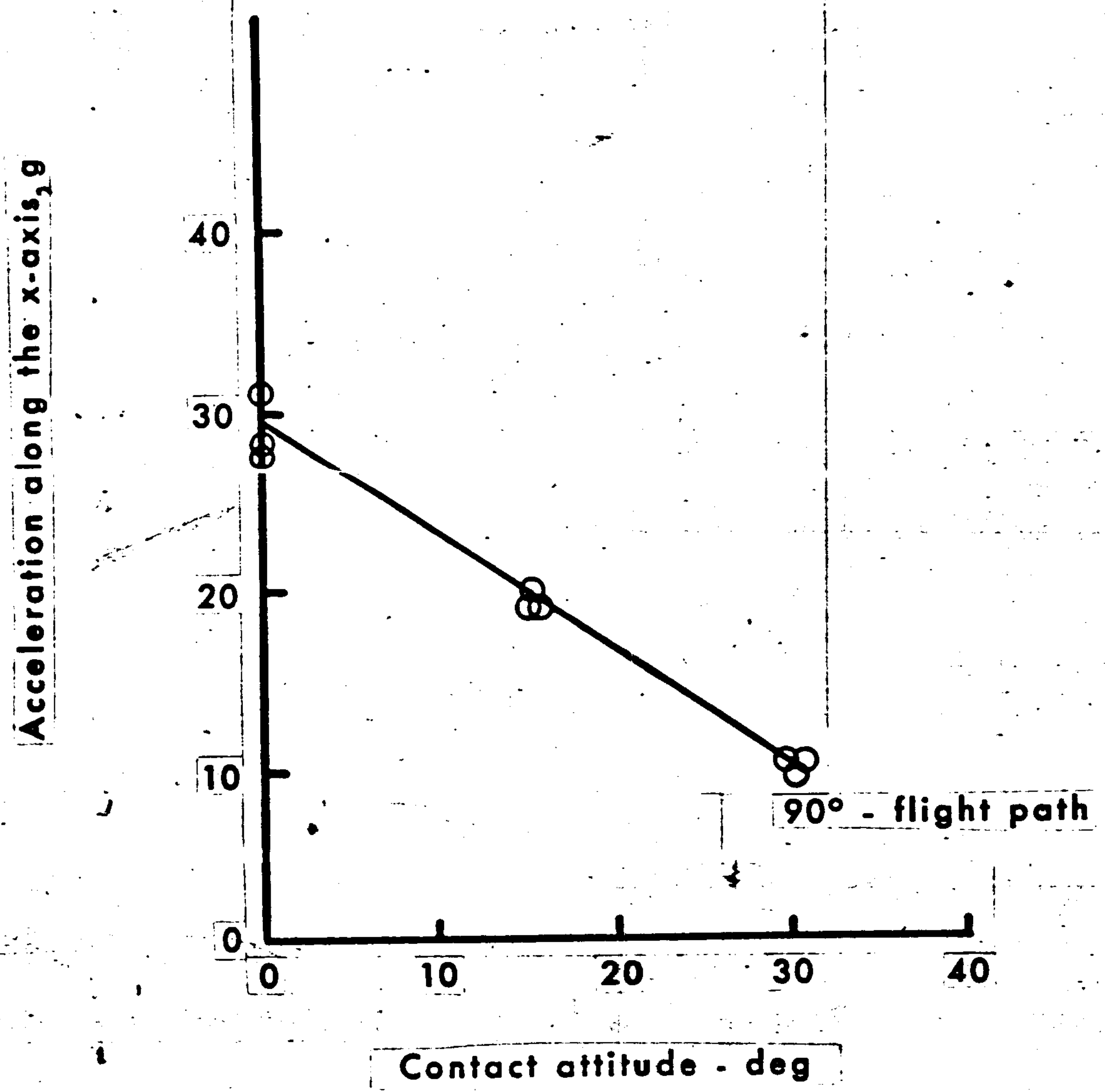

Figure 4.- Maximum accelerations along x-axis for water landings.

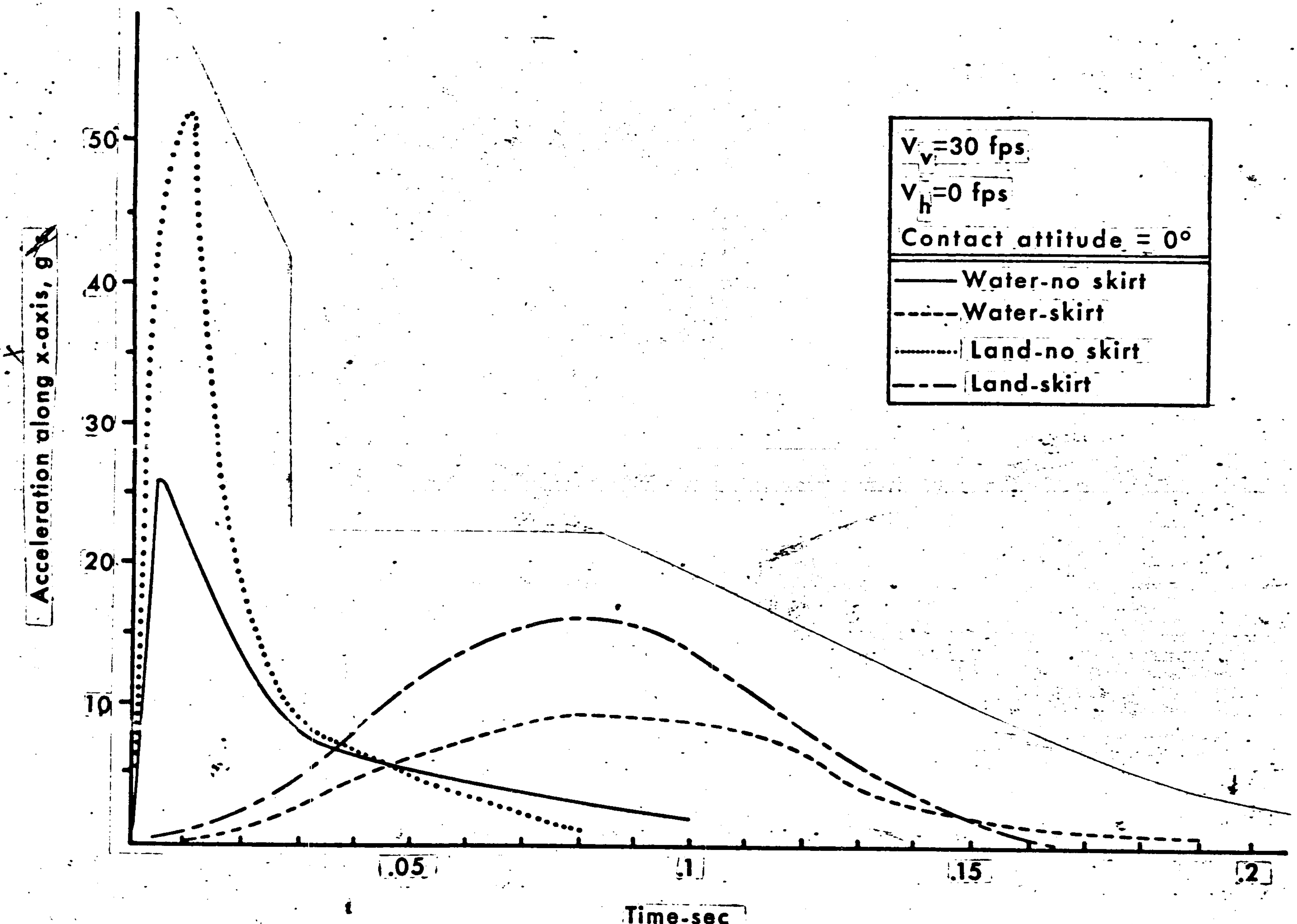

Figure 5.- Typical time histories of accelerations along x-axis for land and water landings.

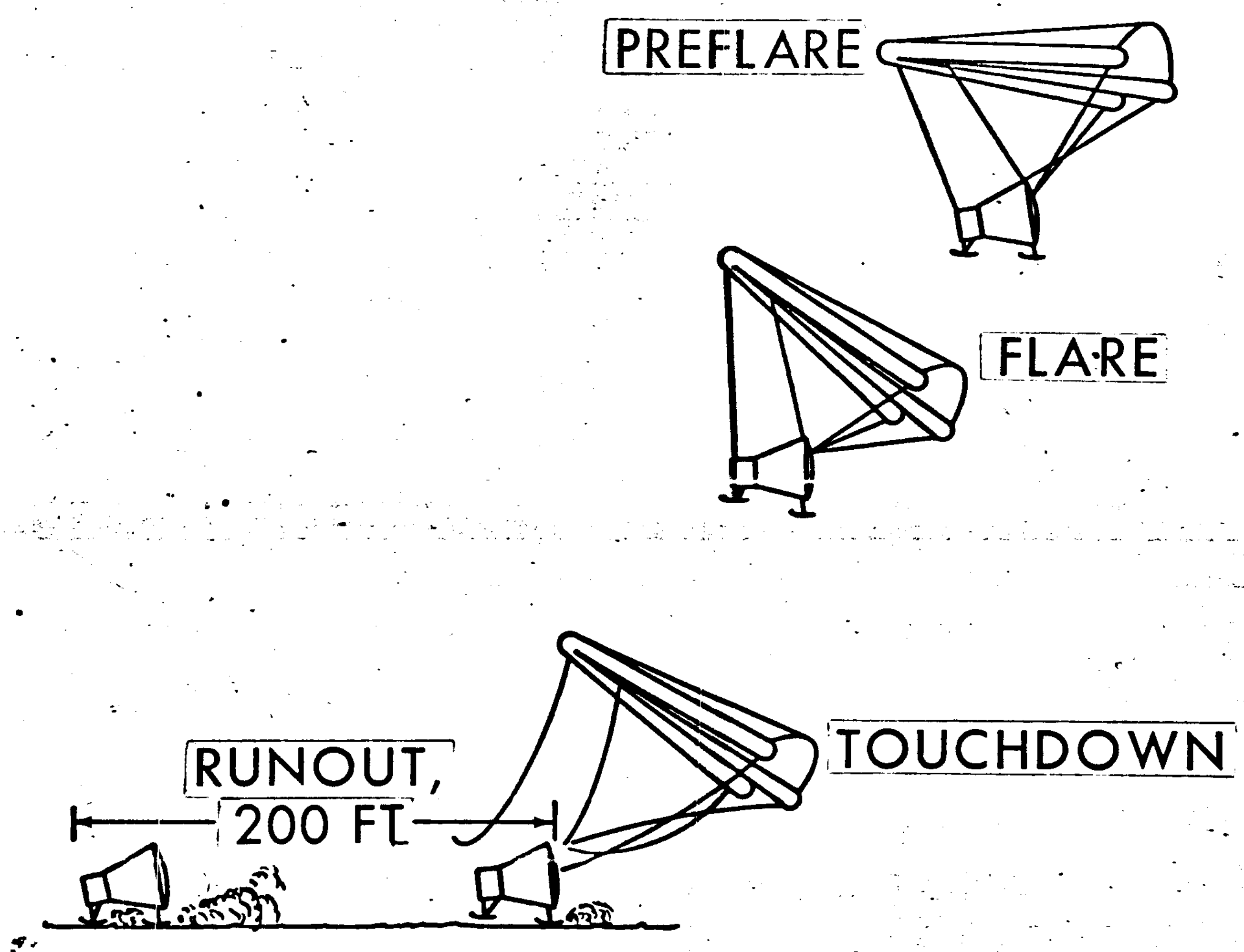

Figure 6.- Gemini paraglider landing phase.

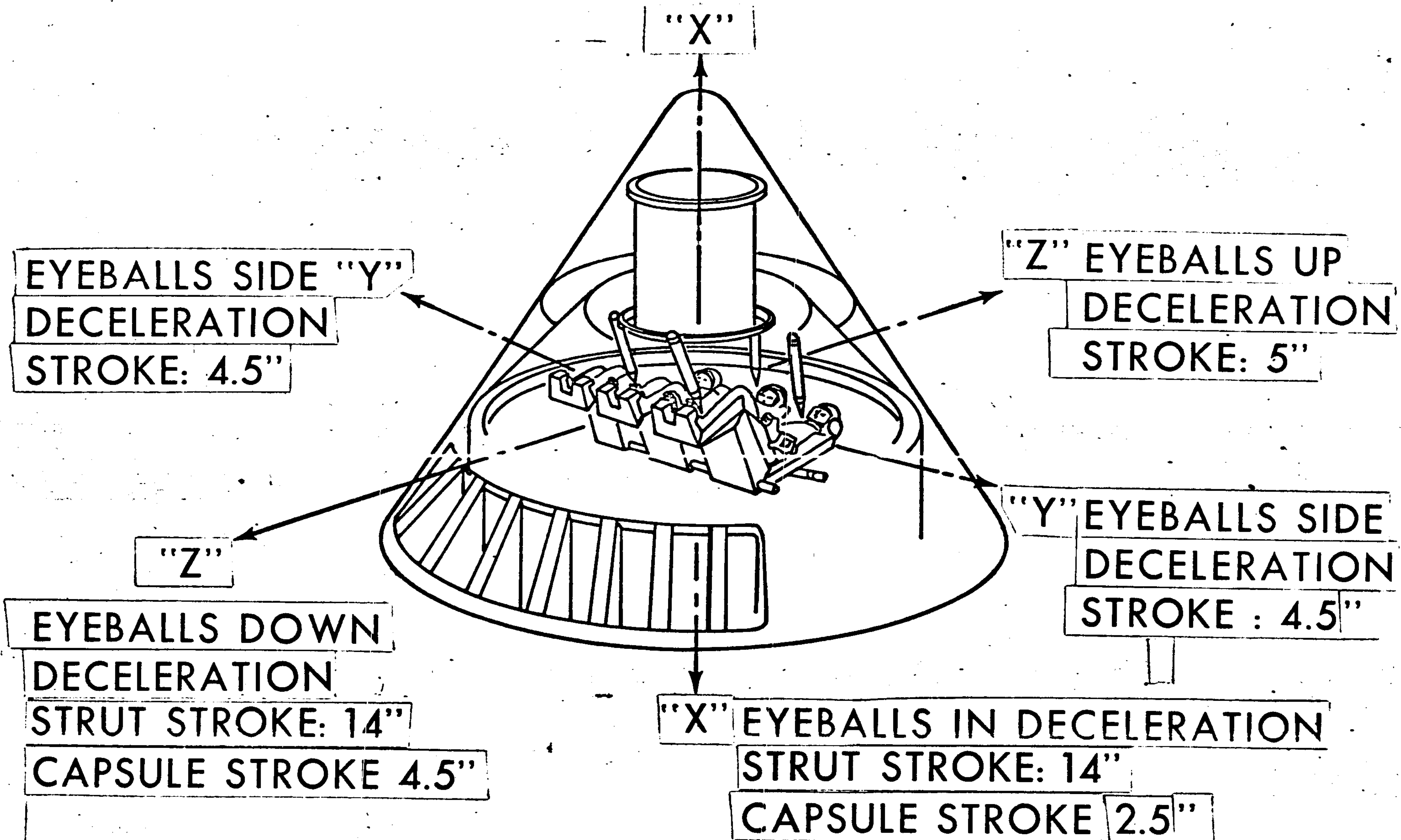

Figure 7.- Apollo attenuating system.

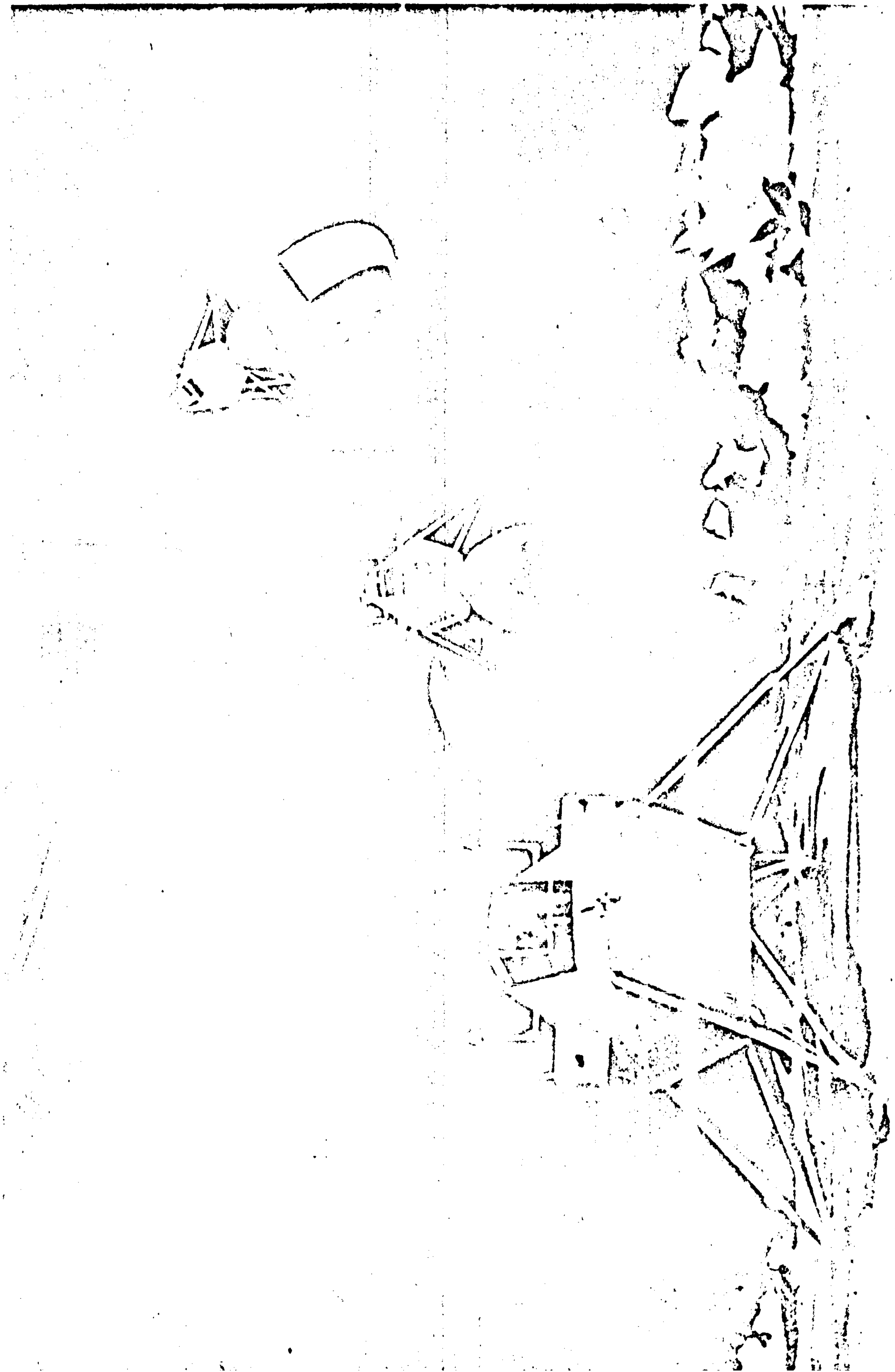

Figure 8.- Lunar landing.