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LANDING IMPACT ATTENUATION FOR NON-SURFACE-PLANING LANDERS

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FOREWORD

NASA experience has indicated a need for uniform criteria for the design of space vehicles. Accordingly, criteria are being developed in the following areas of technology:

Environment
Structures
Guidance and Control
Chemical Propulsion.

Individual components of this work will be issued as separate monographs as soon as they are completed. A list of all previously issued monographs in this series can be found at the end of this document.

These monographs are to be regarded as guides to design and not as NASA requirements, except as may be specified in formal project specifications. It is expected, however, that the criteria sections of these documents, revised as experience may indicate to be desirable, eventually will become uniform design requirements for NASA space vehicles.

This monograph was prepared under the cognizance of the Langley Research Center. The Task Manager was G. W. Jones, Jr. The author was R. H. Jones of Hughes Aircraft Company. A number of other individuals assisted in developing the material and reviewing the drafts. In particular, the significant contributions made by R. J. Black of Bendix Corporation, R. W. Bohlen and G. O. Mount of North American Rockwell Corporation, W. H. Gayman of Jet Propulsion Laboratory, R. E. Hutton of TRW Systems Group of TRW Inc., H. W. Leonard of NASA Langley Research Center, J. J. D. McLaren and J. I. McPherson of McDonnell Douglas Corporation, G. Morosow of Martin Marietta Corporation, W. H. Mueller of Grumman Aerospace Corporation, and D. Nash of Lockheed Missiles & Space Company are hereby acknowledged.

Comments concerning the technical content of these monographs will be welcomed by the National Aeronautics and Space Administration, Office of Advanced Research and Technology (Code RVA), Washington, D. C. 20546.

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LANDING IMPACT ATTENUATION FOR NON-SURFACE-PLANING LANDERS

1. INTRODUCTION

When a space mission requires the landing of a payload, the lander must bring the payload to rest without impairing structural integrity or, if manned, crew safety. The landing should also be accomplished without violating any position and motion constraints. The basic problems at landing are the attenuation of loads and the maintenance of surface clearance and lander motions within acceptable limits. Insufficiently attenuated loads may injure or damage the payload or may prevent payload operation by causing deformation of the lander structure. Insufficient surface clearance could permit an unprotected part of the lander to strike a surface protuberance or penetrate into the surface to a depth that would prevent a successful payload operation. Excessive motions, such as toppling of a nominally stable lander, could produce a final orientation unsatisfactory for payload operation.

No significant failures have occurred during touchdown of any lander (including the Mercury, Gemini, Apollo, and Surveyor vehicles) that has landed within its design constraints, although failures have occurred during some test programs.

Problems during a test or a mission may arise from (1) failure of an attenuator or stabilizer to deploy before landing; (2) degradation of attenuator properties in the pretouchdown environment; (3) out-of-specification touchdown conditions of the lander or of its environment; and (4) inadequacies in modeling, analysis, or test procedures which cause inaccurate predictions of lander performance.

This monograph is concerned with the design of lander attenuation systems and with the analytical and experimental assessment of touchdown dynamics from the moment of surface contact until the lander comes to rest. It presents criteria and recommends practices for attenuator design and for determination of landing decelerations, loads, and stability as needed to ensure structural integrity and acceptability of performance. The monograph is concerned with all landers except those whose primary landing mode consists of surface planing after impact (such as the land-landing X-15 vehicle or possibly surface-planing water landers).

The choice of an attenuation system is influenced by many parameters arising from the environmental history of the lander before touchdown, the environment at touchdown, and the constraints imposed by the payload and other vehicle systems.

The approach advocated herein and generally followed in design and analysis of impact attenuation systems is to choose several candidate design concepts and perform a comparative analysis of the concepts to establish a basis for elimination of all but two or three preferred designs. The preferred designs are then analyzed in detail, and from the results (which may be substantiated and supplemented by test results), one final design concept is determined. The final design is then analyzed in further detail and the design adequacy substantiated with dynamic test vehicles (scaled and/or full size). The accuracy of simulation of both mathematical and test models is of extreme importance in achieving meaningful results.

Related subjects are discussed in other NASA monographs, either published or in preparation. Touchdown conditions are imposed on the lander by descent-deceleration systems. One class of these systems, deployable aerodynamic deceleration devices, will be presented in a monograph now in preparation. Vibrational characteristics, discussed in the monographs on vibration response (in preparation) and natural vibration modal analysis (ref. 1), are useful in determining touchdown loads. At touchdown, shock loads may be imposed on the attenuation system or transmitted to the lander; determination of the response to these loads will be discussed in a monograph on mechanical shock response analysis. Liquid-sloshing loads (ref. 2) and slosh suppression (ref. 3) could affect touchdown dynamics. Thermal protection during entry (ref. 4) can be an important factor in touchdown performance because temperature control of the attenuation system reduces the range of touchdown force levels and may improve landing performance.

2. STATE OF THE ART

Few landing spacecraft have been mission tested thus far, and attenuation-system designs frequently evolve from the experience of individual designers. Design concepts often reflect personal preferences, and include features which affect landing performance in, at best, only a partially understood manner.

The literature generally available as a guide to design is limited. For example, analytical design techniques for attenuation systems of water landers are not readily available; also, a large amount of design literature, existing only in the form of internal documents of various companies, is not distributed throughout the industry. Moreover, the state of the art of attenuation-system design is changing so rapidly that conflicting data may be published. For example, in references 5 to 8, which are standard references used in gasbag-attenuation design, data on the specific-energy-absorption capability of gasbags vary widely. Nevertheless, extremely successful landings have been achieved, both in unmanned and manned spacecraft programs.

The successful manned water landings are evidence of the design adequacy of the impact attenuation systems, which were designed primarily with experimental data. Samples of experimental data used for Mercury and the Apollo Command Module (CM) design are shown in figures 1 and 2. Figure 1 (ref. 9) shows the effect of Mercury touchdown attitude on maximum acceleration at the center of gravity, as obtained from model drop tests. The impact-attenuation system of the Mercury capsule, shown in figure 2 (ref. 9), includes a gasbag as the main attenuator and crushable honeycomb as a supplementary attenuator to reduce loads transmitted to the one-man-crew compartment. Figure 3 shows the effect of touchdown vertical velocity V_n and attitude angle θ on the Apollo CM accelerations at the center of gravity. The $V_n = 30$ fps curve was established experimentally; all other curves are ratioed by $V_n{}^2$ from this baseline. Touchdowns outside the criteria envelope are considered landing failures because they subject the payload to unacceptable load conditions. Figure 3 also shows the CM's 3σ impact boundary that defines a range of impact velocities and attitudes calculated to include 99.7 percent of all CM landings.

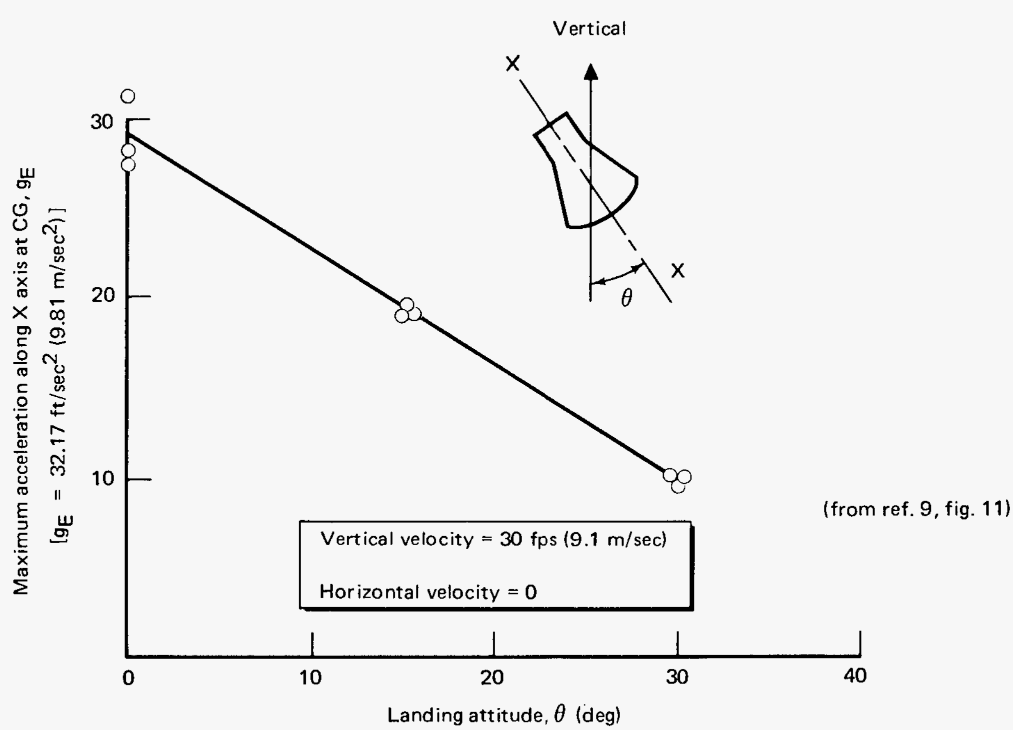
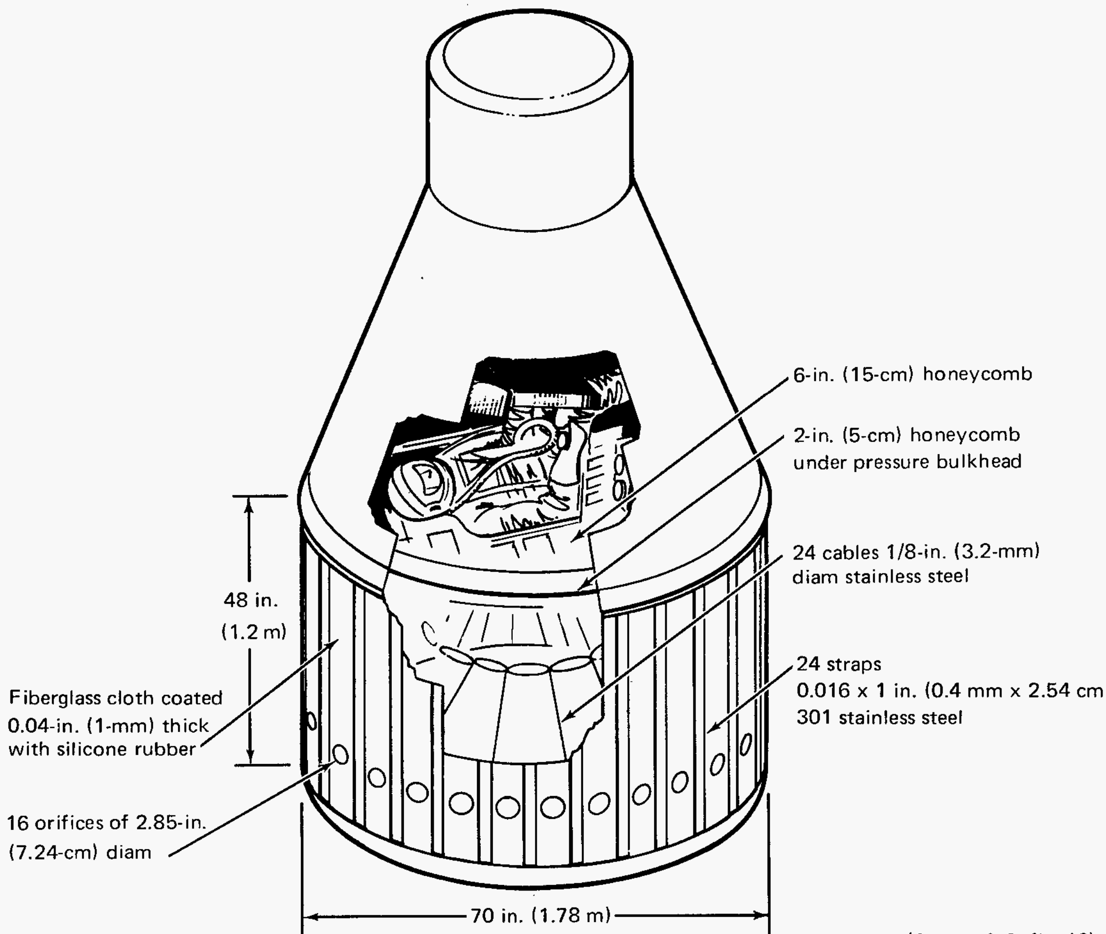


Figure 1. – Maximum acceleration versus landing attitude for Mercury water landing.



(from ref. 9, fig. 10)

Figure 2. – Mercury impact attenuation system.

Design experience has included stable and omnidirectional landers, manned and unmanned landers, and touchdowns on both land and water. The division between stable landers and omnidirectional landers (or omnilanders) is based on their landing characteristics. A stable lander maintains within acceptable limits a preferred orientation relative to the local surface and/or local vertical throughout the landing. It does not topple, and only a limited portion of its peripheral envelope contacts the surface. An omnilander can topple and assume any orientation during landing; but after coming to rest, the lander deploys itself into an operational attitude or configuration. Omnilanders may or may not have a preferred orientation at initial impact. An omnilander with preferred orientation employs descent-attitude control so that it initially impacts the surface on a limited portion of its periphery and thus requires less weight for energy absorption than a lander with nonpreferred orientation.

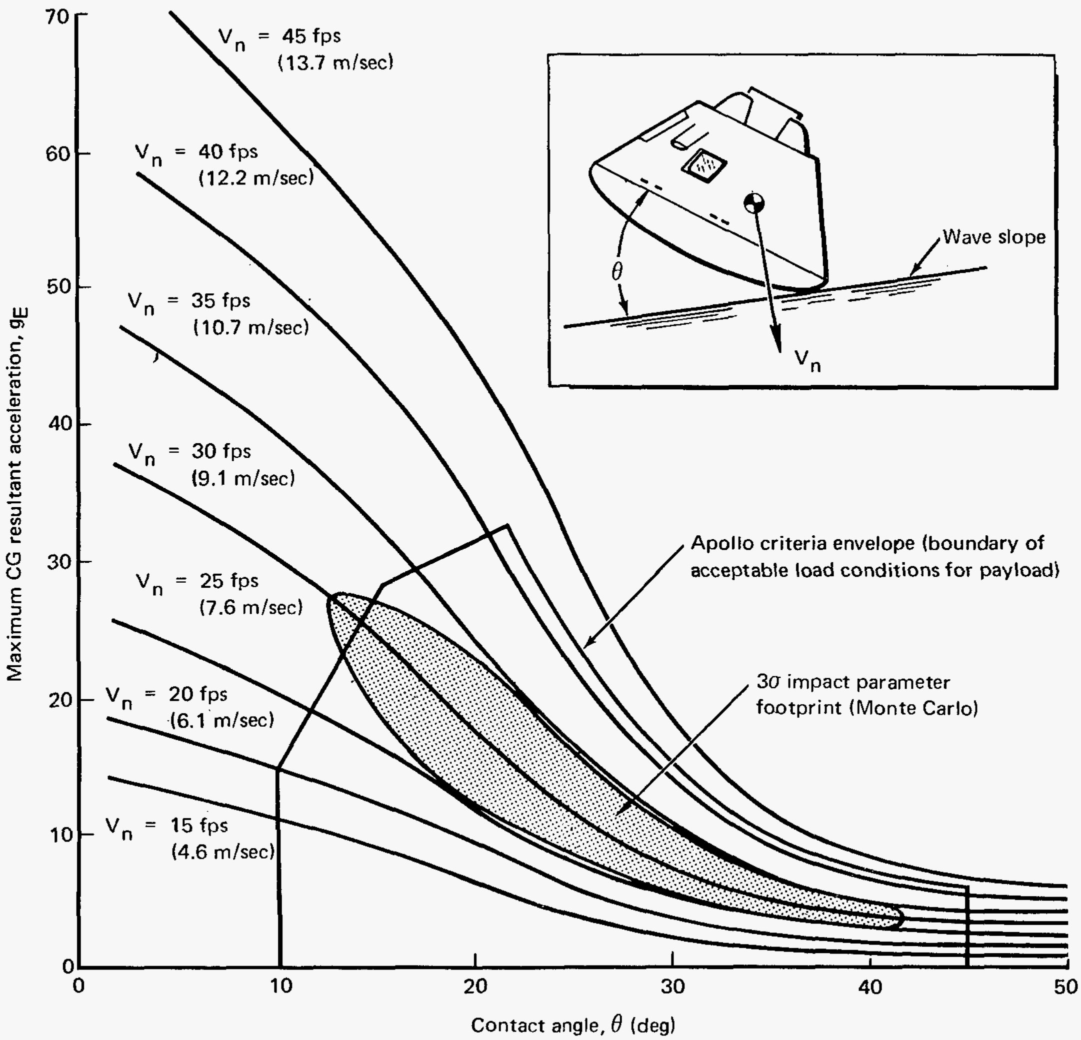


Figure 3. – Maximum acceleration versus landing attitude for Apollo CM water landing.

Spacecraft landers are often described as "soft" or "hard." Hard landers have much higher deceleration levels than soft landers, and undergo permanent deformation of a considerably larger portion of structural material while absorbing impact energy. Hard and soft landers cannot always be distinguished on the basis of impact velocity or stability. Their definition, therefore, requires a degree of arbitrariness which is considered unnecessary for the purposes of this monograph.

The omnilanders and stable landers that provided the design and mission experience upon which the state of the art is largely based are as follows:

1. Ranger Landing Capsule (ref. 10). One of three basic designs, the Block II Ranger was an omnilander with nonpreferred orientation, designed for a

lunar landing with impact velocity up to 200 fps (61 m/sec), and with payload decelerations in the range of 3000 to 5000 g_E ($g_E = 32.174$ ft/sec^2 or 9.8066 m/sec^2). The design objective was to allow the free-floating payload to achieve its desired orientation under the action of gravitational forces after the lander came to rest.

2. Mercury and Gemini (refs. 11 to 15). These manned spacecraft were omnilanders with preferred orientation, designed for injection into earth orbit, followed by entry and landing. Each spacecraft was designed primarily for water landing, with a reduced-performance capability for land landings. The Mercury and Gemini spacecraft carried one and two men, respectively; their respective nominal vertical velocities before impact were 28 fps (8.5 m/sec) and 30 fps (9.1 m/sec).

3. Surveyor (ref. 16). This unmanned spacecraft was designed for a stable lunar landing with vertical-touchdown velocities up to 20 fps (6.1 m/sec). Payload-design decelerations at touchdown ranged from 20 to 40 g_E.

4. Apollo Lunar and Command Modules (LM and CM) (ref. 17). The LM is designed for a stable lunar landing at nominal vertical velocities up to 10 fps (3.1 m/sec), with a two-man crew. The CM is an omnilander with preferred orientation designed for lunar orbit and subsequent earth touchdown on water with secondary capability for land touchdown. It accommodates a three-man crew and descends at a nominal vertical velocity of 32 fps (9.7 m/sec).

2.1 Attenuation System Design Constraints

Many constraints are imposed upon the design of the impact attenuation system by mission-defined environments (e.g., prelaunch, launch, space, landing, and sterilization environments), and by the performance requirements of the lander and payload. The attenuation-system design must be compatible with other space-vehicle systems, and it is further constrained by the materials and manufacturing processes used.

2.1.1 Environmental Requirements

The effects of prelaunch, launch, and other environments that contribute to structural fatigue (refs. 18 and 19) are usually considered in assessing the structural integrity of the lander under landing loads. The effects of space vacuum – cold welding of metals, outgassing of materials, and the breakdown of lubricants and adhesives – are minimized by use of materials and designs developed for this purpose (refs. 20 and 21). Landing-surface characteristics are imposed by the mission. For liquid surfaces, such characteristics may be wave form, velocity, and height, while slopes, protuberances, craters, and bearing strength are the usual landing-surface characteristics of solid surfaces. When sterilization is required, tests are performed to establish compatibility of the design materials and the sterilizing agent. The high temperatures frequently specified for sterilization may prohibit the use of some materials, such as balsa wood. Some sterilization environments are presented in reference 10. Where mission requirements prohibit contamination or erosion of the landing surface, descent-engine thrust levels and cutoff altitudes are established to ensure compliance. Erosion phenomena are described in reference 22, theoretical treatments of erosion are presented in references 23 and 24, and associated experiments are described in references 25 and 26.

2.1.2 Lander-Performance Requirements

Allowable loads of the lander and operational requirements of the payload also serve as constraints on attenuation-system design. For example, allowable instrument-package deceleration levels were 20 to 40 g_E on Surveyor, but as high as 5000 g_E on the Ranger landing capsule. For manned landers, the tolerable load limits for human payloads are presented in detail in reference 27. On most manned landers designed thus far, the payload has been isolated from the structure by an energy-absorption and shock-attenuation system in addition to the lander attenuation system. This design feature increases reliability and appears to be a desirable characteristic for most manned spacecraft.

Payload operational requirements may include appendage deployment, specified orientation, specified surface proximity, and postlanding stability (no spacecraft tottering). These requirements can exert a large influence on attenuation-system design. For example, the three-legged Surveyor landing gear (a typical leg is shown in fig. 4), incorporating pressurized reextendable shock absorbers (ref. 28), was evolved to accommodate payload-orientation and surface-clearance requirements. Later in the program, a simplified payload eliminated these positioning requirements so that a much simpler unpressurized, crushable shock absorber, as used on the LM, would have served the purpose. The hydraulic unit, however, did have the capability to endure repeated loads, thus enabling (1) the flight units to be completely force-stroked in acceptance tests; (2) one set of units to be used throughout a drop-test series; and (3) reuse of the units for the liftoff and second lunar landing achieved by Surveyor VI (ref. 22).

2.1.3 Interface Requirements

The touchdown attenuation system must be compatible with other space-vehicle systems. For example, restrictions are imposed on attenuation systems of both the LM and Surveyor types of vehicles by the shroud dimensions of the launch vehicle. To obtain required leg length for stability, but stay within the shroud dimensions, leg assemblies were developed that could be stowed and then deployed with a high degree of reliability following the shroud separation (figs. 4 and 5).

Thermal-interface problems can also be of consequence. For example, during descent of the Surveyor, fiberglass shields were used to prevent damage caused by impingement of exhaust plumes on the crushable blocks of the landing gear. On the LM, where the descent rocket may still be thrusting at touchdown, shielding was used to protect against a potential shock-absorber-binding problem from the thermal effects of the exhaust plume reflected from the surface.

Additional interface effects which may occur during lander touchdown are engine-nozzle crushing resulting from surface impact, nozzle choking, and reflected pressures on the lander base, as on the LM (ref. 29). When analyzing lander performance, the forces associated with these effects must be considered together with attenuation-system forces. Figure 6 shows the trends of the variations in engine-thrust level and integrated base pressure as the LM nears the surface.

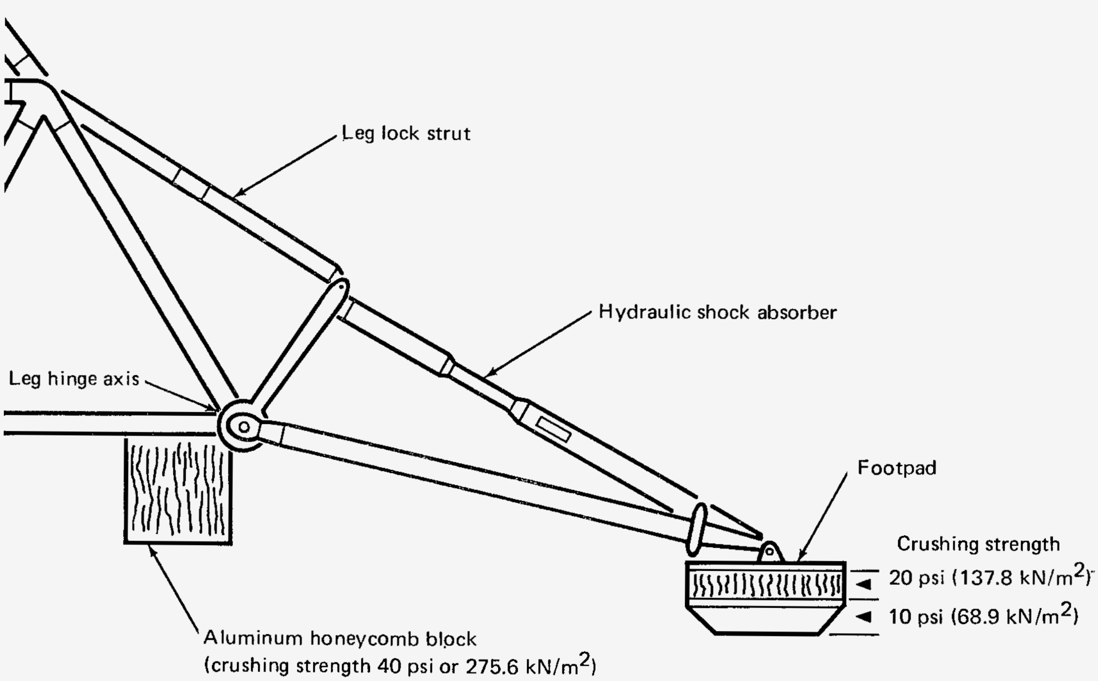


Figure 4. – One-leg assembly of three-legged Surveyor landing gear (in extended position).

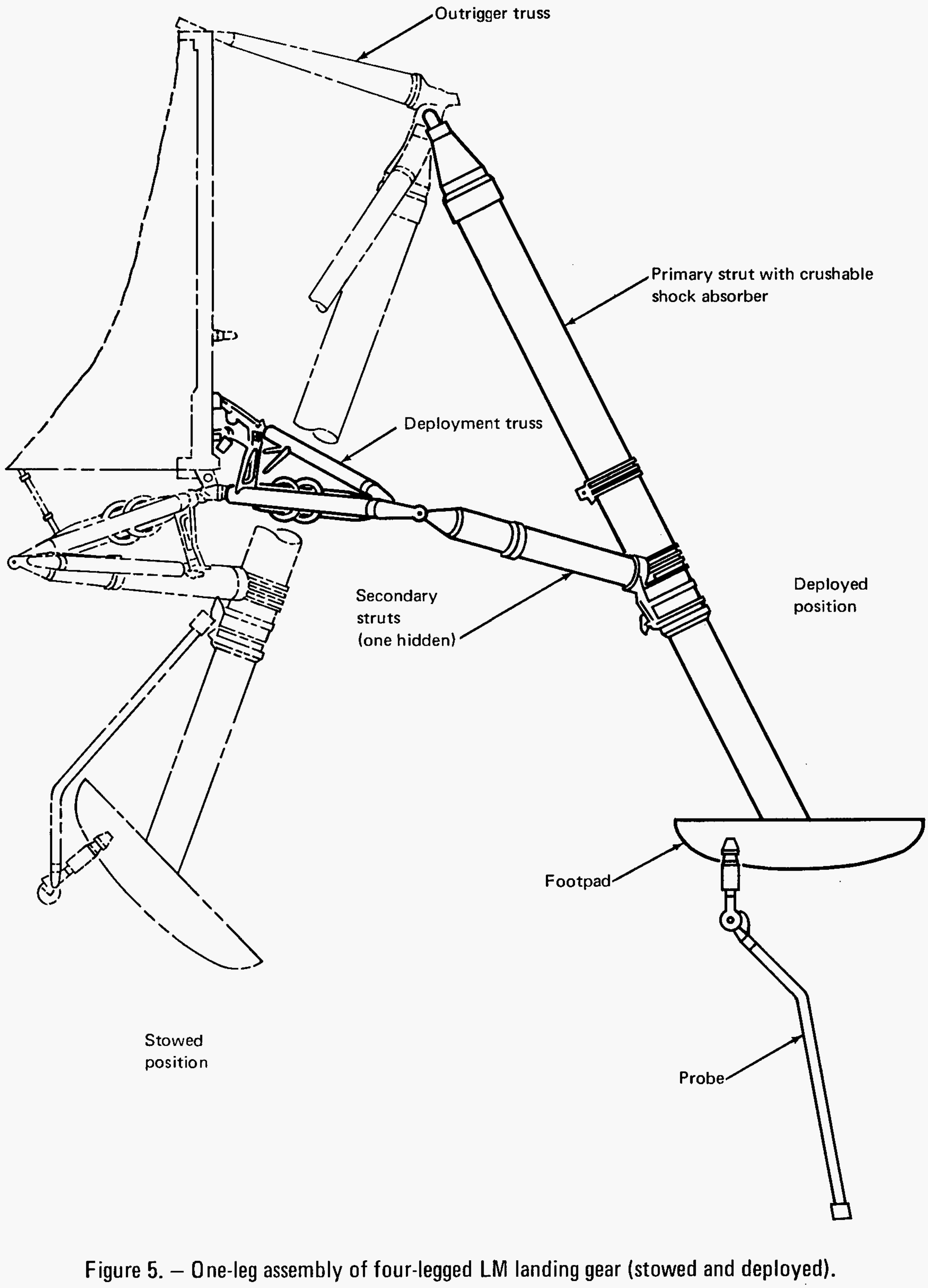


Figure 5. – One-leg assembly of four-legged LM landing gear (stowed and deployed).

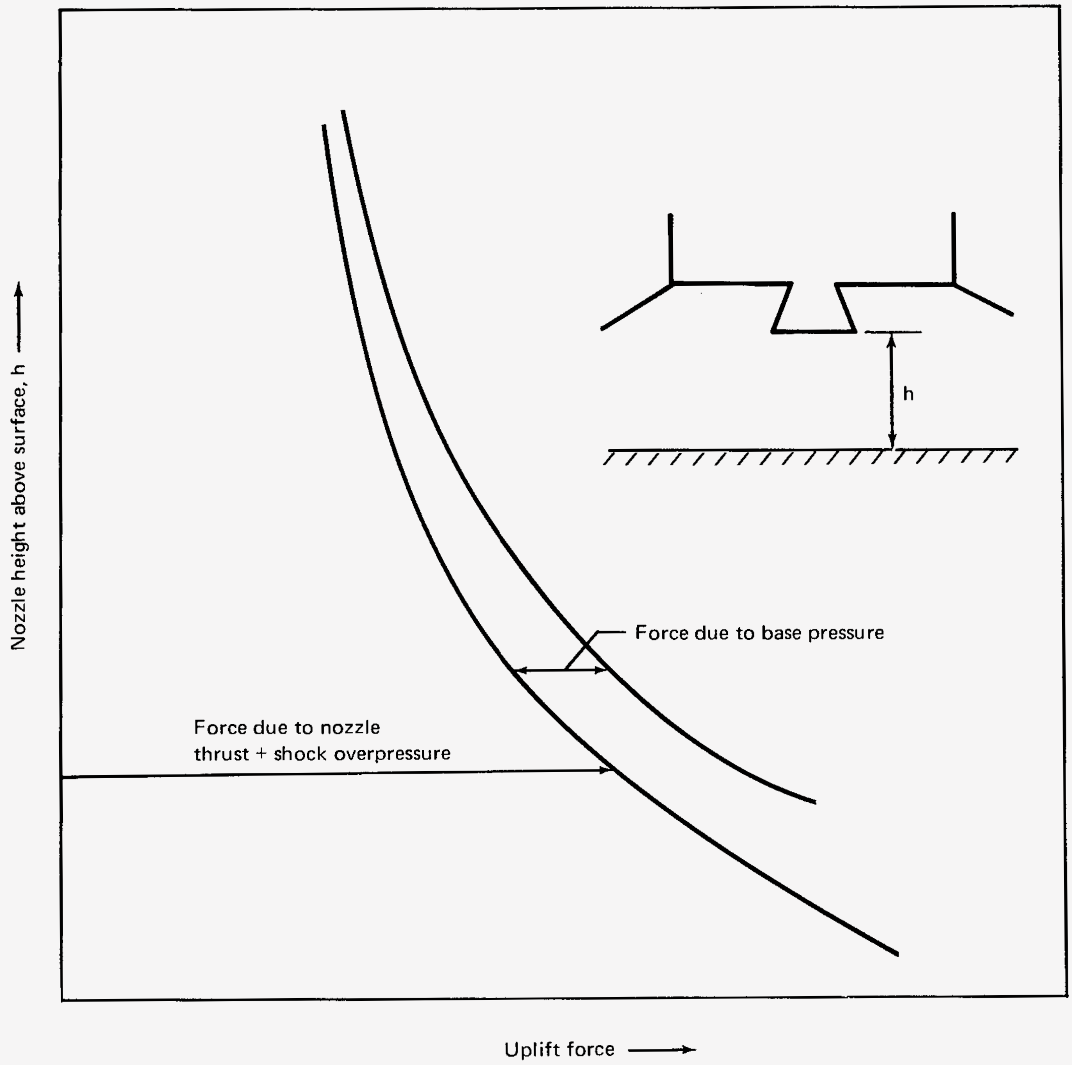


Figure 6. – Schematic rendering of the uplift force on the LM near flat lunar surface.

2.1.4 Material Selection and Fabrication Requirements

The characteristics of materials and fabrication processes constrain touchdown-attenuator design. For example, where crushable honeycomb material is used as an energy absorber, its nonideal load-deflection characteristics (ref. 30, fig. 2) may be substantially improved by small amounts of precrushing to "remove" the initial high-load peak. Material characteristics are usually measured under simulated mission conditions to ensure acceptable performance.

Variations in manufacturing tolerances with different materials are considered in design analysis because they directly affect performance characteristics. Where close tolerances are mandatory, as on the Apollo CM attenuation system, exacting quality-control procedures have been applied to production. The demand for extreme precision in manufacturing and its associated quality assurance can cause rapid escalation of costs.

2.2 Touchdown-Dynamics Analysis

2.2.1 Input Parameters

Analysis of touchdown-dynamics performance is initiated by defining values for the following:

- Spacecraft touchdown conditions, such as velocities, orientations, mass, and geometry.

- Local environmental conditions, such as gravitational force, surface slopes, protuberance height, and bearing strength, or, for liquid surfaces, wave form, height, and velocity.

- Attenuator characteristics, such as force-stroke profiles, anisotropy, and velocity sensitivity.

Some values of these parameters which have been used in space vehicles are given in tables I to III. Table I presents some touchdown conditions for existing spacecraft; tables II and III present environmental data for solid-surface and water landers, respectively. References 6 and 30 to 38 show energy-absorption characteristics for a wide range of attenuators.

For omnilanders with high impact velocities and impact-deceleration levels (refs. 10, 33, 36, 37, 39, and 40), there are two principal design conditions: (1) the maximum-velocity impact on a rigid surface, for which loading and energy-absorption characteristics are determined; and (2) the maximum-velocity impact on the softest specified surface, from which the depth of penetration is determined. Structural-response loads are determined on the basis of the single impulse experienced during the maximum-velocity impact. Thus, relatively few impact conditions are needed for adequate assessment of design and performance. To date, omnilander design has tended to ignore the possibility of the impact of a protuberance, which can severely reduce the capability of landing useful payload weights (ref. 41). This practice is justifiable only if it can be shown that the probability of impact of a protuberance is negligible.

TABLE I. – LANDER TOUCHDOWN CONDITIONS

Spacecraft	Expected mean vertical velocity[a]		Design conditions[b]					Approximate touchdown mass	
			Vertical velocity		Horizontal velocity		Touchdown attitude[c]		
	fps	m/sec	fps	m/sec	fps	m/sec	deg	slugs	kg
Ranger capsule	105	32.0	200	61.0	100	30.5	N/A	2.8	41
Mercury	28	8.5	28	8.5	0 to 51	0 to 15.6	0 ± 15	83	1210
Gemini	30	9.1	30	9.1	0 to 51	0 to 15.6	55 ± 15	135	1970
Surveyor	12.7	3.9	20	6.1	7	2.1	0 ± 8	20	242
Apollo LM	N/A	N/A	10	3.1	4	1.2	0 ± 7	484	7063
Apollo CM	30.4	9.3	34	10.4	0 to 51	0 to 15.6	27.5 ± 8	336	4904

[a]Arithmetic mean of predicted touchdown velocity.
[b]Highest value for which parts of the spacecraft were designed.
[c]Angle between the roll axis of the spacecraft and the local vertical.

TABLE II. – SOLID-SURFACE DESIGN PARAMETERS

Spacecraft	Design parameters
Surveyor	Slopes, <15 deg; protuberances, <10 cm high; surface hardness from 50 psi (345 kN/m²) to 25 000 psi (17.2 MN/m²); surface-friction coefficient from 0 to 1.0. Performance data were also required on a soft surface of bearing strength 0 at the surface and increasing linearly at 10 psi (639 N/m²) per foot (30.8 cm) of penetration.
Apollo LM	Surface mean slope <6 deg; maximum effective slope, including protuberances, depressions, and differential footpad penetrations, 12 deg. Distance from the top of the highest protuberance to the bottom of the lowest depression does not exceed 24 in. (61 cm). Surface-bearing strength at the bottom of a 24-in. depression, or footpad penetration depth, >12 psi (82.7 kN/m²). Surface-friction coefficient varies from 0.4 to complete constraint.

TABLE III. – SURFACE PARAMETERS FOR WATER-LANDER DESIGN

Spacecraft	Water surface parameters		
Gemini	Maximum wave slope, 9 deg		
	Quantity	Frequency distribution	
		99 Percentile	100 Percentile
Apollo CM	Wave slope, deg	7.2	12.5
	Wave velocity, fps (m/sec)	40 (13.1)	40 (13.1)
	Wind-to-wave direction, deg	27.5	170

For stable landers or omnilanders with preferred orientation, angular motions at impact are of primary concern; therefore, design assessment considers all possible combinations of touchdown conditions. References 28 and 42 present rigid-body (i.e., a rigid lander attached to a realistically modeled attenuation system) performance predictions for the Surveyor and LM.

Figure 7, taken from reference 42, shows several performance boundaries calculated for the LM lander. In the figure, each boundary is related alphabetically to a depicted landing condition and the associated critical parameter (e.g., strut stroking or stability) for the landing condition is indicated. Touchdown velocities above the boundaries result in overstroking or instability; velocities below the boundaries result in acceptable landing performance.

The results of extensive test programs and several missions show that the analytical techniques developed to predict stable lander performance are highly reliable. For example, analytical and experimental results for landing stability and for a typical lander structural-response load are compared in figures 8 (ref. 43) and 9 (ref. 44), respectively. Figure 10 compares calculated and measured landing-gear load levels for the lunar landing of Surveyor VI.

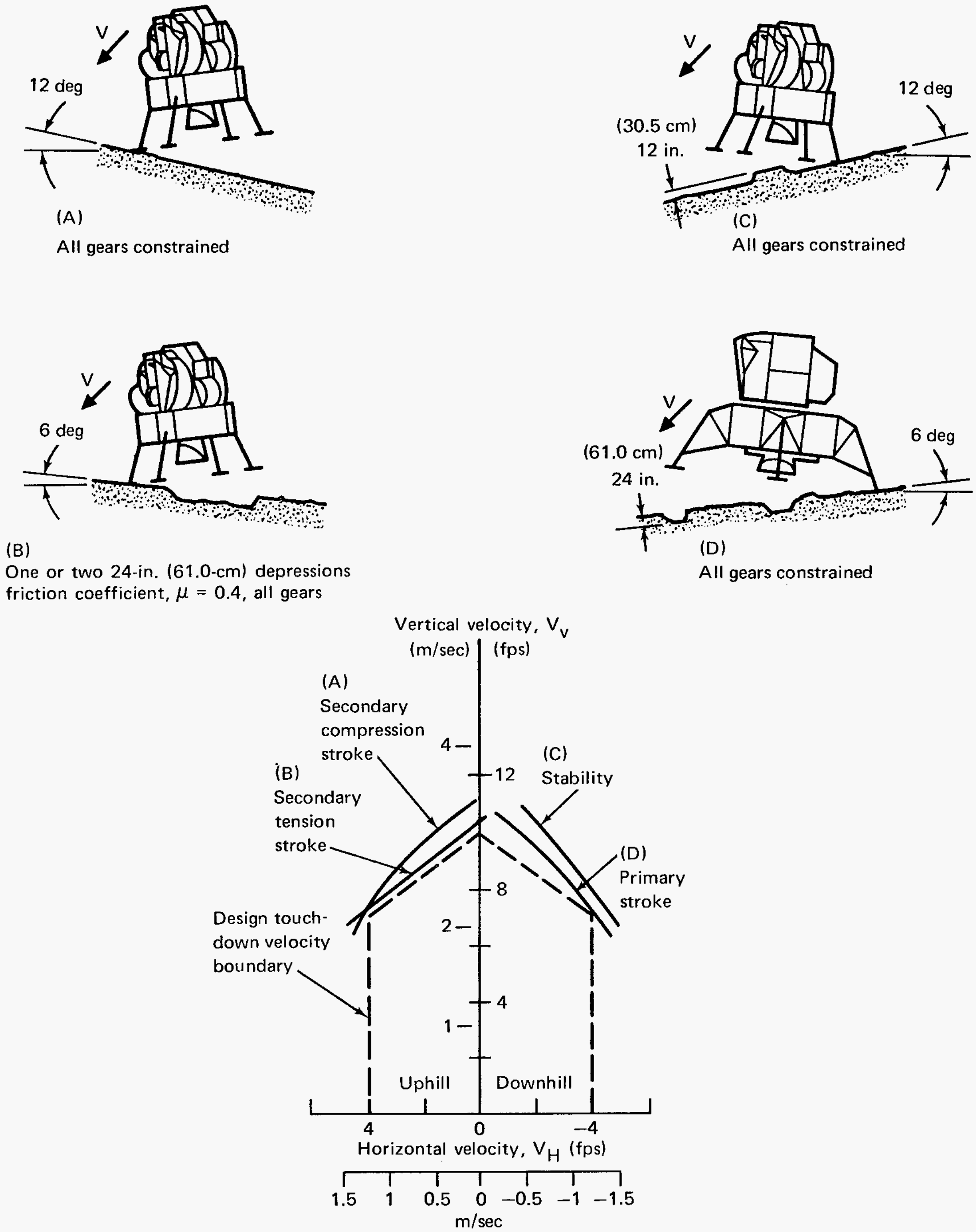


Figure 7. – LM unsymmetric-landing performance.

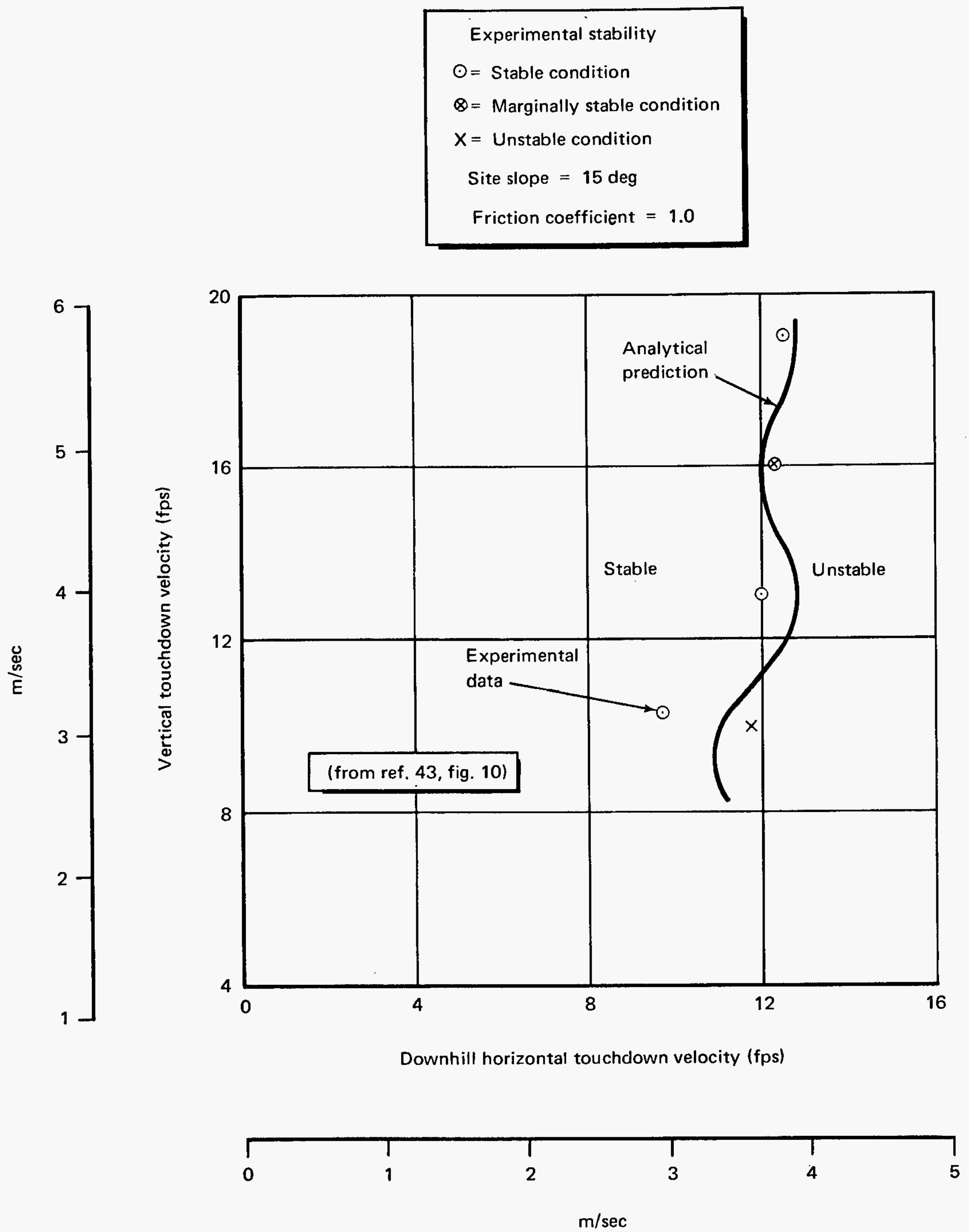


Figure 8. – Comparison of stability boundaries from analytical and test results.

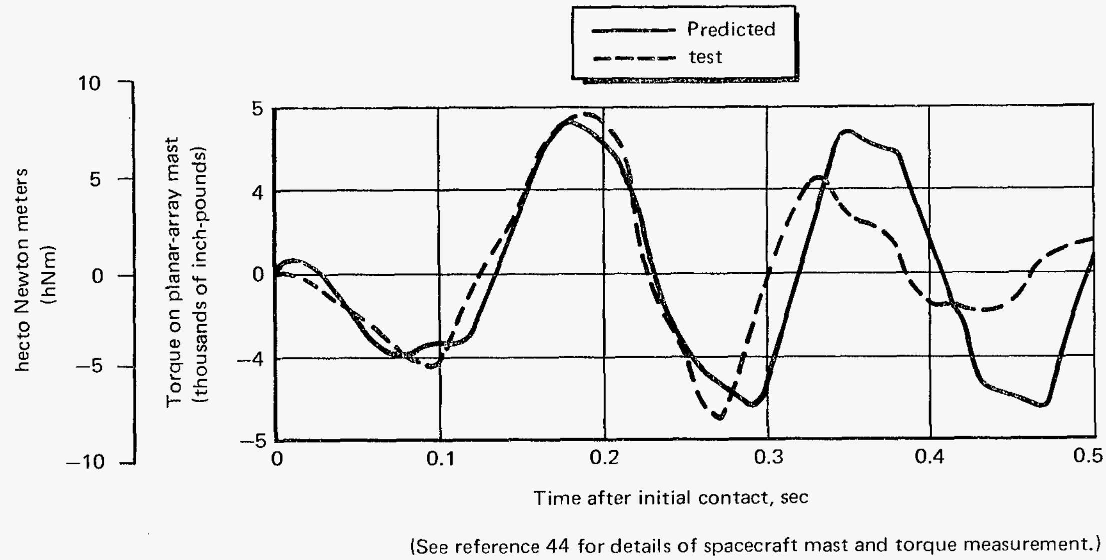


Figure 9. – Comparison of planar-array mast torsion on Surveyor from test results and analytical predictions.

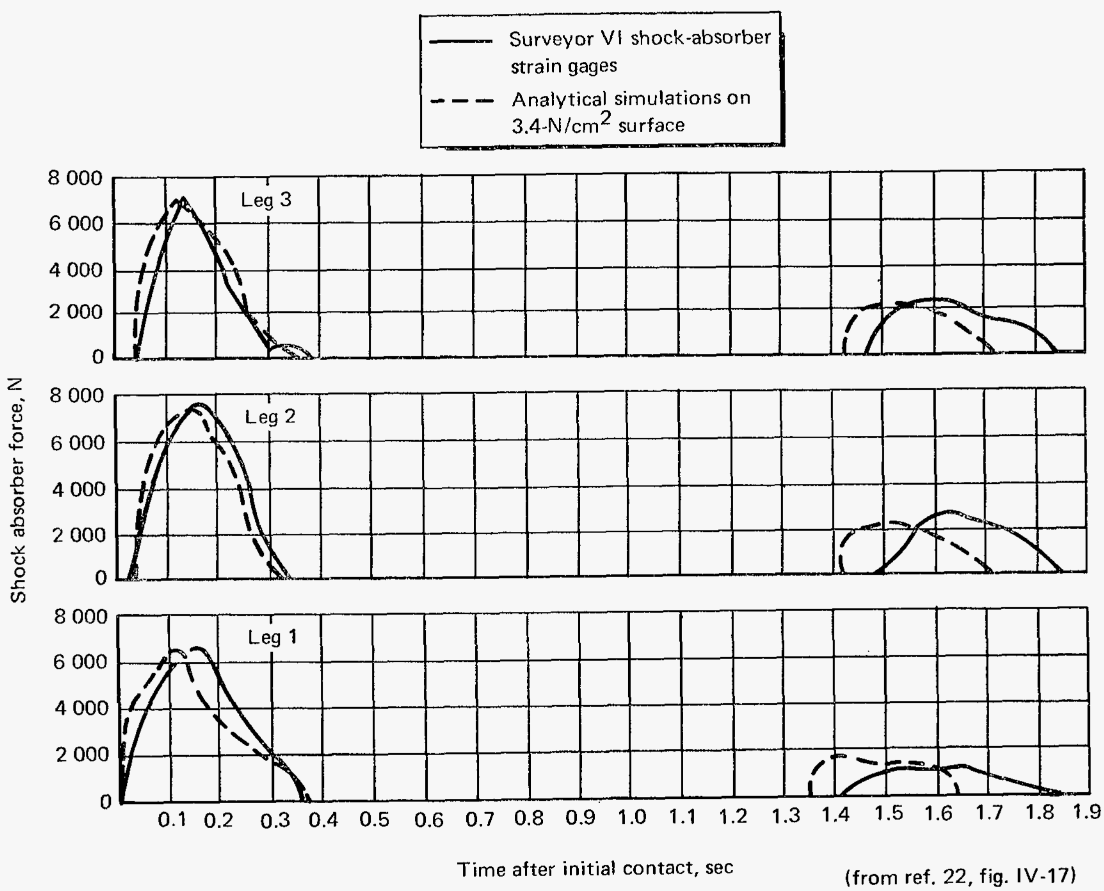


Figure 10. – Comparison of analytical and measured shock-absorber-force histories for Surveyor VI landing.

2.2.2 Analytical Methods

Most design calculations assessing landing dynamics are sufficiently complex to require the use of high-speed digital computers. Some simplified approaches to establishing preliminary structural design parameters for landing attenuation systems are contained in reference 43; this reference also shows that simplified computer analyses using closed-form equations of motion are inadequate to provide accurate performance predictions. Most computer programs and techniques (e.g., refs. 28, 36, 37, 40, and 45 to 48) use numerical integration to achieve solutions; some incorporate error-check procedures to control cumulative errors within predetermined limits. In addition to error control, the error-check procedure can be a valuable aid in "debugging" a program during its development stage. The advantages of error-check procedures are generally offset by a several-fold increase in computational time. When error-check procedures are not incorporated (e.g., on Surveyor design analysis), computer programs are developed with detailed attention to the sensitivity of results to changes in integration interval and integration method.

Analytical predictions of lander performance can be obtained by one of two methods: the statistical method or the absolute-performance method. In the statistical method (refs. 49 to 51), a probability-density function is defined for each touchdown condition and attenuator characteristic; then Monte Carlo techniques are used to establish a probability of successful landing. In the absolute-performance method (refs. 28 and 42) used for Surveyor and LM design, it is required that no landing failure occur for any touchdown condition within a specified range. These two methods can be combined to establish conditional probabilities of successful landing.

Advantages of the absolute-performance method are that (1) once the worst initial conditions within the given range of touchdown conditions are established, they are not likely to change appreciably during vehicle design, fabrication, and testing; (2) critical conditions can be clearly established for test purposes; and (3) the effect of spacecraft-design changes on performance can be rapidly assessed. The main disadvantage is that by combining worst conditions (not necessarily maxima or minima) of all input variables, the landing system can be grossly overdesigned from the viewpoint of actual touchdown requirements.

The statistical method has the advantages of (1) reducing the degree of overdesign of the landing attenuation system and (2) establishing a probability of successful touchdown which can be compared and combined with the probabilities of success of other spacecraft subsystems to predict a probability of mission success. A major disadvantage is that accurate probability-density functions for each landing parameter usually cannot be generated until a lander design has been developed to a condition where design details and parameters are well defined and understood. Another disadvantage is the cost of computer time required for the large number of touchdown simulations needed to achieve a successful touchdown-probability prediction within acceptable confidence limits. Requirements for computer time can become exorbitant when assessing the effects of changes in design or touchdown conditions. Thus, purely statistical analyses are normally not performed until designs have been well established. As an alternative, designs can be assessed by obtaining conditional probabilities of landing success, in which some conditions (e.g., landing velocity) are assigned specific values instead of probability-density functions.

2.2.3 Mathematical Models

Analyses are usually performed with two distinct mathematical models of the lander: a rigid-body model and an elastic model.

Both the rigid-body model and the elastic model can be either two-dimensional or three-dimensional. The two-dimensional model (ref. 28) allows three degrees of freedom (two translations, one rotation) of the spacecraft's center of gravity, and is used to simulate planar landing conditions. In these conditions, the spacecraft is constrained to move in a vertical plane containing a spacecraft plane of symmetry and the line of maximum surface slope. The three-dimensional model (ref. 45) allows six degrees of freedom (three translations, three rotations) of the spacecraft's center of gravity, and is used to simulate both planar and cross-hill landings with unrestricted spacecraft orientation relative to the local surface.

Where spacecraft elasticity can be shown to have a negligible effect on the performance parameters under investigation (such as attenuator forces and deflections, spacecraft stability, or surface penetration), the spacecraft is modeled as a rigid body attached to an attenuation system which simulates the known force-deflection characteristics of the actual attenuation system. Use of these simpler rigid-body models, when justified, can result in large cost reductions. To determine structural-response loads, an elastic model containing the structurally significant flexibility of the lander is combined with the rigid-body model, as described in reference 44.

Solid landing surfaces are also modeled in two distinct ways: as a rigid, unyielding surface, or as a soft surface. When a lander is designed to strike a rigid surface, a friction coefficient or rigid abutment is used to determine forces tangential to the surface. When a lander is to penetrate a soft surface, the model is designed so that penetration forces can be determined. These forces vary with the area, direction, depth, and velocity of penetration, and the penetration characteristics of the surface (refs. 52 to 55). Essentially the same mathematics can be used for soft-surface penetration as for water landers. The simple soft-surface models described in references 52 to 55 are satisfactory for predicting touchdown performance.

2.3 Testing

Because of the short history of spacecraft-impact-attenuation design, the heavy reliance upon experimental data, and the variety of spacecraft and missions involved, a substantial amount of testing is normally performed at various stages of attenuation-system design to assess structural integrity and performance predictions.

2.3.1 Design-Development Testing

Tests of components, such as shock absorbers or footpads, are performed to establish structural integrity and material characteristics. It is frequently the practice to determine some characteristics, such as energy-absorption capability, in atmospheric conditions, even though the absorbers may be required to function outside the atmosphere. One problem that results is entrapping and compressing air during absorber deflection; the effects of air entrapment can significantly influence energy absorption, especially in high-velocity impacts. Suitable venting, as on the Surveyor aluminum honeycomb blocks, provides a satisfactory method of alleviating this problem for low-velocity impacts. Another significant problem when hardware is earth tested is control of mechanical friction. Frictional characteristics in space are maintained within acceptable levels by use of space-approved materials and lubricants (ref. 21).

During design and development of attenuation systems, tests are also performed at the subsystem and system levels to substantiate analytical predictions and to refine and correct analytical models. When these tests are performed before flight hardware becomes available or when it is impractical to test the flight hardware, scale models, including full-scale models, are utilized. Scale-model tests of Surveyor, Apollo CM, and of Apollo LM-type landers are described in references 28 and 56 to 59. The techniques of spacecraft modeling and the associated problems, including that of achieving proper mass and inertia scaling of the impact attenuation system, are discussed in reference 60. The model's weight-distribution problem becomes acute with omnilanders which have a fairly high percentage of the total spacecraft weight in the attenuation system.

To achieve appropriate scaling of structural elasticity, which may have a significant effect on landing performance, it would be necessary to add to the scale factors of reference 60 the requirement that $E_{full\ scale}/E_{model} = \gamma/\alpha\beta$ where E is the Young's modulus of structural material; α, the ratio of full-scale acceleration to model acceleration; β, the ratio of full-scale mass to model mass; and γ, the ratio of full-scale length to model length.

This additional scaling, if required, would add considerable complication to the problem of true dynamic scaling, as would the incorporation of descent-engine plume/surface interaction effects. It is the usual practice to omit these effects as part of model scaling, but their influences must be assessed and taken into consideration in the interpretation of test results.

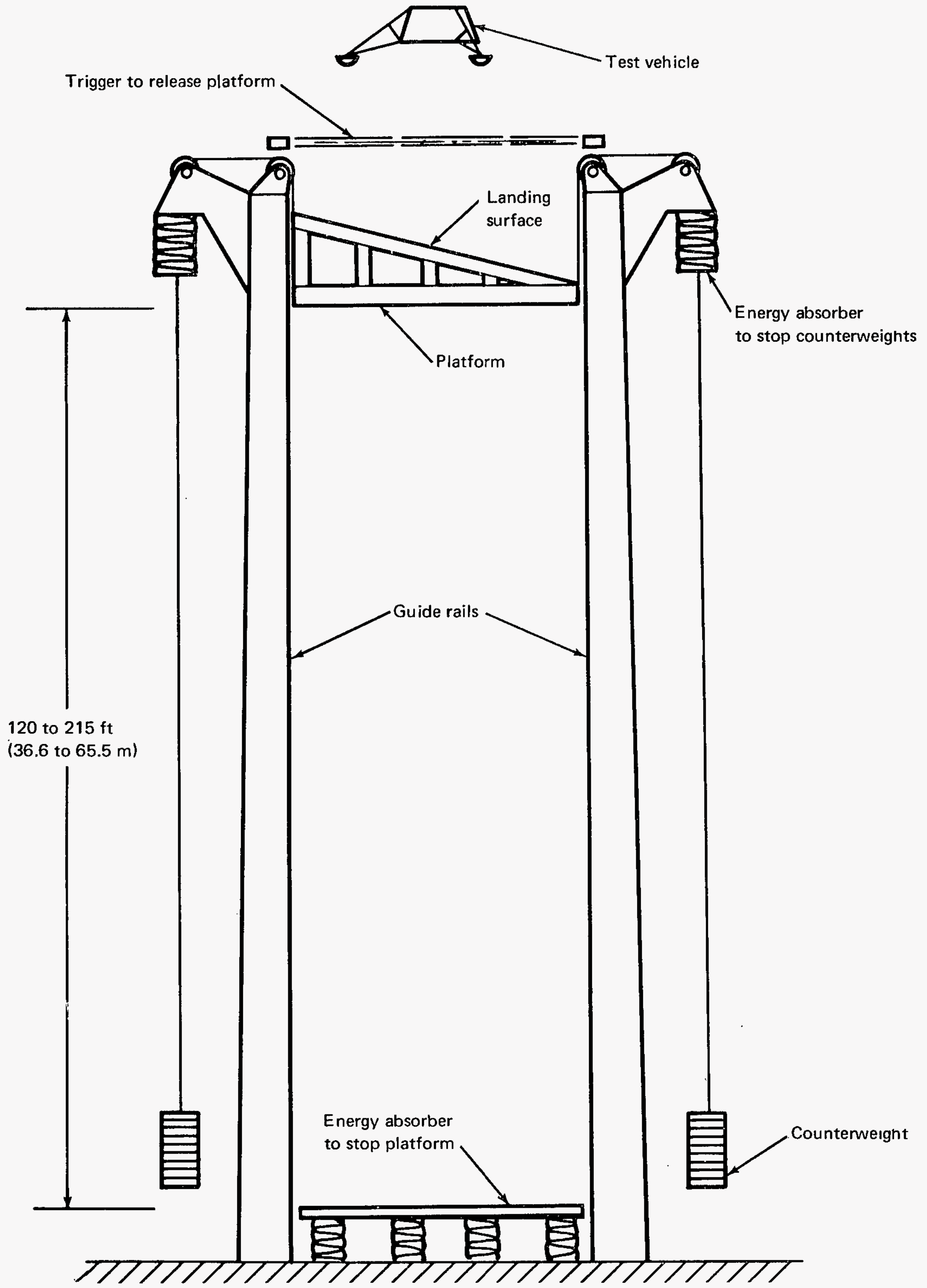
The model's mass distribution can be more realistic if the desired gravitational force field is simulated dynamically in one of three ways: (1) the falling-platform method presented in figure 11; (2) the gravity-component method shown in figure 12; or (3) the lifting-force method described in figure 13. The methods are well known and are described in detail in references 28 and 61.

Of these methods, the most realistic simulation is achieved by the falling-platform method, since the model is unencumbered by restraining cables. However, this method is by far the most expensive to implement and requires extremely tall test structures. The lifting-force method has advantages over the gravity-component method because (1) it simplifies the problem of model setup and handling, (2) it enables simpler fabrication of a rigid platform for rigid-surface simulation, and (3) it enables the simulation of soft surfaces – which would be extremely difficult, if not impossible, in the gravity-component method. However, gravity simulation is normally unnecessary for hard omnilander concepts – omnilander tests are described in reference 10 – or for testing of stable spacecraft to determine maximum loads, where equivalent earth-impact conditions can be used to produce maximum planetary-landing loads (ref. 44).

For tests that establish spacecraft-performance characteristics not affected by spacecraft elasticity, the easiest approach is to fabricate models which represent the attenuation system realistically, but which incorporate a relatively rigid representation of the mass and inertia of the remainder of the spacecraft. When feasible, this is a desirable procedure that simplifies tests and significantly reduces their costs. When the model must simulate a flexible spacecraft, as in tests for maximum structural loads, some simplification and cost reductions are possible by representing certain payload elements by rigid masses with appropriate inertias. This is particularly justifiable when future detailed qualification tests are planned.

2.3.2 Qualification Testing

Lander-qualification tests are performed on flight-quality hardware produced to manufacturing specifications to demonstrate that design-performance goals have been achieved and to determine structural integrity. These tests subject both components and full-scale spacecraft to loading and environmental conditions which usually exceed those anticipated for the service life of the lander. Hardware used in qualification tests is generally not designated for use in subsequent missions.



(from ref. 28, fig. 105)

Figure 11. – Dropping-platform method of lunar-gravity simulation.

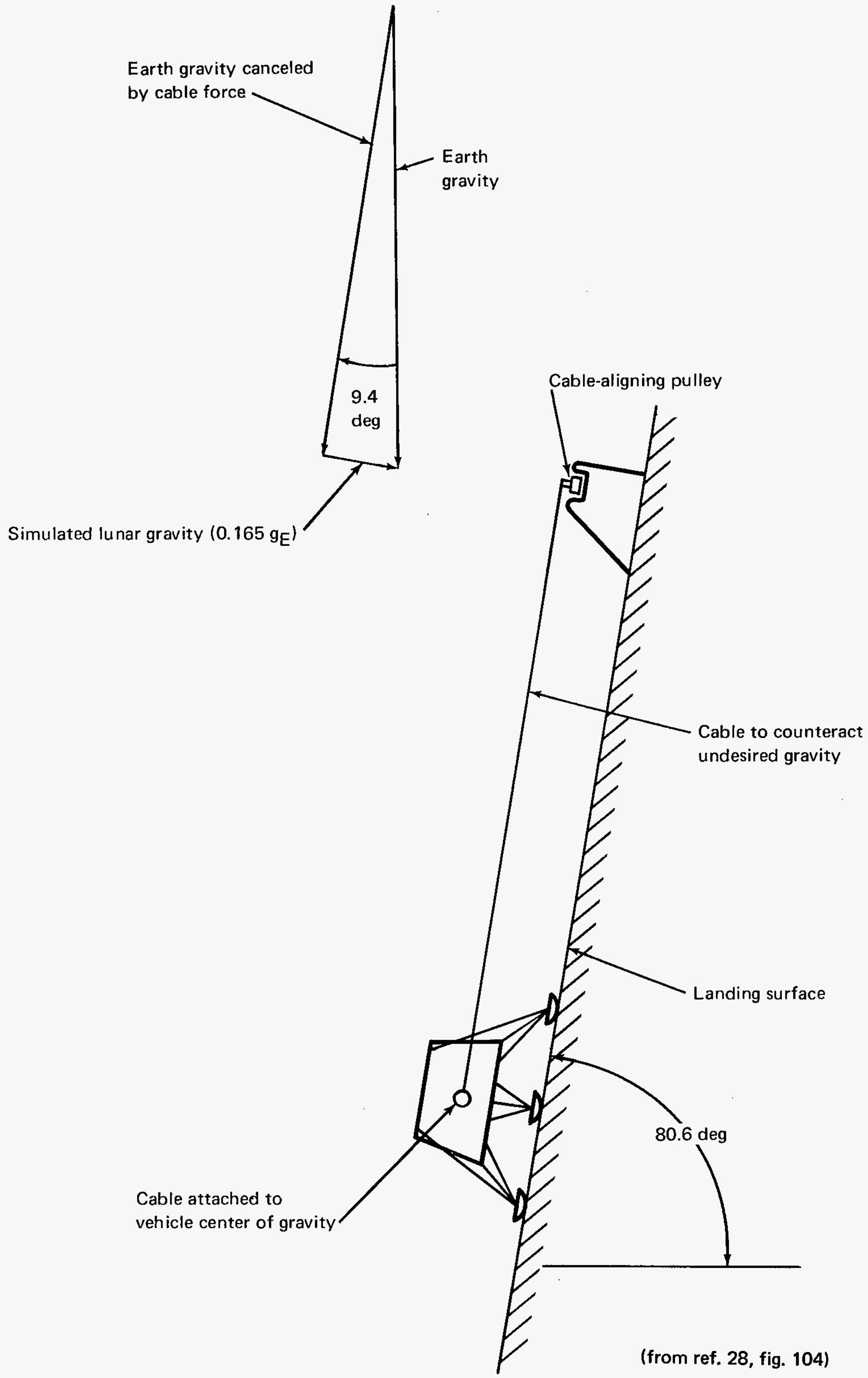
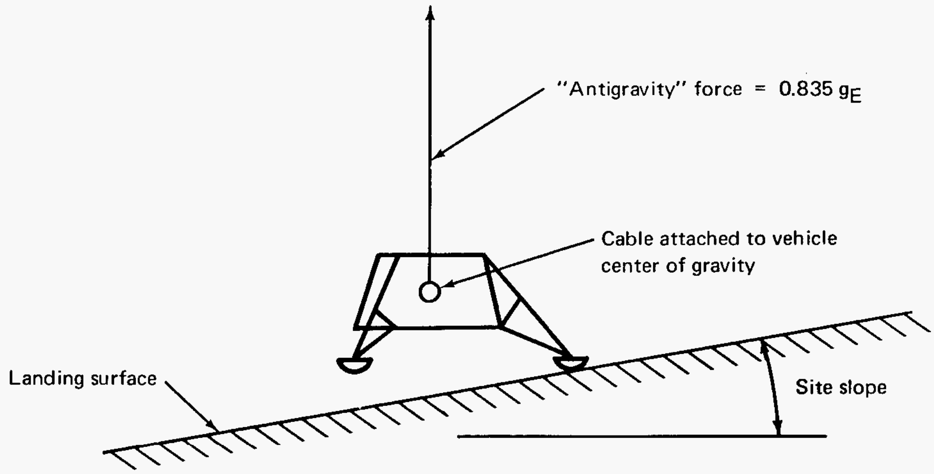


Figure 12. – Gravity-component method of lunar-gravity simulation.



(from ref. 28, fig. 106)

Figure 13. – Lift-force method of lunar-gravity simulation.

The landing attenuation system is usually qualified both separately and in qualification tests of the complete lander system. For example, if the attenuation system is a legged landing gear, a complete leg assembly may be qualified as a unit. The assembly will be first subjected to qualification loads appropriate to the launch and ascent phase of the flight. These loads include vibration, acceleration, and shock, and are obtained from spacecraft-qualification loads for the particular boost vehicle (such as those specified in ref. 62) multiplied by the measured or estimated transmissibilities at the attach points of the leg assembly to surface frame (ref. 62). The leg assembly may then be subjected to several thermal vacuum cycles that duplicate the character and sequence of anticipated flight environments. Finally, the assembly is subjected to qualification loads that simulate the landing loads.

The magnitude of qualification loads depends on the design-analysis method that is used: the statistical method or the absolute-performance method (Sec. 2.2.2). For the absolute-performance method, qualification loads are the maximum loads that can be predicted for all combinations of landing parameters within the specified ranges of the parameters. For the statistical method, qualification loads are obtained by statistical analysis and may be determined, for example, so that the probability of the landing loads exceeding (or being equal to) the qualification loads is as small as 1×10^{-9}. In either case, the qualification loads are dynamically applied by simulating landing impact so that the proper shock loading, as well as maximum load levels, is achieved.

For qualification test of the complete lander vehicle, drop tests are performed to duplicate the desired qualification landing conditions. These conditions are established analytically by either the statistical or absolute-performance method previously mentioned. Several drop conditions may be established to produce maximum loads in the attenuation system and lander structure and conditions of minimum surface clearance, maximum attenuator deflection, and maximum instability of the lander vehicle.

All or only some of these types of tests are performed, depending on the particular attenuation system and lander design being qualified.

2.3.3 Acceptance Testing

Lander-acceptance tests are performed on components and assemblies of actual mission hardware to ensure that engineering requirements and manufacturing specifications have been met. Acceptance tests provide a proof of workmanship and establish confidence in the flightworthiness of the attenuation-system hardware. Such tests are designed so that imposed load levels and environments do not compromise the structural adequacy or operability of the hardware in any way.

If an attenuation system must be deployed before landing impact, then a functional check of deployment is part of the acceptance test. If an energy absorber, such as a hydraulic shock absorber, can be repeatedly stroked without damage, then, as part of the acceptance test, the absorber may be subjected to a load-stroke profile simulating the profile expected for a nominal landing condition. Acceptance tests may also include static loading of an attenuation system to load levels corresponding to nominal landing-load levels, providing the functional capability of the system is not impaired. (For example, a crushable energy absorber obviously must not be deflected during attenuation-system acceptance tests.)

In acceptance tests, the attenuation system may be subjected to all the types of environments to be encountered during the mission. The environmental test levels, however, are usually nominal for acceptance tests.

Depending on the attenuation system and lander design, acceptance testing may be performed at subsystem and/or system levels and may include all or only some of the mission environmental conditions that will be encountered.

3. CRITERIA

The design of a space vehicle's landing-impact-attenuation system shall adequately account for all landing-induced loads and motions imposed on the lander from the time of touchdown until the lander comes to rest. The attenuation system shall satisfy all design requirements imposed by the mission. The performance of the attenuation system and the dynamics of the lander shall be predicted and verified by a suitable combination of analysis and tests.

3.1 Design Requirements

The design of the landing-impact-attenuation system shall satisfy all requirements imposed by the following:

- Mission environments.
- Payload loading and operation.
- Other systems of the vehicle.
- Materials and fabrication techniques.

3.2 Analysis

The analysis of attenuation-system design and landing dynamics shall include all pertinent combinations of the following at touchdown:

- Lander properties and conditions.
- Local environmental conditions.
- Characteristics of the attenuation system.

The analysis shall account for the cumulative effects of the total loadings (ground handling, boost, previous landings, etc.) to be sustained by the lander structure and payload.

The analytical models shall contain sufficient degrees of freedom to represent appropriately rigid-body motions, attenuator deflections, and the significant structural flexibility. Surface models shall enable determination of realistic force levels generated by lander/surface interaction. Computational accuracy shall be established.

3.3 Tests

Tests shall be performed, as necessary, to establish parameters for analysis, to verify structural integrity, to substantiate analytical predictions, and to obtain otherwise unavailable material properties. Test models of hardware and vehicles shall be so designed that the touchdown of the lander on the surface can be realistically simulated.

4. RECOMMENDED PRACTICES

The recommended general procedure for spacecraft-attenuation-system design and touchdown-dynamics analysis is to perform the following functions:

1. Select a number of candidate design concepts on the basis of known design and mission constraints.

2. Make preliminary analyses and tradeoff studies in sufficient detail to reduce the number of candidate designs to two or three.

3. Perform detailed analyses of these designs and, where necessary, perform tests to determine the final design choice.

4. Analyze the final design to predict landing performance over the complete range of anticipated touchdown conditions.

5. Fabricate hardware and vehicles for tests, and conduct the tests to substantiate analytical predictions and to ensure acceptable landing performance.

At all stages of design, the methods and results of tradeoff studies, analyses, and tests should be documented to enable assessment of the reasons for choice of preferred concepts and of the validity of the analyses and tests.

4.1 Design Requirements

4.1.1 Mission-Environment Requirements

A number of constraints or requirements which should be considered in the design of the attenuation system are imposed by the mission environments prior to and following touchdown. Practices for modeling the landing-surface environment are recommended in Section 4.2. Degrading effects of the vacuum environment should be minimized by the use of space-approved materials (ref. 21). Thermal environments encountered, including sterilization temperatures, should be carefully evaluated to determine their effects on material strength. Adverse effects from thermal environments should be prevented by application of appropriate thermal control or by alternative choices of materials. If necessary, tests should be conducted to establish that prescribed sterilization chemicals will not adversely affect performance. In assessing the lander's structural integrity at touchdown, analyses should consider the fatigue effects of all load cycles imposed on the structure before touchdown (refs. 18 and 19).

4.1.2 Payload Loading and Operation

Attenuation-system design and touchdown-dynamics analysis should establish that the loads transmitted to the payload do not exceed allowable values in magnitude, direction, duration, and onset rates. For inanimate payloads, allowables will vary widely, depending on mission requirements. For living payloads (specifically, men), loads should not exceed the allowables given in reference 27. When appropriate, payloads should be isolated from the lander structure by an additional energy-absorption and shock-attenuation system to reduce the requirements on the attenuation system of the lander.

Unless other mechanisms are specifically provided to facilitate payload positioning, attenuation systems should be designed to ensure that payload deployment, surface proximity, orientation, and postlanding stability after touchdown are compatible with operational requirements. Omnilanders are not recommended for manned missions. Manned water landers should be designed to right passively to a "hatches-up" position and float indefinitely in this attitude. When the landing configuration is not compatible with this recommendation, an active system (such as a gasbag-inflation system) should be used to stabilize the lander in a hatches-up position.

4.1.3 Interfaces With Other Systems of the Vehicle

Since the volume of the attenuation system must be kept within the allowable stowage volumes of launch-vehicle shrouds or entry-vehicle aeroshells, it may be necessary that the attenuation system be collapsible or foldable. Folding mechanisms, such as leg assemblies, are highly reliable. However, collapsible devices (e.g., a gasbag system which would require deployment after a six-month journey to Mars) are not recommended unless the necessary reliability can be established by test.

When rocket thrust is used to reduce velocity prior to touchdown, the attenuation system should not protrude into the rocket-exhaust plume. If the rocket is designed to fire close to the landing surface, surface erosion and the possibility of lander contamination should be assessed with the analytical methods indicated in references 22 to 24, and by the test results of references 25 and 26. Where applicable, landing-dynamics analysis should include the base pressures and thermal loads on the lander resulting from rocket-plume reflection from the landing surface (ref. 29). When problems of system compatibility and interaction arise, tradeoff studies should be performed to minimize the adverse effects on mission objectives.

4.1.4 Materials and Fabrication Techniques

For maximum reliability, established materials and manufacturing procedures should be used. Where improved attenuation-system performance is needed, advanced materials, procedures, or processes should be developed and qualified at an early stage in design to ensure manufacturability and reliability. For all designs, an adequate functional life (including shelf life before use) should be established for each attenuation-system component by comparison with established designs or by exhaustive tests. Actual nonideal material properties in the anticipated environments should be used in analysis. If these properties are unknown, they should be established by tests.

4.2 Analysis

4.2.1 Input Parameters

Inputs to analysis of touchdown dynamics should include, but not be limited to, all pertinent combinations of the following:

1. Lander properties and conditions.

   A. Vertical and horizontal velocity.

   B. Attitude and attitude rates.

   C. Mass properties.

   D. Structural flexibility.

   E. Geometry.

   F. Descent-engine thrust characteristics, including angular and linear misalignment and tailoff.

   G. Base pressures resulting from surface reflection of descent-engine-exhaust plume.

   H. Liquid-sloshing characteristics (ref. 2).

   I. Nozzle choking and corresponding thrust amplification and shutdown-transient characteristics.

J. Engine-plume thermal environment and its modifications when near landing surface.

2. Local environmental conditions at touchdown.

   A. Gravitational force.

   B. Solid-surface properties.

      (1) Surface slopes, slope distribution, and length.

      (2) Size and distribution of protuberances and craters.

      (3) Mechanical properties, including friction, density, compressibility, and bearing-strength variation with depth.

   C. Liquid-surface properties.

      (1) Density, viscosity, compressibility, temperature.

      (2) Wave form, height, velocity, shape.

   D. Wind velocity and pressure.

   E. Erosion and permeation effects of engine-exhaust gases on landing-surface properties. (See the analytical methods of refs. 22 to 24 and test results of refs. 25 and 26.)

3. Attenuator characteristics.

   A. Force-stroke characteristics as a function of direction of applied force.

   B. Energy-absorption characteristics.

   C. Strain-energy storage capability.

   D. Properties related to stress-wave propagation (refs. 36, 37, 63, and 64).

   E. Velocity sensitivity.

   F. Ability to withstand impact.

G. Mechanical backlash.

H. Thermal sensitivity.

I. Mechanical friction.

4.2.2 Analytical Methods

The absolute-performance method, the statistical method, or a combination of both methods should be used in the analyses (Sec. 2.3).

For absolute-performance analysis, input parameters should be combined systematically to establish the worst combination for the performance characteristic under investigation. Where absolute values are not specified for a parameter, then the 3-σ limits of a statistical distribution of values should be used. It should be understood that the "worst" combination of values does not necessarily consist of minimum or maximum values for each parameter. Absolute-performance analysis should be used throughout design and development to establish performance capability and evaluate the effects on performance of changes in design and touchdown conditions.

For statistical analysis, probability-density functions should be defined for each input parameter. Where there is interdependency between parameters, appropriate multivariate functions should be defined. Monte Carlo techniques (refs. 49 to 51) should then be used to select sets of input conditions for landing simulations. The accuracy of the probability-density functions is critical to the results achieved with this method; therefore, all known details of spacecraft-operational characteristics and environmental quantities should be carefully considered in determining such functions. If information is not available to use as a basis for probability distributions (e.g., no data presently exist on the mean value of surface slopes on Pluto), then distributions (hopefully conservative) should be assumed. The sensitivity of the analytical results to the assumed density functions should then be assessed.

Because accurate probability-density functions for most landing parameters usually cannot be generated until a lander design has been developed to a condition where design details and parameters are well defined and understood, detailed statistical analysis should be used only when the design is finalized. During design development, however, conditional probabilities of landing success should be obtained by assigning specific values to some landing parameters (e.g., touchdown velocity and incidence). Also, statistical analysis should be used to determine the acceptability of out-of-specification landers.

Until more complex computer programs can be generated, the initial analyses for sizing attenuation systems should be relatively simple, such as those presented in references 31, 33, 43, and 65. Where needed in later designs, complex computer programs should be generated that will enable a three-dimensional landing simulation by including six degrees of freedom for the rigid-body lander. The simplest solution that is adequate for the analysis should always be used. For example, a three-dimensional solution should not be used if performance can be adequately assessed by simulating two-dimensional or even one-dimensional landings.

4.2.3 Mathematical Models

4.2.3.1 Lander Models

When it can be demonstrated that stability, surface clearance, surface penetration, and similar performance characteristics are not significantly affected by structural flexibility, then the spacecraft should be modeled as a rigid body attached to the attenuation system. When modeling the spacecraft to compute structural and payload-response loads, the lander's structural flexibility should be represented (refs. 1 and 44).

4.2.3.2 Surface Models

When solid surfaces are modeled, rigid surfaces and soft surfaces should be treated separately. For rigid surfaces, coefficients that characterize the friction between the lander and the surface, if not specified, should be appropriate to the design and materials used (e.g., friction data of ref. 66) and verified by tests when practical. The static friction forces on any part of the attenuation system that is in stationary contact with the surface should be determined by calculations that establish conditions for achieving zero tangential acceleration (i.e., maintaining zero velocity) at that point. However, when it can be shown that the transition from kinetic to static friction conditions does not adversely influence results, this transition can be modeled simply, as indicated in reference 45.

For soft surfaces, the simple models described in references 52, 53, and 55 are recommended. A range of static bearing pressures consistent with the area of surface contact should be assumed in forming an analytical expression for surface force. This expression is also a function of the geometry of the penetrating portion of the lander. A dependent velocity-squared term based on soil-inertia properties should be included in the expression, when such a term significantly affects the results. Complex expressions for soft-surface forces, based on such quantities as cohesion, porosity, particle size, and internal friction, are not recommended for design purposes unless precise knowledge of these quantities and other landing-surface features exists.

Liquid-surface models should incorporate values for surface density and wave form, height, velocity, and shape, as defined in figure 14. In addition, values for viscosity, compressibility, and temperature effects should be incorporated unless they can be shown to be insignificant. (The values assigned to these parameters are determined by mission constraints.)

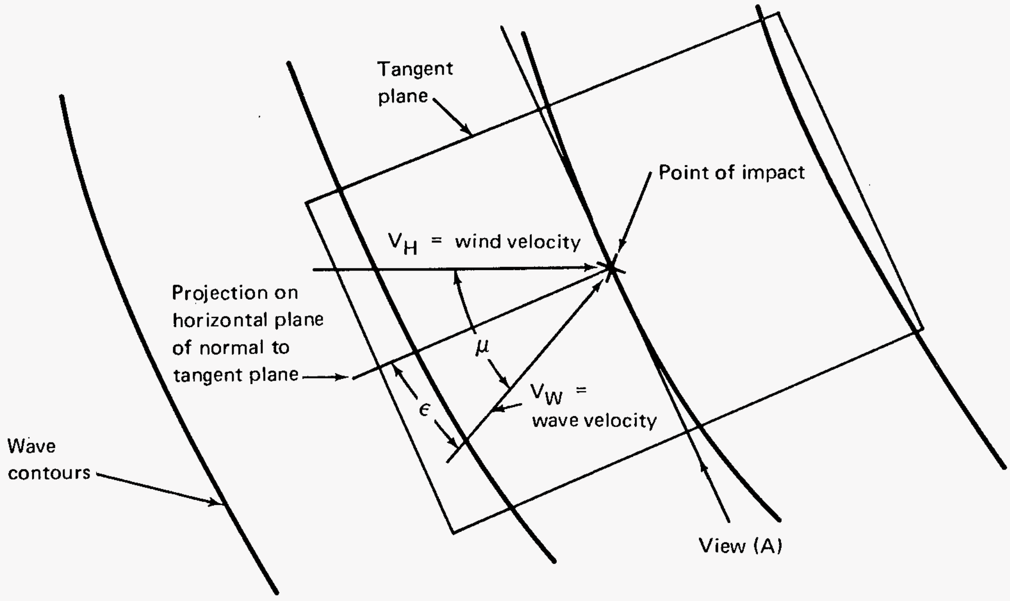
4.2.3.3 Lander-Surface Model Synthesis

For solid-surface landers, both rigid- and soft-surface models should be used in analysis. Analysis of solid-surface touchdowns should consider downhill sliding and rolling and the associated hazards of abutment impact. The kinetic energy that might be gained by downhill rolling should be limited by appropriate geometric design and mass distribution.

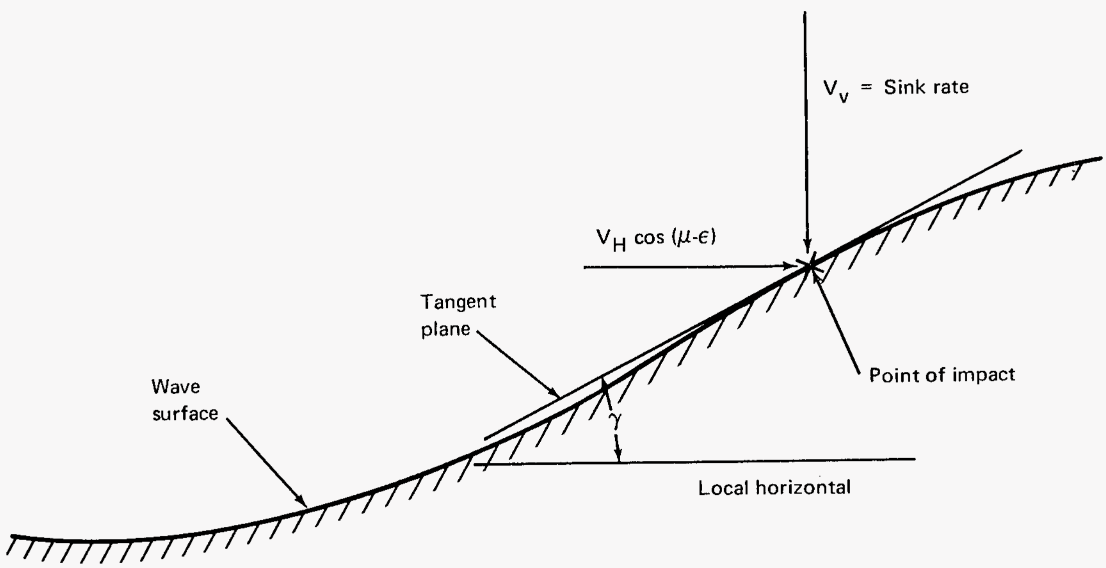
On rigid surfaces, minimum friction values should be used in determining downhill-sliding susceptibility and maximum values should be used in stability calculations. Combinations of friction values should be used for assessing the stability of a lander striking against an abutment, when a significant probability of such an impact exists. For calculating stability on high slopes (without abutment impact), the choice of friction coefficients should be influenced by considerations of downhill sliding and rolling. The calculation of stability boundaries using friction coefficients equal to or less than the limiting friction value is not meaningful. For landers in which crushable material contacts the surface, the shear strength of the crushable material may limit the friction forces that can be applied to the lander.

Surface protuberances and craters should not be considered in analysis as incremental changes to local slopes, but as discontinuities that can decrease lander clearance for a legged lander, or cause local loads on omnilanders, or act as surface abutments to arrest surface tangential motion. Attenuator deflection should not permit the main lander structure to impact protuberances or surface discontinuities unless it can be shown that such impacts will not degrade performance.

Stability boundaries should be established from predictions of actual vehicle toppling. However, since the time required to evaluate stability or instability may be exorbitant for conditions close to the stability boundary, adequate stability should be measured by a specific tilt angle achieved in a specific time interval, or alternatively by an angular-rate and tilt-angle criterion. The criterion used should be established from detailed landing simulations and should always be conservative.



(a) Top view



(b) View (A)

Figure 14. – Significant ocean-environment properties for definition of water-impact conditions.

Design and analysis of all landers should account for the nonideal characteristics of attenuators and for variations in their characteristics due to manufacturing tolerances or due to degradation from environmental conditions. For example, if a constant-force-versus-stroke attenuator is required, the mean force for each attenuator should not vary from a specified desired mean beyond a specified range. Similarly, maximum and minimum forces should remain within specified limits. Thus, when energy absorption is critical, the lowest allowable mean value should be used in analysis; when load capacity is critical, maximum force levels should be used. The effects of these variables on performance should be considered and incorporated in the analysis from the outset of design. In addition, in design analyses, maximum or minimum force levels, as appropriate, should be determined for an expected range of attenuator temperatures.

When mathematical models and analytical procedures have been established to predict touchdown performance, analytical accuracy should be defined by one or both of the following methods:

- Adjust the mathematical model, as necessary, to represent an operational spacecraft for which test results already exist (e.g., refs. 42, 43, 55, and 57). Analytical results should then be compared with these test results.

- Adjust the mathematical model to conform with an existing model whose accuracy has already been proven (e.g., ref. 46). Analytical results from the two models should then be compared to assess the accuracy of the new model.

Computational accuracy should be assessed by either of the following steps:

- Perform an error analysis to determine the sensitivity of results to changes in computational procedures, integration method, or integration interval. The procedures should be adjusted, as required, to achieve an acceptably low sensitivity.

- Incorporate into the computational procedures an error check (such as in ref. 45) which controls cumulative errors within acceptable limits.

4.3 Testing

4.3.1 Design-Development Testing

Functional and mechanical properties of components and subassemblies should be determined by tests and used as inputs to the touchdown-dynamics analysis (e.g., force-stroke characteristics of energy absorbers).

Tests should achieve the highest degree of simulation that is practical and should consider the effects of all anticipated flight conditions. Any deliberate deviation of the test hardware and test conditions from the known mission hardware or anticipated environmental conditions should be carefully assessed and justified. References 67 and 68 describe typical component tests recommended for use in the design and development of an LM-type landing-gear assembly.

If results of tests are required before mission hardware is available, or if certain testing of flight hardware is impracticable, then scale-model tests should be performed. Scale models should be fabricated with due regard to mass properties, geometry, attenuator characteristics, structural characteristics, gravitational force, velocity, and accelerations (refs. 28 and 56). If it is impractical to achieve dynamic similarity for planetary or lunar landers, gravity-force simulation should be considered (refs. 28, 61, 67, and 69). If impact loads considerably exceed the local static gravitational loads, gravity simulation should not be necessary to simulate the maximum load conditions. If the high-impact loads are affected by the lander's touchdown attitude and angular motions, then equivalent earth-impact conditions should be specified to achieve the desired load levels (ref. 44). Where a test is intended to check the accuracy of the analysis and not to establish landing-performance limits, then dynamic-scaling requirements should be relaxed (as in ref. 57).

Full-scale test vehicles should incorporate attenuator and/or stabilizer systems consistent with flight systems. For verification of landing-performance characteristics such as stability, surface penetration, or surface clearance, the test vehicle should have a rigid spaceframe and payload when performance has been shown to be independent of flexibility (refs. 28 and 69). For determination of structural loads, a spaceframe and payload model that exhibit the mass distribution and flexibilities of the mission spacecraft (refs. 1 and 44) should be used.

4.3.2 Qualification Testing

Qualification tests should be conducted to verify the lander's structural integrity. They should subject the hardware and full-scale lander, which are fabricated identically to flight articles, to loading and environmental extremes greater than those anticipated in the service life of the spacecraft. These extremes should include load-application rates, levels, and durations and temperatures that can occur between the time the hardware is fabricated and the time the flight spacecraft has landed and come to rest.

4.3.3 Acceptance Testing

Acceptance tests should be performed to ensure that components and assemblies of mission hardware meet material and manufacturing specifications without loading the hardware to the point where its later operability might be compromised. To ensure that the properties of production lander attenuators conform to design requirements, acceptance tests should be performed on each attenuator whenever possible. If this is impractical, then tests should be performed on random samples of attenuator production components. The selection of a random sample is discussed in reference 70.

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SP-8002	(Structures)	Flight-Loads Measurements During Launch and Exit, December 1964
SP-8003	(Structures)	Flutter, Buzz, and Divergence, July 1964
SP-8004	(Structures)	Panel Flutter, May 1965
SP-8005	(Environment)	Solar Electromagnetic Radiation, June 1965
SP-8006	(Structures)	Local Steady Aerodynamic Loads During Launch and Exit, May 1965
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