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# GEMINI LAND LANDING SYSTEM DEVELOPMENT PROGRAM

## VOLUME I - FULL-SCALE INVESTIGATIONS

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# ABSTRACT

This report presents the development of a working system which would allow a returning spacecraft to make a soft landing on land or on water. This landing system combines a gliding parachute for maneuver and wind-drift negation capability with retrograde rockets fired just above the surface to reduce the descent velocity prior to impact. The report documents the entire 3-year effort which began with scaled-model tests and concluded with a completely successful full-scale land landing demonstration at Gemini spacecraft design conditions.

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# GEMINI LAND LANDING SYSTEM

# DEVELOPMENT PROGRAM

## VOLUME I - FULL-SCALE INVESTIGATIONS

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## SUMMARY

This report documents the accomplishment of an historic milestone in spacecraft landing technology by presenting the development of a working system which would allow returning spacecraft to land softly on land, rather than be restricted to landing in the ocean. The basic program objective was to develop and to demonstrate a system which could be applied directly to the Gemini spacecraft. This objective has been fully met. The program also provided the opportunity and the focal point for the formation of a sound technological base from which land landing systems for future manned spacecraft may be developed.

The Gemini system provides the ability to land the spacecraft on a variety of unprepared farm and ranch lands, with all accelerations controlled to a level of approximately 5g. The system also provides an improved water landing capability should the necessity arise. In both types of landing, the pilot must fly the spacecraft to the desired touchdown point. Near the surface, he is required only to aline with the wind axis, and is not faced with the delicate flare maneuver necessary with other proposed land landing methods.

This landing system has six integrated components, the functions of which have been verified individually and jointly. A controllable, gliding parachute descent system was developed to furnish the necessary range, obstacle avoidance, and wind-drift negation capability without compromising the high deployment reliability of standard parachutes. Retrograde rockets were designed and developed to be fired just above the landing surface to reduce descent velocity prior to impact. The mechanical landing gear, originally designed for the Gemini paraglider, was incorporated without change for attenuation of residual vertical velocity and for post-impact stability. Turn

control motors were developed for steering the parachute; two altitude sensing devices were developed for ignition of the rockets at the precise height; and the pilot display and the visual reference requirements were determined.

This 3-year program began with a realistic evaluation of the conditions under which a land landing system must operate, and the desire to develop the simplest, the safest, and the most reliable landing method possible. System performance and spacecraft interface requirements were analytically determined, then incorporated as preliminary specifications for development of the components. Two scaled-model test programs were conducted to investigate flight and landing characteristics of the integrated system. These studies proved to be remarkably accurate in predicting full-scale performance.

Following the scaled-model tests, a program of 14 full-scale tests was conducted, utilizing a Gemini boilerplate vehicle which duplicated the flight spacecraft insofar as the landing system was concerned. These full-scale flight tests integrated the developed components into a working system, and culminated in a completely successful demonstration of a land landing at Gemini spacecraft design conditions.

In this demonstration, the boilerplate vehicle, ballasted to the correct weight and center-of-gravity location, executed a nominal Gemini parachute deployment and attitude repositioning sequence. The vehicle was maneuvered by radio command approximately 2 miles crosscountry, alined with the wind, and landed 60 yards from the center aiming point in the primary target area. The maximum accelerations experienced were below 5g. This test demonstrated the land landing system in realistic operation, and successfully concluded the development program.

In support of the basic objective, definitive specifications for flight hardware have been prepared. All components were designed for operation in a spacecraft environment, and no known problems exist which would prevent their flight qualification. To complete the Gemini effort, operational mission and recovery plans were prepared for employing the land landing system in the Gemini Program.

This document presents the entire 3-year development and demonstration effort in technical detail, and does not require additional references. It contains a complete description of the system and all test results. This effort was conducted primarily by Manned Spacecraft Center (MSC) personnel, and amply demonstrates the advanced technical capability from which MSC can look with confidence toward the development of new systems.

# INTRODUCTION

At this point in the manned exploration of space, all past and currently planned earth landing systems require that the returning spacecraft land in water. Landing in the ocean allows for a wide reentry dispersion and a considerable saving in weight, since the water relieves the necessity for an onboard attenuation system. Unfortunately, landing in the ocean also dictates a large Naval recovery force, and salt-water corrosion seriously inhibits reuse of the spacecraft. Due to the increasing intensity of manned missions, which makes spacecraft reuse an important factor, and because increased reentry guidance capability now makes point landings feasible, the development of land landing capability is a prime objective of the Manned Spacecraft Center.

During the past 3-1/2 years, the Manned Spacecraft Center has devoted considerable effort toward the investigation of systems and techniques for landing spacecraft on land as a planned means of mission termination. As a part of the investigation, a program was conducted to investigate the feasibility of employing a gliding parachute/retrorocket system. This program, conducted entirely as a Manned Spacecraft Center development effort, utilized a Mercury boilerplate test vehicle, with a glidesail descent system and a manifolded rocket system thrusting through the center of the heat shield for impact attenuation. The very promising results of these tests demonstrated the feasibility of this type of system and provided the basis for the development program described in this report.

In September 1962, a land landing system development program was initiated at the Manned Spacecraft Center with the following objectives: to compile a complete data description of system performance, to prepare detailed specifications for flight hardware, and to determine the operational procedures by which the land landing system could be employed in the Gemini Program.

Detailed descriptions of the topics discussed in Volume I are found in Volume II. These are: Section I, One-Third-Scale Flight Dynamics Tests of the Gemini Land Landing System; Section II, Landing Dynamics of a One-Third-Scale Para-Sail/Landing Rocket Model; Section III, Parachute Development; Section IV, Rocket-Motor Development; Section V, Altitude-Sensor Development; Section VI, Turn-Control Motor Development; Section VII, Investigation of the Visual Reference Requirements for Pilot Control of Gliding Parachutes for Land Landing of Spacecraft; Section VIII, Gemini Flight-Test Vehicle for the Para-Sail/Landing Rocket Program; Section IX, Landing Gear and Test Hardware Verification; Section X, Operational Performance Study; Section XI, Instrumentation and Electronic Systems of the Gemini

Para-Sail/Retrorocket System; and Section XII, Sequencing and Ignition Systems for the Gemini Land Landing Program.

The authors wish to emphasize that the successful system development effort described represents the combined efforts of over 300 Manned Spacecraft Center personnel. Management of the total program was exercised by the Landing Technology Branch, Structures and Mechanics Division. The following organizations participated in and contributed to the development of the land landing system.

Structures and Mechanics Division
Landing and Recovery Division
Technical Services Division
Propulsion and Power Division
Instrumentation and Electronic Systems Division
Photographic Technology Laboratory
Computation and Analysis Division
Crew Systems Division
Gemini Program Office
Procurement and Contracts Division
Pioneer Parachute Company, Inc.
Sylvania Electric Products, Inc., Electronic Systems Division
DeHavilland Aircraft of Canada, Ltd.
Aircraft Armaments, Inc.
Thiokol Chemical Corp., Elkton Division
McDonnell Aircraft Corp.

In the preparation of this document, the writers wish to express appreciation to John W. Kiker, James K. Hinson, and Harold E. Benson, who exercised supervisory responsibility during the development effort, and served as approving editors.

## SYMBOLS

$C_D$ drag coefficient

$d_o$ nominal diameter based on total canopy area, ft

F opening force, lb

KIAS indicated airspeed, n. mi./hr

| | |
|---|---|
| k | constant |
| L/D | lift-to-drag ratio |
| q | dynamic pressure, $1/2\ \rho V^2$, $lb/ft^2$ |
| R/D | rate of descent, ft/sec |
| R/T | rate of turn, deg/sec |
| S | area, $ft^2$ |
| t | time, sec |
| V | velocity, ft/sec |
| W | weight, lb |
| $\rho$ | density |

## GENERAL DESIGN PHILOSOPHY

The design of this system began with a realistic appraisal of the conditions under which a land landing system must operate, and with a desire to develop the most trustworthy and the most reliable system possible. A system designed to land spacecraft on land must first provide a means of getting to an acceptable touchdown area; and then provide impact attenuation which must insure that the resulting accelerations are kept within acceptable limits.

Many investigations of new concepts have been conducted in the high lift-to-drag ratio (L/D), or aircraft-like field, as well as various high-wing-loading, low L/D lifting bodies. In common with modern day aircraft, the operation of each of these devices is based upon a final flare maneuver wherein descent velocity is translated into horizontal velocity. The high horizontal speeds caused by the flare maneuver require a prepared landing surface and a high degree of pilot skill to achieve a successful landing.

Since the attenuation system must account for this horizontal velocity as well as vertical velocity, any forward speed in excess of that required to cancel wind drift creates a problem rather than offers a solution. A preliminary survey of available landing sites within the United States indicated that an L/D of unity would be sufficient to cancel wind-induced drift.

The landing system, as conceived for this development program and demonstrated in the feasibility study, employed a gliding parachute for limited range capability, obstacle avoidance, and negation of wind drift, with retrograde rockets fired just above the ground to reduce descent velocity at impact. This system would be operationally employed by reentering the spacecraft over a selected land zone, and then allowing the flight crew to determine and to maneuver to the touchdown point within the zone through visual observation of wind drift and obstacle coverage. Since the forward velocities encountered are relatively low, this type of system is not restricted to landing on a prepared runway, but can land on a variety of selected farm and ranch lands. Of equal import, the pilot would not be faced with a delicate flare maneuver, but would only be required to maintain the proper heading at touchdown.

## Gliding Parachute

For the descent system, the philosophy followed was to extend existing technology by developing a high-performance parachute which still retained the light weight, low volume, and high reliability of standard parachute systems. A gliding parachute has a forward velocity as well as a rate of sink; hence, it descends at an angle to the vertical. When directional control means are incorporated, the forward velocity allows limited range capability and maneuvering to avoid local obstacles. This forward velocity also allows the cancellation of horizontal drift due to surface winds. This capability is important since it permits control of the direction of impact and presents the attenuation system with a limited set of energy conditions.

## Impact Attenuation

The choice of solid-propellant rocket motors for attenuation of descent velocity was based upon low weight and volume requirements and proven performance in other spacecraft applications. The technology existed; only the concept of final braking, just prior to touchdown, had not been thoroughly developed. In addition to the rockets, a set of landing gear was included to provide mechanical attenuation of residual vertical velocity, slideout dissipation of any remaining horizontal velocity, and post-impact stability.

### System Implementation Hardware

In addition to the parachute, rockets, and landing gear, an altitude sensing device was required for precise control of rocket-ignition altitude. Preliminary studies indicated that existing methods did not furnish the required degree of accuracy. A control package was required to maneuver the system; and finally, the pilot display and visual reference system requirements would have to be determined to allow the flight crew to utilize fully the maneuver capability.

## LAND LANDING SYSTEM

The first order of investigation was the analytical definition of design requirements and of performance parameters, and the translation of these into development specifications for the system components. Next, the components were developed to meet these specifications. At the same time, scaled-model tests were conducted to determine preliminary system flight and landing dynamics. Finally, the developed components were integrated into full-scale system tests. At the end of the development phase, a demonstration of the complete land landing system at Gemini spacecraft design conditions was conducted.

Concurrent with the hardware development effort, an operational evaluation of system capability was conducted to determine exact landing zone criteria, flight operational procedures, and support equipment requirements.

Two boilerplate vehicles were designed and fabricated especially for the land landing system tests. These vehicles duplicated the size, shape, structural hardpoints, and stowage volumes of the flight spacecraft; and were ballasted to the correct weight and center-of-gravity location. Figure 1 illustrates a spacecraft station and nomenclature diagram and indicates the reference axes. Figures 2 and 3 depict the salient test vehicle features and landing system stowage locations. A detailed description of the test vehicles and associated hardware is contained in Volume II, Section VIII.

Figure 4 shows the land landing system in flight, traveling from right to left. The rear of the canopy is comprised of shaped slots which allow the entrapped air to exit down and aft. This exiting flow forces the canopy to its lifting angle of attack. The front of the canopy is composed of smooth, solid panels and assumes a cambered pressure shape, forming a crude airfoil which achieves a one-to-one glide ratio. The row of panels below the skirt on each side furnishes directional stability and creates an elliptical, inflated planform. The four vertical rows of exhaust panels on each side are the turn slots, and turns are accomplished by opening and closing the slots with a miniature cable and winch system which is located in the test vehicle. During tests, these winches, or turn motors, are activated by radio command.

The landing-gear system (fig. 4) consists of two main gears and one nose gear in a tricycle arrangement. The two main gears are cantilever struts connected to a hydraulic damper and are capable of spring-like deflection. The nose gear is composed of a telescoping hydraulic shock-absorber strut. All three gears have flat skid-like shoes to allow slideout. These gears are stowed until attitude repositioning; then they are pyrotechnically deployed.

The twin booms extending beneath the test vehicle are the deHavilland altitude sensors. These devices consist of self-extending metal probes with microswitch heads located at the tips. The sensors extend downward to the desired rocket-ignition height and trigger the rocket motors when the microswitch heads contact the surface. These sensors, either of which is capable of firing both rockets, are stored during the early portion of the flight and are deployed pyrotechnically after attitude repositioning.

The two rocket motors are located on the underside of the vehicle between the main landing gears. Figure 5 shows the bottom of the vehicle and the rockets mounted in the lower equipment bay. The nozzles are angled to allow the thrust line to pass through the center of gravity of the vehicle.

## SUPPORTING ANALYSIS AND TESTING

Many supporting investigations, scaled-model tests, and component development programs were conducted to define and to develop the system prior to testing full scale. To provide an integrated picture of the many preliminary facets, this brief description of the results of each of these efforts indicates how the facets phased into the overall program. Tables I to XII contain a presentation of all the tests conducted during the program. The more important of these studies and tests are presented in detail in Volume II.

### Analytical Design Requirement Studies

Operational and analog flight-simulation studies. - These studies were conducted to define detailed performance parameters for the parachute, to furnish a preliminary description of the landing zone, and to determine operational system performance minimums. The results may be summarized as follows:

1. The parachute must be capable of negating a 30 ft/sec surface wind, and the landing gear must be capable of accepting horizontal velocities of from 0 to 30 ft/sec on an unprepared sod surface.

2. A rate of turn (R/T) of at least 10 deg/sec is required to allow obstacle avoidance and wind alinement.

3. The system is compatible with zone landing in an area described as a 17- by 8-nautical-mile ellipse of selected, but unprepared, terrain. This area must be at least 60 percent free of obstacles and should not contain localized ground slopes of more than 15°. It should be easily accessible by ground vehicles to facilitate landing support and recovery operations. A ground advisory station located in the zone would increase system capability during periods of low visibility.

Parametric optimization of rocket performance. - The rocket-motor performance requirements were determined in an optimization study which considered the coupled effects of performance and environmental variations. Motor envelope and interface requirements were determined by the selection of the lower equipment bay for installation.

This study indicated that the rocket motors should employ a dual thrust level. The first is a high-level thrust (resulting in approximately 2.65g) acting for a short time interval to lower the descent velocity to a nominal value of 3 to 10 ft/sec. The second level is a sustained thrust (resulting in approximately 0.5g) designed to maintain the descent rate near the minimum value during the period in which it is acting on the spacecraft. The sustained low thrust level effectively increases the time during which the descent rate is less than the design landing velocity of 10 ft/sec. A detailed description of the rocket performance analysis is found in Volume II, Section IV.

Gemini spacecraft design integration. - These studies were conducted to insure that the design requirements and the performance parameters were compatible with Gemini spacecraft weight, structural arrangement and limitations, reentry dynamics, and landing system envelope. The initial study was augmented in the summer of 1963 when McDonnell Aircraft Corporation, at the direction of the Gemini Program Manager, conducted a study of the integration of the land landing system into the Gemini flight program. These studies established the following parameters:

1. Utilization of the existing Gemini spacecraft drogue parachute and the rendezvous and recovery system (R and R) canister.

2. Deployment of the Para-Sail, at nominal Gemini reentry conditions, from the separated R and R canister with the spacecraft in the heat-shield-down attitude.

3. Utilization of the existing paraglider pallet disconnect assembly attach points on the nose gear for the Para-Sail riser attach points while the spacecraft is in the heat-shield-down attitude.

4. Change of vehicle attitude to a 13° nose-down flying attitude after Para-Sail opening.

5. Limitation to existing paraglider structural hardpoints for suspension system attach points in the flying attitude.

6. Utilization of the existing Gemini landing gear, gear stowage provisions, and deployment mechanisms. Modification to the gear was permissible insofar as it did not require modification to the spacecraft.

7. Limitation to the existing paraglider control-system envelope within the hatch beam for location of the Para-Sail control actuators.

8. Limitation to the existing paraglider stowage volume in the R and R canister for Para-Sail stowage.

9. Four possible altitude-sensor locations, providing the required stowage volume, could be employed. These are: the landing-gear wells, the lower equipment bay, the area aft of the lower equipment bay, and the conical section forward of the pressure vessel. (The area aft of the lower equipment bay was used during the development program.)

10. Utilization of the existing lower equipment bay for location of the landing rockets.

## Component Development Programs

Para-Sail gliding parachute. - The Para-Sail program was designed to develop the parachute to meet the performance parameters obtained from the analytical studies. Three separate development phases were conducted. The first consisted of low-speed deployment and steady-state evaluation tests conducted with 24-foot-diameter models. The second began with an 80-foot diameter configuration designed by the inventor of the Para-Sail. Four major configuration changes were made during the test program. The third consisted of 14 verification tests of the final configuration, resized to 69.8 feet in diameter, to provide the correct rate of descent. This test program was conducted at El Centro, California. These efforts developed a parachute demonstrating the following performance parameters:

| | |
|---|---|
| Suspended weight, lb | 4 750 |
| Deployment altitude, ft | 10 600 |
| Deployment dynamic pressure (design), lb/ft$^2$ | 80 |
| Rate of descent at 5000-ft pressure altitude, ft/sec | 29 |
| Lift-to-drag ratio | 1.0+ |
| Rate of turn (control-line force = 100 lb), deg/sec | 25 |
| Stability, steady-state (maximum oscillation), deg | ±3 |
| Ultimate strength capability (1.5 design), lb/ft$^2$ | 120 |
| Maximum opening force (design q), lb | 16 000 |

The final configuration meets all the land landing system requirements, and can be qualified for space flight. The total weight of the parachute, deployment bag, pilot parachute, risers, and so forth, is 153 pounds. A detailed description of this development effort is contained in Volume II, Section III.

Landing rockets. - The landing rockets were developed for the Manned Spacecraft Center by Thiokol Chemical Corporation. These rocket motors met the lower equipment bay envelope and interface requirements and furnished the thrust-time characteristics defined in the analytical study. The propellant, designated TP-H-1050, was developed and qualified for the Dyna-Soar acceleration motors; and the Holex 3575 initiator was previously qualified for the Gemini spacecraft retrorockets.

Before the rocket motors were incorporated into the system test program, 12 firings were conducted to verify thrust-time characteristics after the motors were subjected to temperature and vibration environments. These motors met all system requirements and could be qualified for space flight.

The two rocket motors and mounts have a total weight of 60 pounds. (See Volume II, Section IV for a detailed description of the development and verification effort.)

Turn-control motors. - Directional control of the Para-Sail is accomplished by retracting and extending cables that close and open the turn slots on each side of the canopy. At program initiation, the force, the travel, and the takeup time requirements were unknown. To determine these parameters, a set of control motors, based upon analytically determined requirements, was developed, under contract, by Aircraft Armaments, Inc., and included in initial system testing. As the system test program progressed, it was necessary to modify these motors extensively to obtain adequate performance

(Volume II, Section VI). After modification, these motors were used throughout the full-scale test program and functioned satisfactorily. Based upon the results obtained during the full-scale tests, specifications for turn motors, which could be qualified for space flight, have been prepared and will be discussed later. The test turn motors weighed 10 pounds each, for a total of 20 pounds, including mounts.

Altitude sensors. - A preliminary survey of available altitude-sensing devices, conducted at program initiation, indicated that none of the devices furnished the degree of terrain height accuracy required for precise control of rocket-ignition altitude, and that a sensor must be developed for use with the land landing system. To determine the direction this development should take, a study of all known methods and principles that could be applied to sensing altitude was conducted, under contract, by Sylvania Electronics Division. This study recommended a continuous-wave homodyne electronic device backed up by a self-extending mechanical probe. The self-extending mechanical probe was selected for development for reasons of reliability, cost, and development schedule.

Two sensing devices were actually developed. An interim sensor was developed within the Manned Spacecraft Center for incorporation into early tests. This device consisted of a ball-mounted microswitch head which was gravity deployed and suspended beneath the test vehicle by its own electrical lead. While this sensor was never considered a flight quality item, it proved to be completely reliable.

The second altitude sensor, developed under contract by deHavilland Aircraft of Canada, Ltd., is a self-extending tubular probe that can be compactly stored until operation. A sensor head, located at the probe tip, contains a microswitch system which closes when it contacts the landing surface, sending the rocket-firing signal. This head can accept impact angles ranging from vertical to horizontal, on land or water.

To increase the reliability during full-scale tests, two deHavilland sensors were employed, either of which was capable of firing both rockets. Several minor design deficiencies were noted in this device, and a redesign effort was initiated to correct them. This effort was completed in late October 1965, and included 50 reliability firings under simulated air loads. These tests demonstrated that this sensor could be qualified for space flight. Each deHavilland sensor has a total installed weight of 7 pounds.

A detailed presentation of the development of both sensing devices is discussed in Volume II, Section V.

Pilot display and visual reference. - A study was conducted to determine the pilot display and visual reference system requirements necessary to accomplish a land landing solely under crew visual control. The approach followed was to define the minimum requirements and to indicate various types of devices capable of meeting the requirements, rather than to design the specific hardware.

This study consisted of two basic phases. The first was a series of helicopter simulated descents in which a pilot viewed the ground through various reticles and instructed the helicopter pilot to maneuver. The helicopter descended and maneuvered in simulation of Para-Sail performance. The second consisted of 50 drop tests of a 1/3-scale-model system which included a radio-command-actuated control system, and an onboard television system which viewed the ground at various angles through a variety of lenses and reticles. The pilot controlled the vehicle descent by radio command, with the television monitor as the sole visual reference.

These tests determined the following requirements. The field of view should contain as much of the landing area as possible, with a minimum included angle of 30° to the rear and both sides and at least 10° beyond the no-wind landing point to the front. The visual device may be of unity power and should include crosshairs to facilitate determination of wind drift. A compass and an altimeter should be included in the pilot display. With this type of system, visual control can begin at Gemini deployment altitude, and wind drift can be determined beginning at approximately 6000 feet.

Studies indicated four types of devices which can meet the necessary requirements. The types of devices are: closed-circuit television, optical periscope, fiber optics, and pop-up external mirrors. As an indication of visual system weight, a fiber optics bundle, meeting the requirements discussed, would weigh approximately 12 pounds, including mounts. A detailed presentation of the visual reference investigation is found in Volume II, Section VII.

Mechanical landing gear. - The mechanical landing gear incorporated in the system development program was originally designed for use with the Gemini paraglider, but was eliminated from the Gemini Program before its development was completed. This gear system, composed of three tricycle-arranged oleo-pneumatic shock absorbers coupled with struts and skids, was designed to impact on a prepared surface at rates of descent up to 10 ft/sec and horizontal velocities up to 100 ft/sec. These gears were included in the land landing system to attenuate the descent velocity remaining after rocket fire and to provide a stable touchdown system.

Before the gears were included in system tests, a separate test series was conducted to verify gear deployment and to determine attenuation capabilities over the range of vehicle attitudes and velocities expected when these gears were used with the gliding parachute and landing rockets. These tests indicated the following results:

1. The method of pyrotechnic deployment is satisfactory, including the design case of only one of the two available actuators firing.

2. Attenuation capability was established throughout the envelope.

| | |
|---|---|
| Descent velocity, ft/sec . . . . . . . . . . . . . . . | 0 to 12 |
| Horizontal velocity, ft/sec . . . . . . . . . . . . . . | 0 to 30 |
| Vehicle pitch attitude, deg . . . . . . . . . . . . . . | -8 to -18.7 |
| Vehicle yaw attitude (with 30 ft/sec horizontal velocity), deg . . . . . . . . . . . . . . . . . . . | ±15 |

The total weight of the landing gear is 310 pounds, including mounting bracketry. (See Volume II, Section IX for a detailed description of the landing gear functions and the verification tests.)

Test implementation hardware. - This area includes the design, manufacture, and verification of 41 items of test hardware. Volume II, Section VIII contains drawings and functional descriptions of these items, and Volume II, Section IX discusses the verification tests conducted before these items were included in system testing.

## Scaled-Model System Tests

Two scaled-model test programs were conducted to determine preliminary flight and landing characteristics of the integrated system. Both programs proved to be extremely accurate in the prediction of full-scale system characteristics.

One-third-scaled-model flight tests. - The primary objectives of this study were to investigate the dynamic behavior of the vehicle-parachute combination; to determine the parachute load distribution, control-line forces, and response to control inputs; and to obtain a preliminary evaluation of the visual references required.

Twenty-five air drops were made with the 1/3-scaled-model system. These drops featured a radio-command-actuated control system and a 24-foot-diameter Para-Sail, the performance of which was representative of that expected from the full-scale version.

The results of these tests indicated that the vehicle/parachute combination was dynamically stable, with oscillations of less than 5° about all three axes during straight flight. Change of vehicle attitude from heat-shield-down reentry to horizontal posed no stability problems; and control system force-travel requirements were quite reasonable. It was possible to maneuver the system to a preselected area and to land with the correct wind alinement. (A detailed presentation of this investigation is found in Volume II, Section I.)

<u>One-third-scaled-model landing dynamics tests.</u> - Tests were conducted with a 1/3 dynamically scaled model of the Gemini spacecraft with scaled landing-gear force-stroke simulation and cold-gas rocket-thrust simulation. These tests, conducted at MSC on the parallel-bar pendulum impact-test facility, were designed to determine the vehicle dynamics during rocket firing and associated impact and post-impact accelerations over a wide range of velocities and vehicle attitudes. Rocket-thrust simulation was accomplished with a high-pressure cold-gas system programed to provide scaled thrust-time characteristics. The cold-gas system was selected to allow adequate flexibility in varying rocket performance to meet test objectives.

Fifty tests, covering a wide range of conditions, established satisfactory operation of the landing-rocket/landing-gear attenuation system throughout the following envelope:

| | |
|---|---|
| Vertical velocity (initial), ft/sec . . . . . . | 30 |
| Horizontal velocity, ft/sec . . . . . . . . . . . | 0 to 30 |
| Vehicle yaw attitude, deg . . . . . . . . . . | ±15 (about 0) |
| Vehicle pitch attitude, deg . . . . . . . . . . | ±5 (about 13° nose down) |
| Vehicle roll attitude, deg . . . . . . . . . . | ±10 (about 0) |

A detailed description of this test program is contained in Volume II, Section II.

## Flight Operational Analysis and Planning

Concurrent with system development, operational analyses were conducted to design and to plan a Gemini flight mission with the land landing system as the planned means of mission termination. These operational studies are presented in detail in Volume II, Section X, and the results are summarized briefly here.

Many farm and ranch lands located within the continental United States met the landing-zone requirements. Two of these zones, sufficiently separated to insure no meteorological correlation yet available on the same orbital track, were designated as the primary and backup landing zones. These selected zones also allowed for contingency landings in the Gulf of Mexico.

A ground advisory station would be located in each of the two landing zones to exercise advisory control during the parachute descent, either in the event of cloud cover or at night. The station would provide weather observation and wind measurement, record and display spacecraft telemetry signals, and relay the spacecraft communications to the Mission Control Center.

During the final orbital phase, the preferred landing zone would be selected based on surface winds and cloud cover. Once the zone was determined, a reentry maneuver, programed to allow main parachute deployment over that zone, would be executed. Following the main parachute deployment and attitude repositioning, the flight crew would evaluate the available landing area within the zone, select the desired touchdown point, maneuver to it, and land.

## FULL-SCALE SYSTEM TESTING AND ANALYSES

The test program was conducted in two phases. The first phase consisted of nine developmental air drops, in which the vehicle landed in the water without the mechanical landing gear; and two crane drops, including the landing gear, were conducted over land. The water landing site was selected to protect the test vehicle before the addition of the rockets, and during the verification of ignition altitude. These tests also demonstrated water landing capability. Once the system was integrated and verified, the test location was shifted to Fort Hood, Texas, to provide a land site. At this location, the second-phase test program, consisting of three land landings, was conducted.

### Test Procedure

The Gemini spacecraft reentry conditions were simulated as closely as possible on all tests by releasing the vehicle from the drop aircraft with the R and R canister mated; and the test system was allowed to begin its descent on the static-line-deployed drogue parachute (fig. 6). All the events which followed were initiated either by a sequence programer or by a radio command.

As the test vehicle passed through nominal Gemini spacecraft deployment altitude, the R and R canister was jettisoned pyrotechnically, which automatically deployed the Para-Sail. The drogue parachute provided the force for the separation of the R and R canister and the extraction of the main parachute, and also recovered the R and R canister.

During the deployment and the inflation of the Para-Sail, the test vehicle remained oriented in the heat-shield down, or reentry, attitude, with the parachute attached to a single point on the nose-gear disconnect pallet (fig. 7). To further simulate the actual spacecraft flight, the test vehicle remained suspended in this attitude for a short time interval which theoretically would allow the flight crew to inspect the parachute through the hatch windows and, in the event of malfunction, eject from the favorable reentry attitude. At the end of this interval, the test vehicle was repositioned by releasing the riser group pyrotechnically at the single-point attachment and allowing the vehicle to tip over to the flying, 13° nose down, attitude (fig. 8).

In the flying attitude, the vehicle was suspended by the bridle system shown in figures 8 and 9. The rear half of the canopy was connected to hardpoints located on each side of the vehicle just forward of the heat shield, and the front half was connected to the confluence of a V-bridle. The forward leg of the V-bridle was attached to the nose-gear disconnect pallet, and the aft leg was connected to a hardpoint located in the center strip-out channel.

In tests which incorporated the rocket motors, a mechanical blast deflector was added as a test safety device. The blast deflector (fig. 10) was employed to cancel the rocket thrust and vent the exhaust gases, if inadvertent firing occurred while the test vehicle was in the launch aircraft. During drop tests, the blast deflector was jettisoned pyrotechnically shortly after attitude change and was recovered by a separate parachute.

In the normal drop sequence, the turn system was verified immediately following blast-deflector jettison. The turn motors, located in the Paraglider Control System (PCS) hatch beam just forward of the heat shield, were activated by radio command signals sent from a ground console (fig. 11).

The next programed event was deployment of the altitude sensors, either the MSC interim sensor or two deHavilland self-extending probes. Following deployment, the microswitch positions were checked by built-in inspection circuitry, prior to final arming of the rocket-firing circuits. When the turn potential and altitude-sensor deployment were verified, the mechanical landing gear was deployed by radio command, which completed the preparation sequence and placed the system in the final landing mode.

In the last 8000 feet of descent, the test system was maneuvered to the landing area and was faced into the wind near the surface. Since the system was normally released upwind of the target area, 1-1/2 to 2 miles of range capability were available. Wind drift was determined during descent by observing the system visually from the command console station.

When the altitude sensor contacted the landing surface, the microswitches closed, firing the rocket motors and decelerating the test vehicle (fig. 12). The test vehicle then landed on the mechanical gear at the reduced vertical velocity and dissipated any horizontal velocity in slideout (fig. 13). As the main landing gear stroked, it activated a disconnect system which jettisoned the parachute to prevent ground drag. In tests where the rockets were fired during a water landing (fig. 14), the mechanical gear was not included, and the parachute was jettisoned by a salt-water-activated switch.

The test system also included a radio-command-actuated emergency system to recover the test vehicle in the event of main parachute failure. The radio signal actuated the emergency programer which immediately jettisoned the main canopy, by releasing the riser attachments and severing the turn cables. A drogue gun fired 0.8 second later, deploying a 6-foot pilot parachute which extracted and deployed an 84-foot ringsail parachute. All subsequent programed events were locked out automatically by the emergency signal.

## Test Instrumentation and Sequencing

Data collection. - The data collection system employed during the full-scale tests consisted of 43 dynamic measurements, 5 real-time control-function displays, 6 radio-command channels, and an onboard television system which was monitored real time and recorded on video tape.

The basic telemetry measurements included:

Total riser loads
Individual riser loads
Turn-line forces, position, and commands
Three axis accelerations, linear
Angular accelerations, pitch axis
Impact accelerations, three axes
Thrust-line accelerations
Ambient pressure
Ambient temperature at the rocket nozzles
Yaw angular rate
Roll angular rate

Pitch angular rate
Rocket-chamber pressures
Event records

These telemetered data were picked up at the mobile ground station by two receivers and recorded on magnetic tape. These tapes were then discriminated and digitized. An analog-oscillograph record was printed for preliminary evaluation; then the digitized tapes were fed through a computer which reduced, tabulated, and plotted the results. (A detailed description of the data collection system is contained in Volume II, Section XI.)

Photographic coverage. - Initially, four motion-picture cameras were located on board the test vehicle. Two periscope cameras were added later. All cameras were started by the sequence programer, and imprinted with 100-cps timing lights furnished by a central time-pulse generating system. These cameras were as follows:

1. A 16-mm gun-sight aiming point (GSAP), set at 64 frames/sec. It was located in the cylindrical section, and recorded the parachute deployment.

2. A 16-mm Milliken, set at 24 frames/sec. It was located on the upper side of the vehicle just forward of the right hatch, and recorded the parachute after attitude change.

3. A 16-mm Milliken, set at 24 frames/sec. It was located on the underside of the vehicle near the heat shield, and was pointed down at the ground, after attitude change. (The same field of view was covered by the television camera.)

4. A 16-mm Milliken, set at 24 frames/sec. It was located so that it duplicated the command pilot field of view through the hatch window.

5. A 16-mm GSAP, with periscope, set at 64 frames/sec. It was located in the top cylindrical section and recorded the riser and the turn-line strip-out.

6. A 16-mm GSAP, with periscope, set at 64 frames/sec. It was located in the bottom cylindrical section, and recorded the blast-deflector jettison and the altitude-sensor deployment.

Nine exterior motion-picture cameras recorded the tests. The cameras were time-correlated by a large flashbulb mounted at the rear of the drop aircraft, and the bulbs fired automatically when the vehicle was released. These cameras were as follows:

1. Two 16-mm Millikens, set at 100 frames/sec. They were mounted overhead in the drop aircraft and recorded the launch.

2. Two 16-mm Millikens, set at 100 frames/sec. They were operated from a (high-altitude photography) chase aircraft, and recorded the launch, the deployment, and the attitude repositioning.

3. One 16-mm Milliken, set at 100 frames/sec. It was operated from a (low-altitude photography) chase helicopter, and recorded the final descent and the landing.

4. Three 16-mm Millikens, two set at 24 frames/sec, and one with a long-range lens set at 100 frames/sec. All the cameras were operated from the landing surface, and recorded the entire descent and the landing.

5. One 70-mm Hulcher tracking camera, set at various frame speeds. It was operated from the landing surface, and recorded the landing.

Sequencing. - The sequencing system was employed to control automatically the events that were required to function in a specific order and at a precise time. Mission events were obtained through the initiation of pyrotechnically actuated hardware. Two identical sequencing systems were employed for total redundancy. Each of these systems featured monitoring circuitry to facilitate pretest verification, circuit interrupters which allowed shorting of installed pyrotechnics until the drop run, and inspection and lockout circuitry which verified the altitude-sensor and rocket-firing circuits in flight, just prior to rocket arming. (A detailed presentation of sequencing, circuitry, and checkout procedures is contained in Volume II, Section XII.)

## Test Descriptions

Each of the 14 full-scale tests marks a milestone in the development program; and, as such, each test warrants individual attention. This portion of the report describes each test in detail, including specific objectives, onboard systems, landing weight and center-of-gravity location (where applicable), and a brief description of results.

## TEST 1 - TRINITY BAY, FEBRUARY 2, 1964

### Objectives

This was the initial test in the full-scale series. In addition to verifying the cradle drop method, it had the following specific objectives: to verify the R and R canister drogue and separation systems, to verify Para-Sail deployment from the separated R and R canister and measure opening loads, to verify the 1/3-scaled model results with regard to system dynamics during attitude change, and to verify the 1/3-scaled model results with regard to steady-state suspension attitude and stability.

### Onboard Systems

The onboard systems included an 18-ft ringsail drogue, reefed to 13 percent for 10 seconds; an R and R canister separation mechanism; an attitude-change-disconnect mechanism; and an 80-ft $d_o$ Para-Sail, reefed to 12.3 percent, with a 1000-pound reefing line with 6-second cutters.

### Sequencing

| | |
|---|---|
| Release, sec | 0 |
| R and R canister separation, sec | +8 |
| Attitude change, sec | +25 |

### Launch Conditions

| | |
|---|---|
| Altitude, ft | 6000 |
| Velocity, KIAS | 127 |
| Dynamic pressure at R and R canister separation, lb/ft$^2$ | 59 |

### Landing Weight

The landing weight of the test vehicle was 3972 pounds.

## Results

Although the drop method, using a gravity-activated launching cradle, imparted a high tumble rate to the vehicle as it left the aircraft, the reefed drogue essentially had stabilized the system at R and R canister separation. The Para-Sail deployed satisfactorily from the separated R and R canister and disreefed evenly. Attitude change was accomplished with no difficulty, and the canopy/vehicle combination was completely stabilized in slightly less than 8 seconds. The total system was extremely stable during descent, with no discernible oscillation. The vehicle suspension attitude was approximately horizontal. The steady-state rate of descent was 19 to 20 ft/sec. All test objectives were met.

# TEST 2 - TRINITY BAY, APRIL 8, 1964

## Objectives

The turn motors were added to the system for this test. Turn-line length was set by ground-inflating the parachute and marking link zero, then adding 19 inches for riser elongation (measured by suspending the test vehicle from two cranes). Turn-line length determination will be discussed in detail under Results. In addition to validating the operation of the turn motor and command systems, this test had the following objectives: the investigation of the dynamic behavior of the system during maneuvering in flight, the attempted correlation with the 1/3-scaled model results, and the investigation of turn rate, turn-line length, force, and travel.

## Onboard Systems

The onboard systems included an 18-ft ringsail drogue parachute, reefed to 13 percent for 10 seconds; R and R canister separation and attitude change mechanisms; and 80-ft $d_o$ Para-Sail, reefed to 12.3 percent, with a 1000-pound reefing line with 6-second cutters; radio-command-actuated turn motors, with 21 inches of travel (Aircraft Armaments, Inc.); an 84-ft ringsail emergency parachute system (radio commanded); and an impact-switch canopy-disconnect system.

## Sequencing

| | |
|---|---|
| Release, sec | 0 |
| R and R canister separation, sec | +8 |
| Attitude change, sec | +25 |
| Activate turn motors, sec | +35 |
| Disconnect parachute and cut turn lines | Impact |

## Launch Conditions

| | |
|---|---|
| Altitude, ft | 11 400 |
| Velocity, KIAS | 127 |
| Dynamic pressure at R and R canister separation, lb/ft$^2$ | 59 |

## Landing Weight

The landing weight of the test vehicle was 4760 pounds.

## Results

The initial sequence of events occurred as programed. The vehicle separated from the launch aircraft, stabilized on the drogue, and the R and R canister separated cleanly, deploying the Para-Sail. Just prior to full reefed inflation, the reefing line failed structurally, and the subsequent loading, caused by premature disreef, failed several Para-Sail suspension lines. The test was aborted at T + 22 seconds, and the emergency command was transmitted to the vehicle. The emergency system separated the damaged Para-Sail and deployed the recovery parachute. The test vehicle descended at approximately 30 ft/sec and impacted without damage. None of the test objectives were met. The test did serve the unplanned purpose of validating the emergency system.

# TEST 3 - TRINITY BAY, APRIL 29, 1964

## Objectives

This test was scheduled as a repeat of test 2 and had the same objectives. The reefing line was replaced with a stronger one, and drogue time was shortened to reduce deployment dynamic pressure.

## Onboard Systems

The onboard systems included an 18-ft ringsail drogue, reefed to 13 percent for 10 seconds; R and R canister separation and attitude-change mechanisms; an 80-ft $d_o$ Para-Sail, reefed to 12.3 percent, with a 2000-pound reefing line with 6-second cutters; radio-command-actuated turn motors, with 21 inches of travel; and an impact-switch canopy-disconnect system.

## Sequencing

| | |
|---|---|
| Release, sec . . . . . . . . . . . . . . . . . . . . | 0 |
| R and R canister separation, sec . . . . . . . . . | +5 |
| Attitude change, sec . . . . . . . . . . . . . . . | +25 |
| Activate turn motors, sec . . . . . . . . . . . . | +35 |
| Disconnect parachute and cut turn lines . . . . | Impact |

## Launch Conditions

| | |
|---|---|
| Altitude, ft . . . . . . . . . . . . . . . . . . . . . . | 11 100 |
| Velocity, KIAS . . . . . . . . . . . . . . . . . . . . | 127 |
| Dynamic pressure at R and R canister separation (calculated), lb/ft$^2$ . . . . . . . . . . . . . | 40 |

## Landing Weight

The landing weight of the test vehicle was 4760 pounds.

## Results

All systems functioned correctly, with normal parachute inflation and disreef. Following attitude change, the turn system was activated, and both left and right turns were verified as furnishing rates of turn of 10 to 12 deg/sec. Approximately 5 seconds were required for the system to accelerate to a steady turn rate. During turn, the canopy/vehicle combination banked 10 to 15° and pitched nose down 10 to 15°. When the turn was released, the system reoriented to the stable flight attitude in 3 to 4 seconds. Turn-line forces were approximately 80 pounds, with peaks up to 108 pounds. The test vehicle was maneuvered approximately 1-1/2 miles crosscountry to the primary target area and faced into the wind, prior to impact. The vehicle landed approximately 50 yards from the aiming point at target center. All test objectives were met. Figure 15 shows the system at the moment of impact.

## TEST 4 - TRINITY BAY, MAY 14, 1964

### Objectives

The interim altitude sensor was added to the system for this test, and flashbulbs were used to simulate rocket ignition. In addition to the verification of the deployment and performance of the interim sensor, this test had the following objectives: the continuation of the flight dynamics investigation, the measurement of the firing circuitry lag from signal, and the evaluation of the larger drogue parachute (simulated by using 22-ft $d_o$ ringslot with increased reefing percentage).

### Onboard Systems

The onboard systems included a 22-ft $d_o$ ringslot drogue, reefed to 19 percent for 10 seconds; R and R canister separation and attitude-change mechanisms; an 80-ft $d_o$ Para-Sail, reefed to 12.3 percent, with a 2000-pound reefing line with 6-second cutters; radio-command-actuated turn motors, with 21 inches of travel; an 84-ft $d_o$ ringsail emergency parachute system (radio commanded); an interim altitude sensor; and an impact-switch canopy-disconnect system.

### Sequencing

| | |
|---|---|
| Release, sec . . . . . . . . . . . . . . . . | 0 |
| R and R canister separation, sec . . . . | +5 |
| Attitude change, sec . . . . . . . . . . . | +25 |
| Deploy altitude sensor, sec . . . . . . . . | +35 |
| Activate turn motors, sec . . . . . . . . . | +40 |
| Arm altitude sensor, sec . . . . . . . . . | +55 |
| Fire flashbulbs . . . . . . . . . . . . . . . | Sensor closure |
| Disconnect parachute and cut turn lines . . . . . . . . . . . . . . . . | Impact |

### Launch Conditions

| | |
|---|---|
| Altitude, ft . . . . . . . . . . . . . . . . . . . . | 11 000 |
| Velocity, KIAS . . . . . . . . . . . . . . . . . . . | 127 |
| Dynamic pressure at R and R canister separation (calculated), lb/ft$^2$ . . . . . . . . . . . . . . . . . | 40 |

### Landing Weight

The landing weight of the test vehicle was 4791 pounds.

### Results

All systems functioned satisfactorily. One minor malfunction occurred which did not compromise test objectives. At attitude change, the forward V-bridle leg-load cell failed structurally, causing approximately a 20° nose-down suspension attitude. The higher extraction ratio caused by the larger drogue did not affect parachute deployment. Altitude-sensor deployment was satisfactory and closely resembled the static deployment tests.

The turn system was activated, and both left and right turns were verified as furnishing turn rates of 10 to 12 deg/sec. The test system was maneuvered approximately 2 miles crosscountry and faced into the wind, before landing approximately 40 yards from the center aiming point in the primary target zone. The flashbulbs, simulating the rocket motors, fired at altitude sensor closure. Within the accuracy of the measuring equipment, the firing-train time delay was established at just under 10 milliseconds.

## TEST 5 - TRINITY BAY, MAY 26, 1964

### Objectives

This test continued the investigation of descent dynamics and interim altitude-sensor performance. In an effort to obtain higher rates of turn, the turn motors were modified to provide 42 inches of travel. Specific objectives included the evaluation of increased turn-line travel and the resulting system dynamics.

## Onboard Systems

The onboard systems included a 22-ft $d_o$ ringslot drogue parachute, reefed to 19 percent for 10 seconds; R and R canister separation and attitude-change mechanisms; an 80-ft $d_o$ Para-Sail, reefed to 12.3 percent, with a 2000-pound reefing line with 6-second cutters; radio-command-actuated turn motors, with 42 inches of travel; an 84-ft $d_o$ ringsail emergency parachute system (radio commanded); an interim altitude sensor; and an impact-switch canopy-disconnect system.

## Sequencing

| | |
|---|---|
| Release, sec | 0 |
| R and R canister separation, sec | +5 |
| Attitude change, sec | +25 |
| Deploy altitude sensor, sec | +35 |
| Activate turn motors, sec | +40 |
| Arm altitude sensor, sec | +55 |
| Fire flashbulbs | Sensor closure |
| Disconnect parachute and cut turn lines | Impact |

## Launch Conditions

| | |
|---|---|
| Altitude, ft | 11 100 |
| Velocity, KIAS | 127 |
| Dynamic pressure at R and R canister separation (calculated), $lb/ft^2$ | 40 |

## Landing Weight

The landing weight of the test vehicle was 4761 pounds.

## Results

All systems functioned satisfactorily with one exception; at attitude change, the right turn line failed structurally, ruling out any use of right turn during the test. The left-turn system performed satisfactorily, although the steel cable guide damaged the cable. A left turn of approximately 20 deg/sec was obtained. As predicted by the 1/3-scaled model tests, and seen in earlier full-scale tests, the vehicle banked out and rotated nose down during turn.

Altitude-sensor deployment and arming were satisfactory. Using only the left-turn potential, the test vehicle was maneuvered approximately 1 mile crosscountry to the primary target area and faced into the wind. Impact occurred approximately 90 yards from the center aiming point. The flash-bulbs, simulating the rocket motors, fired on sensor closure. Measurements again indicated that the firing-train time delay was approximately 10 milli-seconds. The landing acceleration was below the 5g setting on the impact switch, and the canopy was disconnected by radio command. Figure 16 shows the test system during flight.

## FULL-SCALE CRANE DROP I - ELLINGTON AIR FORCE BASE, TEXAS, JULY 31, 1964

### Objectives

Prior to incorporation of the landing rockets into system drop testing, a full-scale crane drop of the test vehicle, with the landing rockets and landing gear, was conducted to allow evaluation of system performance under closely controlled conditions. Specific objectives of this test included the following: the verification of the analytically determined ignition altitude, the verification of the 1/3-scaled model impact g-time histories, the measurement of loads in landing gear structural members, and the verification of the thrust/vehicle center-of-gravity alinement method and alinement accuracy.

### Onboard Systems

The onboard systems included an interim altitude sensor, the Gemini spacecraft landing gear (predeployed), the landing rockets, and a backup rocket ignition system.

### Sequencing

The sequencing system included an altitude sensor, predeployed and remotely armed; rockets remotely armed; vehicle release accomplished by hardline signal; and rockets fired on altitude-sensor closure.

## Launch Conditions

The vehicle was suspended in the correct landing attitude and with sufficient height to furnish nominal 80-ft $d_o$ Para-Sail descent velocity of 25 ft/sec at altitude-sensor closure. The impacting surface was 1/2-inch steel plate over concrete. The nose landing gear was pressurized to 225 lb/in.$^2$, and the main gear hydraulic dampers were pressurized to 500 lb/in.$^2$. A lanyard block was attached and adjusted, so that it would actuate the rocket-firing circuits 12 inches below the height of altitude-sensor contact in the event of altitude-sensor failure. Data were recorded by means of a direct umbilical from the vehicle to an oscillograph.

## Landing Weight

The landing weight of the test vehicle was 4690.5 pounds.

## Center-of-Gravity Location

The center of gravity of the test vehicle was $Z = 131.79$, $X = +0.048$, and $Y = -1.79$.

## Results

The drop sequence is presented in figure 17. Figure 17(a) shows the test vehicle just after release, with the interim sensor and landing gear predeployed. Figure 17(b) shows the system in free fall just prior to rocket ignition. Note that the altitude sensor is 18 to 20 inches above the surface. Figure 17(c) shows rocket-motor ignition, and 17(d) shows high thrust. During high thrust, the vehicle experienced a 5° change in pitch attitude, which was attributed to mechanical deflection of the rocket support structure. Figure 17(e) shows the test vehicle just after touchdown and gear stroke.

Nominal velocity, acceleration, and roll attitude were achieved, and the descent velocity at touchdown was 7 ft/sec. Maximum accelerations measured in the Y-axis were 3.89g at rocket peak thrust and 3.90g at impact. When these accelerations are compared with flight-test results, the 1g that the pilot would feel during parachute descent must be added to the peak rocket thrust. Impact accelerations are directly comparable. Touchdown occurred prior to tail-off of the low thrust level, as programed. Visual inspection of landing gear components after the test revealed no evidence of heat damage.

### Conclusion

Ignition altitude was correct, and thrust alinement was within tolerance. Accelerations were controlled below a level of 5g, as predicted by the scaled-model tests. All test objectives were met. The motor support structure was stiffened following this test.

## TEST 6 - TRINITY BAY, OCTOBER 16, 1964

### Objectives

This test incorporated the landing rockets into the drop-test program. To provide realistic system data, the altitude for ignition was set as for a land landing. (In a water landing, with the landing gear stowed, this setting is approximately 2 feet higher than optimum.) In addition to the verification of rocket-motor performance, this test had the following objectives: the verification of the blast-deflector jettison and recovery systems, the evaluation of g-time histories during rocket fire and impact, the evaluation of thrust/vehicle center-of-gravity alinement, the continued evaluation of system dynamics, and the demonstration of water landing capability.

### Onboard Systems

The onboard systems included a 22-ft $d_o$ ringslot drogue parachute, reefed to 19 percent for 10 seconds; R and R canister separation and attitude-change mechanisms; an 80-ft $d_o$ Para-Sail, reefed to 12.3 percent, with a 2000-pound reefing line with 6-second cutters; radio-command-actuated turn motors, with 42 inches of travel; an 84-ft $d_o$ ringsail emergency parachute system (radio commanded); an interim altitude sensor; a blast deflector, jettison mechanism, and recovery system; landing rockets; and a salt-water-activated switch for parachute disconnect.

### Sequencing

| | |
|---|---|
| Release, sec . . . . . . . . . . . . . . . . . . . . . . . | 0 |
| R and R canister separation, sec . . . . . . . . . . . . | +5 |
| Attitude change, sec . . . . . . . . . . . . . . . . . . . | +25 |
| Blast-deflector jettison, sec . . . . . . . . . . . . . . . . | +40 |
| Deploy altitude sensor, sec . . . . . . . . . . . . . . . . | +55 |

| | |
|---|---|
| Activate turn motors, sec . . . . . . . . | +65 |
| Arm altitude sensor, sec . . . . . . . . . | +75 |
| Arm rocket motors, sec . . . . . . . . . | +95 |
| Fire rockets . . . . . . . . . . . . . . . . | Sensor closure |
| Disconnect parachute and cut turn lines . . . . . . . . . . . . . . . . | Impact |

### Launch Conditions

| | |
|---|---|
| Altitude, ft . . . . . . . . . . . . . . . . . . . . . . | 11 200 |
| Velocity, KIAS . . . . . . . . . . . . . . . . . . . . | 126 |
| Dynamic pressure at R and R canister separation (calculated), $lb/ft^2$ . . . . . . . . . . . . . . . . . . | 40 |

### Landing Weight

The landing weight of the test vehicle was 4768 pounds.

### Center-of-Gravity Location

The center of gravity of the test vehicle was $X = +0.19$, $Y = -1.62$, and $Z = 132.3$.

### Results

All systems functioned correctly through attitude change, blast-deflector jettison, and altitude-sensor deployment. When the turn motors were activated, the parachute did not respond. Investigation later proved that the turn lines were too short, which overloaded the motors, causing the fuses to blow. The same method of setting the turn-line length had been employed for this test, as with the one parachute used for all previous tests, but the method proved to be ineffective due to differences in suspension line elongations.

The test system made a long gliding descent and impacted crosswind, approximately 2-1/2 miles from the primary target area. The rocket motors fired at altitude-sensor closure, decelerating the vehicle prior to impact. Maximum accelerations recorded were approximately 4.8g. There was no discernible change in the attitude of the test vehicle during rocket firing, which indicated a correct thrust/center-of-gravity alinement. Figure 14 shows the system during rocket high-thrust burning.

## TEST 7 - TRINITY BAY, DECEMBER 11, 1964

### Objectives

This test incorporated the improved version of the 70-ft $d_o$ Para-Sail and the deHavilland altitude sensor into system testing. Specific objectives of this test included the following: the evaluation of system performance with the improved parachute configuration, the evaluation of the deHavilland altitude sensors, and the determination of system dynamics during rocket firing and impact.

### Onboard Systems

The onboard systems included a 22-ft $d_o$ ringslot drogue parachute, reefed to 19 percent for 10 seconds; R and R canister separation and attitude-change mechanisms; a 70-ft $d_o$ Para-Sail, reefed to 12.35 percent for 8 seconds; radio-command-actuated turn motors, with 42 inches of travel; an 84-ft $d_o$ ringsail emergency parachute (radio commanded); two deHavilland altitude sensors; a blast deflector, jettison mechanism, and recovery system; landing rockets; and a salt-water-activated switch for parachute disconnect.

### Sequencing

| | |
|---|---|
| Release, sec . . . . . . . . . . . . . . . . . . | 0 |
| R and R canister separation, sec . . . . . | +5 |
| Attitude change, sec . . . . . . . . . . . . . | +25 |
| Blast-deflector jettison, sec . . . . . . . . . | +40 |
| Deploy altitude sensor, sec . . . . . . . . | +55 |
| Activate turn motors, sec . . . . . . . . . | +60 |
| Arm altitude sensor, sec . . . . . . . . . . | +75 |
| Arm rocket motors, sec . . . . . . . . . . . | +95 |
| Fire rockets . . . . . . . . . . . . . . . . . . | Sensor closure |
| Disconnect parachute and cut turn lines . . . . . . . . . . . . . . . . . . | Impact |

## Launch Conditions

| | |
|---|---|
| Altitude, ft . . . . . . . . . . . . . . . . . . . . . . | 7000 |
| Velocity, KIAS . . . . . . . . . . . . . . . . . . . . | 127 |
| Dynamic pressure at R and R canister separation (calculated), $lb/ft^2$ . . . . . . . . . . . . . . . . . . . | 40 |

## Landing Weight

The landing weight of the test vehicle was 4747 pounds.

## Center-of-Gravity Location

The center of gravity of the test vehicle was $X = +0.1402$, $Y = -1.822$, and $Z = 130.78$.

## Results

All systems functioned correctly through attitude change, blast-deflector jettison and recovery, and altitude-sensor deployment. One deHavilland sensor bent near the root and trailed approximately 15° aft. When the turn motors were activated, the system did not respond. Investigation later indicated that the turn cables failed structurally at attitude change, due to tensile load and possible fouling. The system made an uncontrolled, spiral descent, and impacted approximately 200 yards from the target area. The rockets fired at altitude-sensor closure, decelerating the vehicle prior to impact. Again, the nearly constant attitude during rocket firing indicated correct thrust/center-of-gravity alinement.

# TEST 8 - TRINITY BAY, JANUARY 14, 1965

## Objectives

Due to successive failures in the turn system, the test order was changed at this point. Prior to this test, a series of static attitude-change tests was conducted to determine riser and turn-line strip-out characteristics. At the same time, an intensive analytical effort was conducted to determine actual zero turn length for the advanced version of the Para-Sail. As a result of these studies and tests, turn-line length was increased 18 inches, and the lines were stowed alongside, but separate from, the main riser bundle in the strip-out channel.

The objectives of this test were to evaluate the modified means of turn-line stowage, to acquire additional turn-motor force-travel and turn-line-length data, and to acquire more controlled flight experience with the 70-ft $d_o$ Para-Sail.

### Onboard Systems

The onboard systems included a 22-ft $d_o$ ringslot drogue parachute, reefed to 19 percent for 10 seconds; R and R canister separation and attitude-change mechanisms; a 70-ft $d_o$ Para-Sail, reefed to 12.35 percent for 8 seconds; radio-command-actuated turn motors, with 42 inches of travel; an 84-ft $d_o$ ringsail emergency parachute (radio commanded); and an impact-switch canopy-disconnect system.

### Sequencing

| | |
|---|---|
| Release, sec | 0 |
| R and R canister separation, sec | +5 |
| Attitude change, sec | +25 |
| Activate turn motors, sec | +35 |
| Disconnect parachute and cut turn lines | Impact |

### Launch Conditions

| | |
|---|---|
| Altitude, ft | 11 400 |
| Velocity, KIAS | 127 |
| Dynamic pressure at R and R canister separation (calculated), lb/ft$^2$ | 40 |

### Landing Weight

The landing weight of the test vehicle was 4543 pounds.

## Results

All systems functioned correctly. Although both turn lines were set at the same length, the parachute exhibited a built-in left turn of approximately 25 deg/sec. With the right-turn motor, it was possible to trim the built-in turn, or overpower it, and obtain a right-turn rate of 14 to 15 deg/sec. The maximum left-turn rate obtained was 40 deg/sec. Several turns were executed in flight, and the vehicle was maneuvered into the primary target area and faced into the wind prior to impact. Examination of the turn-motor cables after the test indicated evidence of wearing and fraying. All test objectives were met.

# TEST 9 - TRINITY BAY, FEBRUARY 25, 1965

## Objectives

The objectives of this test were to evaluate the effect of increased turn-line length and acquire additional force-travel data. This test also served as a final rehearsal on release point/maneuvering accuracy for the land landing.

## Onboard Systems

The onboard systems included a 22-ft $d_o$ ringslot drogue parachute, reefed to 19 percent for 12 seconds; R and R canister separation and attitude-change mechanisms; a 70-ft $d_o$ Para-Sail, reefed to 12.35 percent for 6 seconds; radio-command-actuated turn motors, with 42 inches of travel; an 84-ft $d_o$ ringsail emergency parachute (radio commanded); and an impact-switch canopy-disconnect system.

## Sequencing

| | |
|---|---|
| Release, sec . . . . . . . . . . . . . . . . . . . . . | 0 |
| R and R canister separation, sec . . . . . . . . . . | +5 |
| Attitude change, sec . . . . . . . . . . . . . . . . . | +25 |
| Activate turn motors, sec . . . . . . . . . . . . . . | +35 |
| Disconnect parachute and cut turn lines . . . . . . . . . . . . . . . . . . . . | Impact |

## Launch Conditions

| | |
|---|---|
| Altitude, ft . . . . . . . . . . . . . . . . . . . . | 11 200 |
| Velocity, KIAS . . . . . . . . . . . . . . . . . . . | 128 |
| Dynamic pressure at R and R canister separation (calculated), lb/ft$^2$ . . . . . . . . . . . . . . . . . . | 40 |

## Landing Weight

The landing weight of the test vehicle was 4735 pounds.

## Results

This drop was made in very high winds aloft, and with approximately 15 knots of surface wind. All systems functioned correctly. The new 70-ft $d_o$ Para-Sail used on this test had a 12 to 15 deg/sec built-in turn to the left. For this test, as in test 8, the turn lines were set at equal lengths. Approximately 40-percent travel of the right motor counteracted the built-in turn, while full right travel furnished a 6 to 8 deg/sec right turn. Full left turn, coupled with the built-in turn, resulted in a 45 to 50 deg/sec turn rate and high bank and pitch angles.

The test vehicle was approximately 1200 yards east (downwind) and 2500 yards north of the target when the turn system was activated. The canopy was immediately faced into the wind but could not advance toward the target. By heading the canopy at an oblique angle to the target, the north to south distance was compensated for, but only a slight gain was made to the west, due to the high winds. The system impacted approximately 1000 yards downwind from the primary target area.

The turn-line load data indicated that the left line should be lengthened 14 inches, and the right line should be lengthened 8 inches for this particular parachute. Since this parachute was to be used for the first land drop, this modification was made. Tests 8 and 9 validated the turn-line length and stowage, and concluded the water landing phase.

## Landing Accuracy and Wind Alinement

In the evaluation of the system's ability to be maneuvered to a specific landing point, it must be recognized that a basic difference existed between unmanned testing and an actual orbital reentry and landing under crew control. In an orbital flight case, reentry would be programed so that the main parachute opening would occur over a previously selected landing zone. Following main parachute opening, the flight crew would select their landing area, based on obstacles, wind profile, and so forth, and maneuver toward it. As the spacecraft descended, the crew would continuously reevaluate their attainable landing area and select the touchdown site. Below 500 feet, the spacecraft must maintain wind alinement, and hence would be committed to a landing point.

For unmanned tests in which the vehicle was maneuvered by ground radio command, the test area was selected first. The test vehicle was then released from the drop aircraft at a point in space which allowed the system to be maneuvered into the landing zone. A small target or aiming point was normally placed in the center of the primary zone. The accuracy data were concerned with the distances from the actual landing spot to the target. It was realized that landing dispersion resulting from drop testing was also affected by release-point accuracy. During the test programs, the release-point accuracy was sufficiently good to discount this factor. It must also be noted that the test vehicle was alined with the wind 200 to 500 feet above the surface, and that no further maneuvering for point accuracy was attempted.

In unmanned drop testing, the means of determining wind drift also differed from the orbital-reentry case. A descending flight crew would have a true picture of wind drift at all times since the crew would be looking down from the spacecraft. A test controller, located on the ground, was severely handicapped in determining drift when the vehicle was heading at right angles to his line of sight and was some distance removed.

As determined by the 1/3-scaled-model impact dynamics studies, the landing gear stability envelope was bounded in yaw by ± 15°. In terms of wind alinement at touchdown, these results indicated that the system must be alined with 15° of the wind for a stable landing. These criteria were used as a basis for determining successful alinement during tests. A variety of devices, primarily flags and streamers, was used to determine ground wind direction when control was accomplished by visual observation. Ground drift, as determined through the television monitor, was used in the visual reference-system study.

Three basic development program phases were conducted which allow accuracy and wind alinement data to be extracted. These were as follows:

One-third-scaled model test program of 25 drop tests. - One controller basically conducted all of the flights. Control was accomplished by observing the system in the air from a point on the ground near the target. During this program, the controller experienced no difficulty in alining the vehicle within 3 to 5° of the wind-direction indicators. As a measurement of accuracy, as previously defined, the controller was able to aline with the wind and consistently land the scaled-model spacecraft within an overall average of 150 feet from the target. As the controller grew more proficient toward the latter part of the program, this landing dispersion was reduced from 150 feet to 50 feet.

Visual reference-system program of 50 tests. - Many controllers were employed. The only visual reference was the television receiver monitoring a television camera mounted in the scaled-model vehicle. In this evaluation, it was found that two or three flights were required before an adequate level of proficiency was attained. After these flights, all the controllers were able to determine wind drift, select a landing area, control the vehicle to that area, and aline with the wind prior to touchdown. It was found during these tests that satisfactory landings could be made in zones that were 60 percent covered with obstacles. After the initial practice flights, these controllers could aline consistently with the wind and land the scaled-model spacecraft within 200 yards of the target. This test program most closely simulated actual flight conditions; hence, the results were most significant.

Full-scale system test program of 12 tests. - One controller conducted all of the flights. Control was accomplished by observing the system in the air from a point on the ground approximately 1/2 mile from the target. Six full-scale tests were conducted from which data could be drawn. One of these, test 9, impacted approximately 1000 yards from the target due to a faulty release point and high winds and was excluded from consideration. In the remaining five tests, the controller experienced no difficulty in alining within 5° of the wind-direction indicators when the control point was located near the surface wind line. In the final test, the winds were very light (1 to 2 knots), and wind direction varied over a wide range. The wind direction shifted approximately 100° between the time final alinement was initiated and touchdown. At an altitude of approximately 100 feet, the controller could discern no side drift and held the existing heading. The system touched down approximately 10° from downwind. Since the surface winds were very light, they were of little significance. In all tests, the average landing point distance from the target was slightly under 100 yards.

From the results of these three investigations, the following conclusions concerning wind alinement and accuracy were drawn. Wind alinement within acceptable tolerances was readily accomplished. With proper training, returning flight crews should be able to land the spacecraft within 200 yards of a desired point on the ground. The scaled-model vehicle with the onboard television system provided a valuable training tool.

## Analysis of Results

This section is devoted to an analysis and summation of the results presented in the foregoing section and the applicable component and scaled-model results discussed in Volume II.

Parachute system. - The 70-ft $d_o$ Para-Sail parachute (70A-5), developed during the program, satisfactorily demonstrated the performance required for a land landing of the Gemini spacecraft and could be qualified for space flight.

Data for a nominal opening force-time history for this parachute when it was employed with the land landing system, represented by a dotted line in figure 24, were described. (Volume II, Section III contains a detailed discussion of the parachute opening history when the parachute is deployed at both nominal $\left(80 \text{ lb/ft}^2\right)$ and ultimate strength conditions.)

As shown in figure 24, the parachute met the 16 000-pound structural limit on opening loads when deployed at nominal conditions. When deployed at the emergency condition of 120 lb/ft$^2$, the parachute exhibited an average reefed opening shock value of 21 900 pounds occurring approximately 3.0 seconds after initiation of deployment, decelerated to a constant velocity of 120 ft/sec prior to disreef, and followed a nominal force-time history thereafter.

System turn performance and motor requirements. - Studies and tests conducted during the development program determined the desired turn performance and the associated force and travel required by the parachute to achieve this performance. A force of approximately 92 pounds was required to furnish the maximum desired turn rate of 25 deg/sec (fig. 32), and 46 inches of travel are required to achieve this force (fig. 31).

With this turn potential, the system could be maneuvered to within 200 yards of a preselected point and landed with correct wind alinement. The purpose here was to define the performance required from a flight turn-motor system based on the preceding discussions of turn rates, force and travel requirements, motor speed, and trim. (A discussion of the mechanical design and functioning of the test turn motors is contained in Volume II, Section VI, along with the conclusions reached concerning design problem areas.)

The turn motors shall employ dc power and provide a force to overcome turn-cable loads from 0 to 250 pounds. Deflection of the controller system shall provide a controlled reel-in of 6.0 ± 0.3 feet in 6 ± 1 seconds. The motor design shall be such as to assure repeatable cable positions and smooth feed. Reel-out shall be accomplished in as little time as possible, not to exceed one-half the takeup time. Positive braking shall be employed to hold the commanded cable position at loads of from 0 to 500 pounds until the input signal is changed. Cables must be capable of withstanding loads up to 1500 pounds when the takeup drum is bottomed out, or when the brake is energized to furnish an adequate margin during the loading experienced at attitude change.

The motor system shall provide the capability to trim a 15 deg/sec turn rate. The 6 feet of travel previously specified is sufficient to provide the trim capability and to furnish the desired range of turn rates.

Visual reference. - Volume II, Section VII presents a detailed discussion of the visual reference investigation and defines the minimum acceptable system requirements. In summary, the flight crew must be able to see the area in which they might land and must have some means of determining wind drift.

Figure 41 presents the field of view from the command pilot's window in the straight flight and the high-rate turn attitudes. It can be seen from the figure that a high-rate turn maneuver does afford the command pilot a view of the outer circumference of the possible landing area, but does not include a view of the major portion of the zone. This is unacceptable since it does not provide a view of the probable touchdown point, and because high-rate turns cannot be accomplished near the surface. The minimum required field of view, as determined during the development program, includes all possible touchdown points in the direction the vehicle is heading (fig. 42).

Expected landing dynamics. - The results of the model and full-scale test programs demonstrated that the landing rocket/landing gear combination provided a stable land landing impact-attenuation system with accelerations controlled to a magnitude of approximately 5g when landings were made within the design envelope. This envelope was bounded by horizontal velocities from 0 to 30 ft/sec; initial vertical velocities of approximately 30 ft/sec; and limiting vehicle attitudes of $\pm 15^\circ$ in yaw, $\pm 10^\circ$ in roll, and $\pm 5^\circ$ in pitch. It should be noted that the landing gear was capable of landing with horizontal velocities up to 100 ft/sec on a prepared surface.

A comparison of the high- and low-horizontal velocity land landings showed the peak accelerations to be the same order of magnitude in both instances. Since these tests were made at the end points of the horizontal velocity range, it was reasonable to conclude that these two tests were representative of the entire landing envelope. This conclusion was substantiated by the 1/3-scale landing dynamics tests which indicated peak accelerations from 3g to 6g throughout the horizontal velocity envelope.

The slideout distance predicted by the scaled-model tests for the high-horizontal velocity condition was 60 feet. This value compared favorably with the 55 feet of measured slideout in test 12.

In an uncontrolled landing which featured an extreme yawed condition and/or rough terrain, the vehicle would probably tumble. Tumbling occurred in the 1/3-scaled-model program when the vehicle exceeded 15° in yaw, and in the full-scale program when the vehicle landed in an uncontrolled turn. However, the peak accelerations recorded in landings where the vehicle overturned were on the order of 8g to 10g, which was well within acceptable crew tolerance levels. Consequently, a flight crew could be expected to undergo, without receiving injury, a landing in which the vehicle tumbled.

A double-rocket-motor failure would result in unacceptable accelerations. The possibility of either motor failing was extremely remote, due to the redundant ignition system and the high reliability of solid-propellant fuels. Should one motor failure occur, the impact velocity of 18 ft/sec would exceed the limits of the existing landing gear and possibly injure the crew. The rocket motors developed during the program adequately demonstrated the required performance and could be qualified for space flight.

The landing gears originally developed for the paraglider were incorporated without change, and their operation was verified within the design landing envelope. These gears were not optimized for this type of application and did not necessarily represent the most efficient mechanical attenuation system. It should also be pointed out that all possible combinations of landing conditions have not been tested, and that additional model and full-scale tests would be required to man-rate the impact attenuation system.

A comparison of the water landing acceleration histories presented in figure 40 showed that the accelerations resulting from impact were decreased by the addition of the rocket motors. However, the accelerations experienced at high thrust with the rockets were higher than the peak impact acceleration without rockets. Based on this fact alone, it would appear that the water landing case had been penalized rather than improved. Actually, considerable improvement in the water landing case had been obtained.

Design parameters for the existing Gemini spacecraft landing system included a vertical descent velocity of 30 ft/sec, surface winds up to 51 ft/sec, parachute oscillations of ± 15° at a swing velocity of 14 ft/sec, and 9° wave slopes. Tests conducted at these conditions by the spacecraft contractor indicated vertical impact accelerations in the 12g to 16g range, with some measured peaks up to 22g (ref. 1).

The reduced accelerations experienced when the land landing system made a contingency water landing were due to considerable reduction in the governing parameters discussed. The descent velocity was reduced from 30 ft/sec to a nominal 8 ft/sec, and the horizontal velocity due to surface wind was reduced from a maximum of 51 ft/sec to a maximum of 30 ft/sec. The existing Gemini spacecraft parachute oscillated ± 15°, and the swing velocity thus created added to the resultant impact velocity. The gliding parachute, used with the land landing system, exhibited a maximum of ± 5° oscillation and negligible swing velocities; hence this factor was removed from impact considerations. Wave effect was greatly reduced by impingement of the rocket exhaust on the water surface. Analysis of high-speed film coverage of water landings indicated that the rocket exhaust had a cratering effect which created a dish-shaped impact surface free of waves. In addition, the high-speed agitation of the water surface by the rocket exhaust created a foam, or froth, which effectively softened the surface by increasing the compressibility.

Based on the results of the tests conducted during the development program, the expected impact accelerations, when a contingency water landing was made, were the same as those presented for a land landing. These accelerations were approximately one-third of those expected with the existing Gemini spacecraft landing system at the most adverse design conditions.

Rocket-thrust alinement. - At the beginning of the program, it was believed that alinement of the rocket-thrust line with the vehicle center of gravity would present a major problem area, since any misalinement would create a moment tending to upset the vehicle. The rocket mounts were designed (see Volume II, Section VIII) to allow travel in both the roll and pitch axes to allow precise alinement. The vehicle center of gravity was located prior to each test by suspending the test vehicle in three planes and calculating the center of gravity.

A special alinement fixture was designed to determine the thrust line (Volume II, Section VIII). The plug end of the fixture was placed in the nozzle (fig. 43), and the mounts were adjusted until the pointer physically coincided with the center-of-gravity mark (fig. 44).

Five air drops and two crane drops were conducted in which data on thrust alinement could be drawn. This was done by monitoring pitch and roll rates during rocket fire, and by a detailed analysis of high-speed test film coverage. During crane drop I, a 5° pitch change occurred immediately following peak high thrust. Analysis indicated that this resulted from deflection of the motor supports. These supports were structurally strengthened prior to the other tests in which the rocket motors were employed. In every remaining test, analysis of the data and film coverage showed no evidence of vehicle motion due to thrust misalinement. It was concluded that the thrust/center-of-gravity alinement method was successful, and that this area did not present the degree of technical difficulty originally expected.

Soil erosion. - Erosion damage to the landing surface, due to impingement of the rocket exhaust, and the subsequent effect on landing stability was recognized as a problem area at the initiation of the program. It was not believed that the high thrust level posed the major erosion threat since it occurred while the vehicle was 6 to 8 feet above the surface; rather, it was believed that low thrust occurring on or near the surface could create a stability problem.

To provide some insight into this phenomenon, two 1/3-scaled-model tests were conducted to obtain quantitative erosion data. During the first test, the scaled-model vehicle was placed on the compacted soil surface, and the cold-gas system was activated at the low-thrust level while the vehicle remained stationary. The simulated rockets eroded a crater 30 inches in diameter and 8 inches deep (Vol. II, fig. II-10). This amounted to a displaced surface volume of approximately 3.8 cubic feet.

In the second test, the model slid along the surface at a constant velocity of 3.62 ft/sec while the cold-gas system was activated at the low-thrust level. The erosion rut created was approximately 8 inches wide, 2 inches deep, and extended the total distance in which the simulated rockets fired (Vol. II, fig. II-11).

Analysis of high-speed film coverage of these two tests indicated distinctly different erosive patterns. In the test with no horizontal velocity, the crater was explosively formed as large chunks (approximately 6 inches in diameter) were blown away; whereas, when horizontal velocity was present, small particles were displaced and trailed aft in the high-velocity gas in a much more gradual manner. From these tests, it was evident that erosion surface damage could be significant during a purely vertical landing, and that this significance would be considerably lessened at higher horizontal velocities when slideout on the landing gear would occur.

To acquire full-scale information on erosion and resulting landing stability in the vertical descent case, the full-scale test vehicle with the landing rockets and landing gear was dropped from a crane to a rain-saturated sod surface. The erosion pattern closely resembled that of the first scaled-model test since the surface was explosively blown away in large chunks (approximately 1 cubic foot). The crater formed was 138 inches wide, 96 inches long, and 28 inches deep, as shown in figure 45. This amounted to a displacement of approximately 205 cubic feet of soil. No adverse stability effects were noted other than that the final attitude of the test vehicle included approximately 5° roll because the left main gear was at the end of the crater.

In drop test 10, the vehicle had a low horizontal velocity but landed on extremely hard, sun-baked soil. The erosion crater which was formed was almost identical in length and width to the crane drop but was only 12 inches deep due to a solid rock layer encountered at that depth. Since the vehicle tumbled during this landing for other reasons, no information on stability effect could be derived.

In test 12, the vehicle landed with a horizontal velocity of approximately 35 ft/sec. The landing surface showed no erosive effect resulting from high thrust other than the removal of fine surface dust. During low thrust, a narrow rut approximately 6 inches deep was formed (fig. 46). In this test, landing stability was in no way affected.

These tests verified landing stability at the horizontal velocity end points. Additional tests should be conducted to determine erosion and resulting stability effect at midrange velocities. The Manned Spacecraft Center is currently conducting a comprehensive investigation of this phenomenon.

Weight summary. - The weights of the land landing system components used during the development program are as follows:

| Component | Weight, lb |
|---|---|
| Parachute (bag, risers, and so forth) | 153 |
| Landing rockets (and mounts) | 60 |
| Two altitude sensors (deHavilland) | 14 |
| Visual system (fiber optics) | 12 |
| Two turn-control motors | 20 |
| Landing gears (installed) | 310 |
| Total | 569 |

The weights listed are those of the test hardware components used to develop and demonstrate the land landing system and do not represent flight components in which serious weight reduction efforts have been exercised.

# FULL-SCALE CRANE DROP II - ELLINGTON AIR FORCE BASE, TEXAS, MARCH 11, 1965

## Objectives

This test was conducted to furnish a final verification of impact dynamics under closely controlled conditions. It had the following additional specific objectives: the verification of the stiffened rocket-motor support structure, and the evaluation of soil erosion resulting from rocket impingement.

## Onboard Systems

The onboard systems included the landing rockets, the predeployed landing gear, the predeployed interim altitude sensor, and the lanyard-block backup rocket ignition system.

## Sequencing

The sequencing system included the rockets, remotely armed; the altitude sensor, remotely inspected and armed; the vehicle release accomplished by hardline signal; the altitude-sensor closure; and the rocket fire.

## Launch Conditions

The test vehicle was suspended in the nominal flying attitude from two cranes at a height of 18.11 feet, which was sufficient to furnish a nominal parachute rate of descent at sensor closure. The nose landing gear was pressurized to 225 $lb/in.^2$, and each of the main gear hydraulic dampers was pressurized to 500 $lb/in.^2$. A backup lanyard block system was installed which would ignite the rockets 12 inches below the height of altitude-sensor contact in the event of altitude-sensor malfunction. Data were recorded by means of a direct umbilical from the vehicle to an oscillograph.

## Landing Weight

The landing weight of the test vehicle was 4751 pounds.

## Center-of-Gravity Location

The center of gravity of the test vehicle was $X = +0.048$, $Y = -1.79$, and $Z = 131.79$.

## Results

All systems performed correctly. The drop sequence is presented in figures 18(a) to 18(g). Figure 18(a) shows the test setup. The interim sensor is in the center of the figure. Figure 18(b) shows the system free-falling, just prior to sensor closure. In 18(c), the precise moment of rocket ignition is shown. Figure 18(d) to 18(f) represent the progressive sequence of smoke and dust. Figure 18(g) shows the vehicle standing on the landing gear in the erosion crater as the smoke cloud dissipates.

Due to the absence of horizontal velocity, this test represented the most severe landing condition from the standpoint of surface erosion (fig. 19). The dimensions of the eroded crater were 132 inches wide, 96 inches long, and 28 inches deep. The surface blew out in large chunks. This was accounted for by the high moisture content of the soil.

The peak thrust acceleration of 4.3g occurred at a height of just over 8 feet, immediately following sensor contact. The peak acceleration was 3.25g, well within design conditions. No noticeable change in attitude occurred during rocket firing, again verifying proper thrust alinement with the center of gravity of the vehicle.

## Conclusions

The stiffening of the rocket-motor support structure following the first crane drop resulted in elimination of thrust misalinement due to mount deflection. The test data further verified the analytical and the scaled-model results of the system dynamics. It also indicated that surface erosion, due to rocket impingement, can be extensive at low horizontal velocities. Further study is needed to determine the effect of surface erosion with higher horizontal velocities, and operational procedures must be determined to satisfy the relationship of the vertical descent with the type of sod used in the test.

## TEST 10 - FORT HOOD, TEXAS, APRIL 21, 1965

### Objectives

This was the initial test of the system conducted over land. The objectives were to evaluate system performance in flight and landing operations.

### Landing Area

The selected test site was a mile-square tank assault range located on the Fort Hood, Texas, military reservation. Figure 20 presents an aerial view of the primary target area. In preparation for the test, the tank ruts were leveled with a scraper blade in a 1/16-square-mile area in the center of the zone. The command console station was located on the mound in the upper center of the figure.

### Onboard Systems

The onboard systems included a 22-ft $d_o$ ringslot drogue parachute, reefed to 19 percent for 12 seconds; R and R canister separation and attitude-change mechanisms; a 70-ft $d_o$ Para-Sail, reefed to 12.35 percent for 6 seconds; radio-command-actuated turn motors, with 42 inches of travel; an 84-ft ringsail emergency parachute (radio commanded); two deHavilland altitude sensors; a blast deflector, jettison mechanism, and recovery system; landing rockets; the Gemini spacecraft landing gear, retention system, and deployment system; television and video-tape systems; and a gear-stroke switch for parachute disconnect.

### Sequencing

| | |
|---|---|
| Release, sec | 0 |
| R and R canister separation, sec | +5 |
| Attitude change, sec | +25 |
| Activate turn system, sec | +35 |
| Blast-deflector jettison, sec | +40 |
| Altitude-sensor deployment, sec | +70 |
| Landing gear retaining-strap release, sec | +105 |
| Landing gear deployment, sec | +108 |
| Disconnect switch arm, sec | +113 |

| | |
|---|---|
| Begin circuitry inspection, sec | +120 |
| Arm rocket motors, sec | +150 |
| Rocket ignition | Sensor closure |
| Parachute disconnect | Gear stroke |

### Launch Conditions

| | |
|---|---|
| Altitude, ft | 11 500 |
| Velocity, KIAS | 127 |
| Dynamic pressure at R and R canister separation (calculated), $lb/ft^2$ | 40 |

### Landing Weight

The landing weight of the test vehicle was 4726 pounds.

### Center-of-Gravity Location

The center of gravity of the test vehicle was $Z = 131.64$, $X = -0.07$, and $Y = -2.9$.

### Results

All systems functioned correctly through main deployment, attitude change, and blast-deflector jettison. Prior to attitude change, the test system made a long run downwind toward the target. After attitude change, the system was faced away from the target and regained the lost ground. Two 360° left and two 360° right turns verified the turn system and indicated maximum turn rates of 15 to 18 deg/sec. At +70 seconds, the altitude sensors deployed, and the left sensor, bent at the root, trailed 5 to 7° aft of its normal extended position. After verification of altitude-sensor deployment, the landing gears were deployed by radio command. At approximately 3500 feet, the vehicle was again turned downwind for the run to the target area. At approximately 1400 feet, a right-turn input was radio commanded to set up a hook approach into target center. At this command, the right-turn line separated, and the system began to turn left. Several attempts were made to trim out this built-in turn, but they were ineffective, due to the severed right-turn line. Peak impact accelerations recorded were approximately 8g and occurred when the vehicle rolled over about the main gear axis.

Figures 21(a) to 21(f) depict the landing sequence. The parachute in the foreground is the blast-deflector recovery parachute, and may be used in these figures to indicate direction of ground wind. Figure 21(a) shows the system approximately 200 feet in the air. Note the bank angle indicating left turn. Also, the left-turn line is visible while the right line is missing. Figures 21(b) and 21(c) show the system in descent, just prior to impact. Note the rotation to the left. Figure 21(d) shows rocket ignition. In figure 21(e), the canopy continues to turn as the rockets fire. Note the roll angle of the test vehicle. In figure 21(f), the parachute has jettisoned, and the test vehicle has rolled on its side. Damage was limited to a sheared bracket on the nose landing gear.

## Failure Analysis

Two causative factors for the turn-cable failure were determined, as follows:

1. At attitude change, the vehicle nose pitched down momentarily. This bent the turn cables about the aft end of the cable cutter slots and put a permanent set in the cables, which then did not wind smoothly onto the drum.

2. The tension-measuring device inside the turn motor (Vol. II, fig. VIII-20) was slot mounted. An impact load placed on it (such as attitude change) could force it against the cable guide, jamming the roller. With the roller jammed, the cable did not wind evenly on the drum and could be severed by the ridges between the cable grooves. Microinspection of the cable guide, drum, and severed cable indicated this was the failure mode. The failure was subsequently reproduced in a static test.

## Corrective Action

A 360° Teflon fairing block was mounted above the cable-cutter slot to provide a rounded bearing surface for the cable. Cable diameter was increased to 3/16 inch. The tension-measuring device was removed, and new cable drums were fabricated with deeper and more widely spaced grooves to accommodate the larger diameter cable. This modification reduced travel to 37 inches.

## TEST 11 - FORT HOOD, TEXAS, MAY 27, 1965

### Objectives

This test repeated test 10. The objectives were to evaluate the system performance and to demonstrate the land landing. Prior to this test, the turn motors were modified to eliminate the cause of failure in the previous test.

### Onboard Systems

The onboard systems included a 22-ft $d_o$ ringslot drogue parachute, reefed to 19 percent for 12 seconds; R and R canister separation and attitude-change mechanisms; a 70-ft $d_o$ Para-Sail, reefed to 12.35 percent for 6 seconds; modified radio-command-actuated turn motors, with 37 inches of travel; an 84-ft ringsail emergency parachute (radio commanded); two deHavilland altitude sensors; a blast deflector, jettison mechanisms, and recovery system; landing rockets; the Gemini spacecraft landing gear, retention system, and deployment system; television and video-tape systems; and a gear-stroke switch for parachute disconnect.

### Sequencing

| | |
|---|---|
| Release, sec | 0 |
| R and R canister separation, sec | +5 |
| Attitude change, sec | +25 |
| Activate turn system, sec | +35 |
| Blast-deflector jettison, sec | +40 |
| Altitude-sensor deployment, sec | +70 |
| Landing gear retaining-strap release, sec | +105 |
| Landing gear deployment, sec | +108 |
| Disconnect switch arm, sec | +113 |
| Begin circuitry inspection, sec | +120 |
| Arm rocket motors, sec | +150 |
| Rocket fire | Sensor closure |
| Parachute disconnect | Gear stroke |

## Launch Conditions

| | |
|---|---|
| Altitude, ft . . . . . . . . . . . . . . . . . . . . | 11 500 |
| Velocity, KIAS . . . . . . . . . . . . . . . . . . | 127 |
| Dynamic pressure at R and R canister separation (calculated), $lb/ft^2$ . . . . . . . . . . . . . . . . . | 40 |

## Landing Weight

The landing weight of the test vehicle was 4726 pounds.

## Center-of-Gravity Location

The center of gravity of the test vehicle was Z = 131.95, X = +0.127, and Y = -2.71.

## Results

At R and R canister separation, the vehicle pitched up approximately 100°, fouling the front parachute risers on the nose-gear torque arm. When the vehicle pitched downward, the torque arm and one leg of the front riser failed. The riser failure effectively cut six suspension lines, causing an asymmetrically inflated shape and a 10 deg/sec built-in turn to the left, with a loose skirt. Following attitude change, several attempts were made to trim out the turn, but all were unsuccessful. One deHavilland altitude sensor deployed as programed, but trailed at a slight angle. The other sensor failed to deploy. Since the Para-Sail rate of descent appeared unaffected by the severed riser, the decision was made to retain the Para-Sail for vehicle recovery, rather than to employ the emergency parachute system. The landing gears were not deployed. The rockets did not fire as programed, due to a double malfunction in the altitude sensors, but the rockets fired 3.5 seconds after impact. The nose gear and the television camera received major damage, coupled with slight structural deformation of the test vehicle.

## Failure Analysis

Parachute system. - The pitch oscillation is imparted to the vehicle by the drop method. The drogue parachute did not damp the oscillation, because of its short period of employment (5 seconds) and because it was attached to the R and R canister at two points, allowing a fulcrum about which the vehicle could pitch. The 100° pitchup which occurred just after R and R canister separation allowed the front riser bundle to slip behind the nose-gear torque arm (which protrudes at right angles to the gear body in the stowed position), and thus caused the parachute system malfunction. Three separate modifications were incorporated to prevent this failure mode, as follows: a roll bar was added that prevents riser entanglement with the nose gear, the drogue parachute bridle and the R and R canister were modified to a four-point attachment to provide omnidirectional damping, and drogue parachute employment time was increased to 10 seconds.

DeHavilland altitude sensors. - Separate malfunctions occurred in each of the two deHavilland altitude sensors, as follows:

Sensor 1: Sensor 1 deployed as programed and trailed slightly, bent back at the root. Examination of this sensor indicated that the trail was caused by the root clamp which failed to lock properly. The trail would not have affected rocket firing; however, the inspection circuitry which monitors the altitude sensors, prior to rocket-motor arming, detected a microswitch closure and locked-out rocket firing on that system. Microswitch sensitivity was controlled by placing small silicon rubber blocks between the closure elements. In this sensor head, the manufacturer had substituted blocks of stiffer silicon rubber. Post-test investigation revealed that these blocks took a permanent set while stored in the closed position; hence, the blocks did not prevent microswitch closure resulting from normal vehicle motion.

Sensor 2: Sensor 2 did not deploy as programed, although post-test examination showed that the bolt cutter fired, releasing the lid. The lid was found at the initial impact point, which indicated that the lid was bound at the hinge and did not fall free to release the sensor. After impact, the vehicle rolled three revolutions. During the second roll, the sensor head came free and fired the rockets where the sensor struck the ground during the roll. Testing with this sensor was suspended, and a redesign effort was initiated. The interim sensor was reinstated in the test program.

Turn-control system. - The turn-control system functioned normally. The lack of parachute turn response was due to the cut riser which resulted in a loose skirt and excess fullness. Test data indicated that the turn motors executed reel-in and reel-out as commanded. Post-test examination showed no evidence of cable damage. Special film coverage proved the necessity of the 360° Teflon cable guide fairing.

## TEST 12 - FORT HOOD, TEXAS, JULY 30, 1965

### Objectives

The purpose of this test was to obtain system performance data, and to demonstrate land landing at Gemini design conditions. Drogue parachute employment time was increased to allow the system to accelerate to nominal deployment dynamic pressure at main parachute deployment.

### Onboard Systems

The onboard systems included a 22-ft $d_o$ ringslot drogue parachute, reefed to 19 percent for 12 seconds; R and R canister separation and attitude-change mechanisms; a 70-ft $d_o$ Para-Sail, reefed to 12.35 percent for 6 seconds; modified radio-command-actuated turn motors, with 37 inches of travel; an 84-ft ringsail emergency parachute (radio commanded); an interim altitude sensor; a blast deflector, jettison mechanism, and recovery system; landing rockets; the Gemini spacecraft landing gear, retention system, and deployment system; television and video-tape systems; and a gear-stroke switch for parachute disconnect.

### Sequencing

| | |
|---|---|
| Release, sec | 0 |
| R and R canister separation, sec | +10 |
| Attitude change, sec | +30 |
| Activate turn system, sec | +40 |
| Blast-deflector jettison, sec | +45 |
| Altitude-sensor deployment, sec | +75 |
| Landing gear retaining-strap release, sec | +110 |
| Landing gear deployment, sec | +120 |
| Disconnect switch arm, sec | +128 |
| Begin circuitry inspection, sec | +135 |
| Arm rocket motors, sec | +155 |
| Rocket fire | Sensor closure |
| Parachute disconnect | Gear stroke |

## Launch Conditions

| | |
|---|---|
| Altitude, ft . . . . . . . . . . . . . . . . . . . . . . . | 12 200 |
| Velocity, KIAS . . . . . . . . . . . . . . . . . . . . . | 127 |
| Dynamic pressure at R and R canister separation (calculated), $lb/ft^2$ . . . . . . . . . . . . . . . . . . . | 72 |

## Landing Weight

The landing weight of the test vehicle was 4711 pounds.

## Center-of-Gravity Location

The center of gravity of the test vehicle was $Z = 131.43$, $X = +0.49$, and $Y = -2.43$.

## Results

All systems performed as programed. Immediately following blast-deflector jettison, both left and right turn potentials were verified and furnished a 12 to 15 deg/sec rate of turn. The test vehicle was then maneuvered approximately 2 miles crosscountry to the primary landing zone.

Figure 22 depicts the landing sequence. At the left, the vehicle is in final approach. The tarpaulin in the right center is the center aiming point. In the center, the rockets have ignited and have begun to decelerate the vehicle. At the right, the system has touched down, disconnected the Para-Sail, and is sliding out.

Surface winds were 1 to 2 knots, light and variable. The maximum accelerations recorded were approximately 4.8g. The gear touched down 40 feet from the point of rocket ignition, and the vehicle slid an additional 55 feet. All test objectives were met. This test successfully concluded the program.

## RESULTS

### Parachute Suspension System and Loads

Bridle geometry and steady-state loads. - In the final suspension system (fig. 23), all of the risers were constructed from two-ply, 10 000-pound nylon webbing, type XIX, MIL-W-4088B, condition R. The total riser length was determined by the rear-riser strip-out channel-routing distance, and by the parachute confluence angle. To provide the 13° nose-down flying attitude, it was necessary to shift the force line of the front risers aft, by attaching them to the confluence of a V-bridle. The average loads measured during straight and level flight of the 70-ft $d_o$ Para-Sail were as follows:

| Bridle segment | Total load, percent |
|---|---|
| Front V-bridle leg | 17.9 |
| Rear V-bridle leg | 23.3 |
| Left rear riser | 29.4 |
| Right rear riser | 29.4 |

Opening loads. - Load data were presented for the final (70-ft) Para-Sail configuration only, and were obtained by means of a universal load link at the single attach point.

A composite presentation of opening loads was correlated to a common time scale (fig. 24), wherein line stretch occurred at 5.0 seconds. Deployment was initiated at a dynamic pressure of 40 lb/ft$^2$, as shown by the solid band representing the force-time histories for tests 9 and 10. As shown in the figure, reefed opening shock occurred approximately 2 seconds after line stretch, and reached an average peak value of 10 000 pounds. The system then decelerated to a steady reefed-descent state (1g) an average of 3.5 seconds after the reefed opening shock, and maintained that condition until disreef. Approximately 1.2 seconds after disreef, the parachute reached an average full-open shock value of 16 000 pounds, rebounded slightly, then descended in the fully open, 1g state.

The reefing parameters were developed to balance opening forces when the parachute was deployed at a dynamic pressure of 80 lb/ft$^2$. On test 12, which closely approximated the nominal, or design, deployment case, opening load records were lost due to a faulty strain gage. However, in test 6 of the El Centro parachute test series (reported in detail in Volume II, Section III), the canopy was deployed with an identical payload at a dynamic pressure of 84 lb/ft$^2$. The force-time history from this test with the time correlated to adjust for sequence, presented in figure 24 as a dotted line, indicated that the parachute exhibited a reefed opening shock of 15 600 pounds 2.1 seconds after line stretch, and reached a peak of 15 800 pounds 1.1 seconds after disreef. The parachute reached a 1g condition, representing a descent velocity of approximately 120 ft/sec, in the reefed state in both cases; and disreef was initiated from this point, independent of initial opening velocity, as indicated by examination of the two curves. Consequently, the portion of the load history following disreef should be identical in both cases.

A detailed discussion of the opening loads experienced during the El Centro development tests (Volume II, Section III) indicated that the parachute demonstrated excellent repeatability in opening history. The dotted line in figure 24 not only presents the results of a particular test, but also furnishes an accurate representation of the expected opening forces whenever the parachute is deployed at nominal Gemini conditions.

Attitude change loads. - After the test system had achieved a steady-state descent on the fully opened parachute, the single-point disconnect was activated, allowing the vehicle to tip over to the flying attitude, where the load was assumed by the individual risers. Figure 25 presents a typical force-time history for the separate risers during this maneuver. High-speed camera coverage of attitude change indicated that the vehicle fell in its existing attitude until the rear V-bridle leg and rear risers assumed the load; then it rotated, nose downward, about these points. The leading edge of the vehicle rotated well past the normal position before rebounding. The test vehicle then exhibited a minor oscillation in the pitch plane that damped in 9 to 12 seconds.

The load traces supported the observed results. As shown, the rear V-bridle leg received a peak loading of 2500 pounds, 0.9 second after the single point was released. At the same time, each rear riser assumed a momentary peak load of 2000 pounds, while the forward V-bridle leg was slack. At this point, the vehicle began to rotate. The load in the rear risers dropped momentarily as the system rebounded, then peaked again at approximately 2500 pounds for each riser. The pitch rotation was clearly evident as the load in the rear V-bridle leg went to zero, and the forward leg built to a load peak of almost 4000 pounds as the vehicle pitched nose down.

From this point, the oscillation in the pitch plane was evidenced by the harmonic load variation in the V-bridle legs. The pitch oscillation damped rapidly at a frequency of approximately 1 cps. Twelve seconds after initiation of the maneuver, the riser loads had stabilized.

A vertical axis g-time history for the attitude-change maneuver (fig. 26) showed that the peak acceleration of 1.87g occurred 1.2 seconds after the single-point release. An analysis of the high-speed film coverage indicated that this was the point of maximum travel when the vehicle had pitched nose down past the normal flying attitude.

The nonrigid parachute system, by which the test vehicle was suspended, facilitated the low g-loading during this maneuver. A static test of this same attitude repositioning, with the vehicle rigidly suspended, produced accelerations of approximately 4g. The elastic properties of the nylon suspension system, which contributed to the reduction in peak g, also created the short duration pitch oscillation (figs. 25 and 26).

Parachute contribution during rocket fire. - In analytically determining the desired thrust-time characteristics for rocket-motor development, the vertical decelerating force exerted by the parachute was assumed to be proportional to the square of the instantaneous descent velocity of the vehicle. These calculations began with the basic steady-state drag equation

$$F = W = C_D \rho/2 \, V^2 S \tag{1}$$

and assumed that $C_D \rho/2 \, S$ remained constant; therefore

$$F = kV^2 \tag{2}$$

at any time during rocket fire. This assumption did not account for velocity difference between the parachute and the vehicle, due to elasticity in the suspension system, or for change in $C_D S$ due to velocity change.

The parachute riser load was calculated by equation (2), using instantaneous velocities obtained by iterative integration of the rocket accelerations and the parachute vertical retardation force. These data are presented as a dotted line in figure 27. The area under this curve represents the impulse in pound-seconds contributed by the parachute and is sufficient to reduce descent velocity approximately 4 ft/sec. This amounts to approximately 18 percent of the total velocity reduction and, as such, is highly significant in thrust-time determination.

To determine the actual variation of parachute loading, riser loads were measured during rocket fire during full-scale drop tests. These data, also presented in figure 27, illustrated that the riser load dropped off quickly during high thrust, bottomed out at the end of high thrust, then increased again as the vehicle slowly accelerated during the sustained low-thrust portion of rocket fire. At impact, the parachute was disconnected, and the load went to zero. It should be noted that the analytically determined curve closely approximates the measured test data once the residual load, due to suspension system elasticity, has deteriorated.

## Stability

Steady state. - In the parachute development tests at El Centro, efforts were made to measure canopy oscillation during steady-state descent. These oscillations were extremely low, and accurate measurement was impossible. It was determined that total canopy oscillatory travel was less than $\pm 3^\circ$ in any axis, but no specific value was discerned.

Vehicle pitch, roll, and yaw rates were measured during the system test program, and were then integrated to determine vehicle oscillatory travel. Typical steady-state rate data are presented in figure 28. The vehicle exhibited peak roll rates on the order of $\pm$ 2 to 3 deg/sec at a frequency of 1 cycle every 5 to 6 seconds. Integration of this curve showed the average oscillatory travel in roll to be $\pm 5.85^\circ$. The peak vehicle pitch rates were on the order of $\pm$ 1 to 2 deg/sec, at a frequency of 1 cycle every 3 to 4 seconds. Integration of this curve showed the average oscillatory travel in roll to be $\pm 2.75^\circ$. In yaw, the peak vehicle rates were on the order of $\pm$ 1 to 2 deg/sec, with random cycling. Integration of this curve showed the average oscillatory travel in yaw to be $\pm 2.35^\circ$.

An examination of these data indicated that all vehicle oscillations were quite low both in rate and in travel, with roll oscillation being larger. A corresponding analysis of test film coverage indicated oscillatory travel values of less than $\pm 5^\circ$ in all three axes, with pitch oscillation being predominant. Although there was an apparent discrepancy in these two separate data sources as to whether pitch or roll was relatively greater, the important fact was that both sources independently verified the high degree of vehicle stability.

Attitude during turn. - As predicted by the 1/3-scaled model, the full-scale vehicle banked out and pitched nose down during a turn maneuver, both actions being a function of turn rate. The banking phenomenon occurred because the forward inertia of the vehicle tended to keep it traveling in a straight line while the canopy rotated away from it. The change in pitch attitude during turn was caused by the strong pitch-down characteristic of the parachute.

The variation of bank angle with turn rate (fig. 29) indicated that the vehicle began to bank as soon as any turn was present, and reached an angle of 18° at a turn rate of 20 deg/sec. The effect of turn rates on pitch attitude (fig. 30) indicated that pitch attitude changed when any turn was present and reached an angle of approximately 28° at a turn rate of 20 deg/sec.

The separated riser suspension system (fig. 23) provided a strong couple, such that rotation of the canopy induced the same rotational travel in the vehicle. During the acceleration up to a constant rate of turn, the vehicle lagged the parachute some 5 to 10° in yaw. Once a constant turn rate had been established, there was no yaw variation between the parachute and the vehicle. When the turn line was released, the parachute immediately ceased to turn, and the vehicle/parachute combination returned to nominal attitude in 2 to 3 seconds with no oscillation.

## Turn System

Required turn performance. - Analytical studies, conducted prior to testing, indicated that a maximum turn rate of at least 10 deg/sec was required for adequate maneuvering. No attempt was made to determine an upper limit analytically. During the test programs, turn rates up to 75 deg/sec were obtained on the 1/3-scaled-model system and up to 50 deg/sec on the full-scale system. In both cases, the maximum rate attainable was in excess of the optimum operating range.

The objective of the turn-performance investigation was the determination of the range of turn rates which allowed full utilization of system capabilities. To establish end points, the minimum usable rate was defined as the lowest attainable rate which still allowed the system to be maneuvered into a preselected landing area and alined with the wind. The maximum usable rate was defined as the highest rate of turn at which the system's direction could be accurately changed to a new heading.

Determination of the minimum usable rate was accomplished by flight tests of the 1/3-scaled system, with turn-line travel restricted to furnish maximum turn rates of 10, 12, and 16 deg/sec. In these tests, the 10 deg/sec rate was marginally acceptable. The maximum usable rate was found to be somewhat a function of turn-motor reel-out time, since a fast reel-out aided the flight controller to aline accurately on the desired headings.

In the 1/3-scaled-model program, reel-out was accomplished almost instantaneously, causing the parachute, when commanded, to stop rotating. The governing factor in determining maximum usable rate then became the ability of the controller to recognize the correct heading, and to transmit a signal. The program results indicated that an experienced controller could handle rates up to 45 deg/sec, with an optimum somewhere in the 25 to 35 deg/sec range.

In the full-scale program, reel-out was accomplished by releasing the motor brake and allowing the force in the turn line to pull the cable out. This action required from 3 to 6 seconds, depending upon line load, and resulted in additional turn travel after the reel-out signal was transmitted. With this type of system, the controller could handle rates up to 25 deg/sec, with an optimum in the 18 to 20 deg/sec range.

It was concluded from these results that the desired operational turn-rate range for the land landing system was from 10 to 25 deg/sec, with an optimum value around 18 deg/sec, and that motor reel-out time should be held to a minimum to allow for accurate heading alinement.

Line arrangement. - The final turn-line arrangement consisted of a nylon webbing attached to the turn-motor cable and extending upward into the parachute, dividing into four branches below the skirt. Each of the branches passed through reefing rings on the turn slots and was sewed to the radial seam at the vent band. When the turn line was shortened, these branches formed cords inside the canopy, which pulled the turn slots toward the canopy center.

Turn-line length. - The determination of the correct turn-line length was one of the most difficult technical problems encountered during the development effort. To provide a fixed reference point, a turn zero was defined as the point on the turn lines corresponding to the junction of the suspension lines and the risers when the turn slots were fully open, and any shortening of the turn line would begin to close them. Below this point, the precise length was determined by suspending the vehicle in the flying attitude and measuring the distance.

Before the full-scale test program began, relative elongations of the parachute suspension lines and turn lines were calculated, and an analytical turn zero was determined. Next, the 70-ft Para-Sail was ground inflated in a 20 to 25 ft/sec wind and turn zero was marked. Both of these efforts indicated a line length of approximately 98.5 feet, and the 70-ft Para-Sail tests were initiated with this setting. These techniques proved to be ineffective since they did not account for the added elongation resulting from opening shock, a part of which remained in the suspension lines until all load was removed. Turn zero was then determined by increasing turn-line length by increments on succeeding tests, and measuring the line force. Figure 31 presents these data, and establishes turn zero at approximately 101 feet 4 inches.

In addition to the determination of the correct turn zero, several other pertinent facts regarding line length were ascertained from the development program, as follows:

1. It was impossible to achieve zero load in a turn line, since air drag on the line accounted for 5 to 10 pounds of force. It was also found that any length turn line, if placed on one side of the canopy only, caused the canopy to rotate, due to the weight of the line and air drag.

2. Every parachute tested exhibited a built-in turn due to minor fabrication asymmetries or to variation in true turn zero between the two sides of the canopy. During the system test program, it was necessary to trim out this built-in turn potential by shortening the opposing turn line. In most cases, the built-in rate was 10 deg/sec or less, and required only a few inches of travel on the opposite side to trim it out. In the worst instance, the canopy exhibited a built-in rate of approximately 25 deg/sec and required 17 inches of travel to trim out. It should be noted that turn zero was shorter on this test than the finally derived value.

3. The way in which the turn lines were measured to set the length was critical, due to the large amount of elongation under very low force. The practice followed was to place the parachute on the rigging table with the suspension lines under 20 pounds of tension, and to measure the turn lines under 10 pounds of tension.

Rate of turn. - In presenting the rate of turn as a function of line force, the attempt was made to correct for individual parachute idiosyncrasies resulting in built-in turn, and for incorrect settings on the opposite turn line which retarded turn rate. These data (fig. 32) did not reflect the performance of any given canopy during the test program, but presented the average rate of turn with factors other than turn-line force removed. The method whereby these data were corrected was to equate built-in turns, resulting from canopy asymmetry and foreshortened opposing turn lines, to a turning moment. Since the built-in turn rates were known, the extraneous moment and the built-in turn could be subtracted out. The residual moment was then expressed as a turn-line force furnishing the remaining turn rate.

Travel time. - A typical variation of turn-line position and load was presented as a function of time in reel-in and reel-out (fig. 33). In reel-in, the curve showed an initial load of 25 pounds, which increased to 110 pounds (motor stall) in approximately 1 second. During this time interval, the turn-line position plot showed a foreshortening of zero from 42 inches to approximately 6 inches. In reel-out, the load decreased to the 25-pound residual load in approximately 4 seconds. The position plot showed the extension of turn zero from the reel-in position back to full out.

In a turn system in which trim was accomplished by the primary turn motors, either reel-in time or reel-out time must be paced to allow for trim adjustment. In the development program, trim was accomplished during reel-out since this speed was approximately 1 ft/sec, as opposed to the reel-in time of approximately 3.5 ft/sec. This method was not specifically designed, but resulted from the operating characteristics of the turn motors. The 1 ft/sec reel-out speed was adequate to allow trim.

## Landing Dynamics

At the initiation of the landing sequence, the vehicle was nominally suspended in a -13° pitch, 0° roll, 0° yaw attitude, and was descending at approximately 29 ft/sec with a horizontal velocity of from 0 to 30 ft/sec. When the main landing gears were approximately 8.7 feet above the surface, the altitude sensor ignited the rocket motors. (Figure 34 presents a nominal rocket-system acceleration history.) The combined motors furnished a peak acceleration of 3.60g, an average high-thrust acceleration level of 2.65g for 0.35 second, and a sustained low-thrust acceleration level of approximately 0.5g for an additional second.

Figure 35 shows the approximate effect of the rocket system on descent velocity. Descent velocity decays in an essentially linear manner during the high-thrust phase, then increases again during the sustained-thrust phase. This velocity variation, discussed in detail in Volume II, Section IV, is correctly expressed as a family of curves which accounts for variation in initial rates of descent, temperature, motor performance, and so forth, just as true rocket-system acceleration history must be expressed as a family rather than a single trace.

The two individual curves chosen approximated the conditions of the majority of the drop tests, and were used in the following discussion of impact accelerations. Rocket ignition height was established to insure that gear touchdown occurred at some point during the sustained thrust level, while the velocity was within the landing-gear envelope. The nominal setting approximated the midpoint of this range to allow the widest possible variation in initial conditions.

The gear touchdown plane was at an angle of -18.3° to the vehicle centerline. This allowed the main landing gear to absorb the initial impact before the vehicle rotated the additional 5.3° and the nose gear touched down. This method provided the greatest margin of impact stability for the Gemini spacecraft in the same manner as for conventional aircraft. Following touchdown, horizontal velocity was dissipated in slideout.

In a nominal land landing, the vehicle should experience an initial acceleration history approximating that furnished by the rocket system, coupled with the initial 1g felt by the crew during parachute descent. During sustained thrust, the main gear should touch down, furnishing an initial peak acceleration of the same magnitude as the peak rocket acceleration, then exhibiting an acceleration history that follows the main gear force-stroke characteristics (Volume II, Section IX). Shortly thereafter, the nose gear should touch down, furnishing a second impact peak slightly lower than the initial impact, since the main gear will have absorbed part of the impact energy. If any horizontal velocity is present, the vehicle should slide forward on all three gears, registering a constant acceleration of about 0.25g along the Z-axis. This value corresponds to a surface friction coefficient in the 0.4 to 0.5 range. The 1/3-scaled landing dynamics studies predicted slideout distances of 0 to 60 feet at the end points of the horizontal velocity range.

In the event the vehicle lands in water, the acceleration history prior to impact should be the same as the land landing case, and an initial impact acceleration slightly lower than the rocket-system peak should occur as the bottom of the spacecraft impacts on the water surface. Some minor rebound accelerations would also occur as the vehicle sought a flotation attitude.

Land landing accelerations. - The accelerations for test 12, in which the vehicle had an initial forward velocity of approximately 35 ft/sec, and for the second crane drop, in which the vehicle had no horizontal velocity, were presented to provide a broad base data description of land landing.

Impact accelerations were measured by X-, Y-, and Z-axis accelerometers mounted at the vehicle center of gravity. Rocket accelerations were measured by an additional accelerometer mounted on the thrust-line axis, and also located at the vehicle center of gravity. Rocket performance was further monitored by recording chamber pressure during firing.

Figure 36 illustrates the thrust-line accelerations recorded during test 12 and only one of the rocket chamber-pressure traces, since both the left and the right motor pressures were identical.

The corresponding X-, Y-, and Z-axis accelerations were presented as curves on a time scale with zero arbitrarily set just prior to altitude-sensor signal (fig. 37). During this test, the vehicle had just completed a left turn and was in a 10° left roll at rocket ignition.

As shown by the chamber pressure, peak high thrust occurred at approximately 0.075 second, and the rocket motors provided an essentially nominal thrust-time history. (The correlation of chamber pressure and thrust is discussed in Volume II, Section IV.) The thrust-line and Y-axis (13° from thrust line) accelerometers also showed peak high thrust at this time, recording 4.68g and 4.67g, respectively. These values corresponded to an initial 1g parachute descent condition and a nominal 3.60g peak rocket acceleration. After the initial peak, the thrust-line and the Y-axis accelerations repeated the rocket output indicated by the chamber-pressure trace, reaching the sustained thrust level at 0.5 second and maintaining it until touchdown. During rocket fire, the vehicle remained in the 10° roll attitude. Consequently, the left main gear touched down first at 0.70 second, recording accelerations of 1.76g on the thrust line, 1.9g on the Y-axis, 0.6g on the X-axis, and 1.2g on the Z-axis. At 0.8 second, the right main gear touched down as the gear system corrected the rolled condition. The nose gear touched down at approximately 1.0 second, registering accelerations of 1.4g on the thrust line, 1.9g on the Y-axis, and 1.08g on the Z-axis.

The time period from 1.0 to 4.0 seconds contained the accelerations that occurred during slideout. As shown, the Y-axis accelerometer recorded an expected 1g average, and the Z-axis accelerometer recorded an average of 0.22g, indicating the correct surface friction coefficient of 0.45. The total slideout distance was 55 feet.

An examination of figures 36 and 37 showed that all accelerations resulting from the land landing in test 12 were successfully controlled below a level of 5g, and that the vehicle maintained stability throughout the landing.

The thrust-line accelerations were recorded during the second crane drop in which the vehicle landed on a sod surface (fig. 38). An arbitrary zero time was selected 0.2 second prior to release of the vehicle from the overhead crane. Since no horizontal velocity or roll conditions were present, the X- and Z-axis accelerations were essentially zero and were not included in the figure. The Y-axis trace generally repeated the thrust-line accelerations shown.

Since the vehicle was allowed to free fall to achieve parachute descent velocity at altitude-sensor contact, the previously seen 1g initial descent condition was absent. Consequently, 1g must be added to the initial acceleration values shown when these results are compared with parachute drop test results. The accelerations, beginning with the sustained thrust level, were directly comparable.

As shown in figure 38, the vehicle free fell at a 0g condition until rocket ignition, and received a peak thrust acceleration of 3.89g. The acceleration trace then followed the nominal rocket-system acceleration history shown in figure 34. Initial impact occurred as programed at the approximate midpoint of the low thrust level. The main gear touched down at 1.90 seconds, and the initial impact peak was recorded at 3.9g. The nose gear touched down at 2.20 seconds, and a secondary peak of 1.7g was recorded.

In the crane drop, as in test 12, all landing accelerations were controlled below a level of 5g.

Unstable land landings. - Due to a control system failure during test 10, the vehicle attitude at touchdown considerably exceeded the stable envelope. As predicted by the 1/3-scaled landing dynamics study, the vehicle tumbled on its left side and rolled over. While this test was considered unsuccessful from the standpoint of landing stability, it provided valuable acceleration data by which an unstable landing could be evaluated and the possible effect on a flight crew could be determined.

The accelerations in the X-, Y-, and Z-axis plus the angular roll rate were recorded (fig. 39). The vehicle attitude, at significant times during the landing, was determined from high-speed film coverage and was included in the figure.

Zero time was arbitrarily set just prior to rocket fire. At this point, the vehicle was pitched down 25°, rolled left 22°, and in a left turn which represented a yaw of approximately 45°. The resultant acceleration, due to the 1g descent and built-in turn, was 1.12g.

At peak rocket thrust (0.07 second) the accelerometers recorded 0.9g in the X-axis, 4.25g in the Y-axis, and 1.0g in the Z-axis. The roll attitude derived from the film analysis at this point was 22°, whereas the X- and Y-acceleration resultant indicated it to be 26°.

Initial impact occurred when the left gear touched down. Film analysis showed that the roll angle was 36°, while the X- and Y-acceleration resultant indicated it to be 39.5°. The peak resultant acceleration occurred at this point and was 8.9g, which correlated closely with the 8g peak predicted by the 1/3-scaled landing dynamics study. The vehicle continued to rotate about the left main gear and received a second impact of 1.4g as the side of the vehicle struck the ground.

At approximately 1.5 seconds, the accelerometers recorded a high spike which was discounted as telemetry signal interference when film analysis did not indicate any vehicle motion of a nature to explain the possible high accelerations; but the film did show that the high spike occurred at the precise instant the vehicle rolled over on the telemetry antenna. The vehicle continued to roll until the right main gear contacted the ground at approximately 2.15 seconds.

While the maximum acceleration of 8.9g was approximately twice that expected during a stable land landing, it compared favorably to the 5g to 16g range expected with the existing Gemini spacecraft water landing system, and was well within acceptable tolerance levels.

Water landing. - During tests wherein the vehicle landed in water, impact dynamics data were collected to evaluate system performance. When the rockets were incorporated, the ignition height was set at that established for land landing to demonstrate the retention of water landing capability, and to obtain a realistic data picture of the contingency water landing case.

The representative water-impact acceleration histories with and without the landing rockets are presented in figure 40. The accelerations presented were those in the vertical, or Y-axis, since these represented the major accelerations experienced. It should be noted that the accelerometers mounted in the vertical axis read 1g during steady-state descent, and all impact accelerations were recorded from that point. The nonrocket test presented did include an altitude sensor which fired flashbulbs. Consequently, it was possible to correlate both tests to a common sensor impact time.

For the test without rockets, the vehicle continued to descend at 1g until impact (fig. 40). A peak acceleration total of 3.4g occurred 0.41 second after altitude-sensor signal. This corresponded to a descent velocity of 26 ft/sec. The portion of the acceleration history following impact and shown at 1g was lost due to immersion of the telemetry antenna in salt water. A study of high-speed film coverage of this test indicated a rebound existed as the vehicle returned to its flotation attitude. Analysis of similar tests indicated rebound accelerations were below those experienced at impact. Following this test, the telemetry antennas were moved to the top of the test vehicle.

For the test which included rockets, a peak acceleration of 4.6g occurred 0.03 second after altitude-sensor signal. This acceleration value corresponded closely to the initial 1g state and the nominal 3.60g peak high thrust furnished by the rockets. The accelerations then followed the rocket thrust-time history, shown in figure 34, until impact. The peak impact acceleration recorded was approximately 1.8g. As predicted, this occurred just prior to rocket burnout, rather than near the center of the sustained thrust level where deployed landing gear would have impacted. From a comparison of figures 34 and 40, the accelerations experienced at impact were shown to be considerably lower when the rockets were employed.

As shown in the weight summary, the mechanical landing gears comprise the major portion of the weight. These gears do not necessarily represent the most efficient design, but were incorporated without change.

The total weight shown is approximately 444 pounds heavier than the parachute system currently used to land the Gemini spacecraft in the ocean. This weight difference can be reduced by the partial elimination of the various flotation and location aids required to support water landing and recovery.

Manned Spacecraft Center
National Aeronautics and Space Administration
Houston, Texas, September 23, 1966
904-02-15-01-72

## REFERENCE

1. McDonnell Aircraft Report 052-052.32.01. Nov. 12, 1963, p. 133.

TABLE I. - INITIAL DEPLOYMENT TESTS FOR A 24-FOOT-DIAMETER PARA-SAIL

[September 1962 - October 1962]

| Number of tests | Objectives | Launch conditions | | Vehicle weight, lb | Results |
|---|---|---|---|---|---|
| | | Altitude, ft | Velocity, knots | | |
| 9 | Determine the deployment characteristics. | 600 to 5000 | 0 to 120 | 125 | The 24-ft $d_o$ Para-Sail can be packed and deployed at velocities up to 120 knots when certain modifications are made to the risers. Opening shock values are high. Can be skirt reefed. |
| 8 | Determine method and rate of turn including dynamic behavior. | 600 to 5000 | 0 to 75 | 125 | Turn can be achieved by riser foreshortening and by closing vents on the canopy. Maximum rate achieved was 29.3 deg/sec. The parachute system is extremely stable. |

TABLE II. - EIGHTY-FOOT PARA-SAIL TESTS, HOUSTON, TEXAS

| Test | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|
| | | Altitude, ft | Velocity, knots | | |
| 1 | 2500 | 2000 | 110 | First attempt to deploy a Para-Sail larger than 24 ft in diameter. Determine preliminary deployment characteristics. 7.9 percent reefing. | Front of canopy tucked in at inflation and stayed in for 19 sec. It then came out for a fully inflated descent. The skirt was malformed and essentially closed, with front of canopy near rear exhaust slots. At disreef, the rear inflated first and drove over the front of the canopy. |
| 2 | 2500 | 2600 | 110 | Determine effect of increased reefing line length on deployment characteristics. 14.2 percent reefing. | Front half of canopy folded back against the centerline and rear of canopy. At disreef, front tucked all the way to the rear and inflated inverted through two side suspension lines. |

TABLE II. - EIGHTY-FOOT PARA-SAIL TESTS, HOUSTON, TEXAS - Continued

| Test | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|
| | | Altitude, ft | Velocity, knots | | |
| 3 | 2500 | 2600 | 125 | Determine the deployment characteristics of slotted front canopy with seven solid gores. | Canopy inflated slowly in the reefed state, with the rear fully inflated and the front folded against the center-line and the rear. At dis-reef, the right suspension line tucked through adja-cent suspension lines and canopy inflated inverted. |
| 4 | 2500 | 2600 | 125 | Determine the deployment characteristics with the pilot parachute perma-nently attached. | Front tucked in at inflation and came out slowly. Pilot parachute bridle seemed to decrease the severity of front tuck-in. |
| 5 | 2500 | 2600 | 125 | Evaluate deployment charac-teristics to determine if the increased bridle length to the pilot parachute would decrease crown pinching of the Para-Sail. | Front of canopy tucked in at initial inflation, and the skirt panel of the front gore tucked through two suspension lines in the rear of the canopy and held for the entire descent. No canopy damage. |

TABLE II. - EIGHTY-FOOT PARA-SAIL TESTS, HOUSTON, TEXAS - Continued

| Test | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|
| | | Altitude, ft | Velocity, knots | | |
| 6 | 2500 | 2600 | 110 | Determine deployment characteristics of a solid front canopy with a semi-elliptical cutout in the front skirt. | Front of the canopy was tucked in during initial inflation, indicating air flow was through the skirt and out the rear. The front of the canopy indicated an improved tendency to inflate to the normal shape. |
| 7 | 3600 | 2600 | 110 | Determine deployment characteristics with a solid front, semielliptical cut-out skirt and the 21 rear gores zero reefed at the skirt. | The front tucked back during initial inflation. Rear 2000-lb reefing line failed during reefed state, and blew one rear gore and seven rear exhaust panels. |
| 8 | 3600 | 2600 | 110 | Evaluate the deployment characteristics with the rear risers foreshortened 4 ft 3 in., and pocket bands added to the front 21 gores. | The front center gore and two rear gores blew during inflation, including the skirt band. The front came out slowly. Severe canopy damage. |

TABLE II. - EIGHTY-FOOT PARA-SAIL TESTS, HOUSTON, TEXAS - Continued

| Test | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|
| | | Altitude, ft | Velocity, knots | | |
| 9 | 2500 | 2600 | 110 | Evaluate the deployment characteristics with a 10-ft-diameter guide surface internal parachute added to the system. | The canopy inflated with the front crown tucked in. The skirt had a reasonably good shape. The reefing line failed, and the canopy opened rapidly. Failure was caused by an uneven skirt. This test marked a significant improvement in deployment characteristics. |
| 10 | 2500 | 2600 | 110 | Investigate further the effect of the internal parachute as an inflation aid with the rear risers pulled down 4 ft 3 in. to effect an even skirt during inflation. | Canopy inflated in the reefed state with a large reefed airball, front crown tuck-in, and a good skirt shape. Closing rear slots did not eliminate front crown tuck-in. |

TABLE II. - EIGHTY-FOOT PARA-SAIL TESTS, HOUSTON, TEXAS - Concluded

| Test | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|
| | | Altitude, ft | Velocity, knots | | |
| 11 | 3500 | 2400 | 110 | This test was to evaluate deployment quantitatively. The pilot parachute confluence was removed to prevent front crown tuck-in. | Reefed airball was very large. Reefed opening shock, 11 200 lb. Disreef was rapid and even. Disreef opening shock, 8900 lb. Pilot parachute with no confluence was considered ineffective. The decision was made to begin El Centro drop-test series. |

TABLE III. - EIGHTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA[a]

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions: Altitude, ft | Launch conditions: Dynamic pressure, q | Purpose | Results |
|---|---|---|---|---|---|---|
| 1 | 80 L-2 | 3600 | 5000 | 120 knots | Determine deployment characteristics. | Good deployment. Full inflation, 10.8 sec. |
| 2 | 80 L-3 | 3600 | 5350 | 40/120 knots | Determine deployment and performance characteristics. | Good deployment. Full inflation, 9.2 sec. L/D = 1.15 at 19 ft/sec R/D. |
| 3 | 80 L-3 | 4823 | 5325 | 41/120 knots | Determine deployment and performance characteristics with increased payload. | Front crown tucked in. Good skirt shape. Rear riser failed at release point. Severe canopy damage. |
| 4 | 80 L-4 | 4823 | 5300 | 45/120 knots | Determine deployment and performance characteristics with increased payload, and 12.37 percent reefing. | Deployment bag failed pilot parachute bridle, which rebounded into the main vent and released the centerline. Telemetry, cameras, and parachute destroyed. |

[a]Initial configuration: Pocket bands added to front 21 gores; 10-ft-diameter internal parachute; 36-line bridle to 8-ft guide surface vent parachute; and 14.5 percent reefing.

TABLE III. - EIGHTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA[a] - Continued

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Dynamic pressure, q | | |
| 5 | 80 A-1 | 2605 | 5300 | 45/120 knots | Determine deployment characteristics with crown area replaced with flat circular section, centerline removed, vent diameter increased to 5 ft. | Slow inflation, with looseness in crown. L/D = 1.2 at 15.5 ft/sec R/D. |
| 6 | 80 A-1 | 2605 | 5300 | 34.5/120 knots | Determine deployment characteristics with increased reefing line length. 13 percent reefing. | Slow inflation. Fold in lateral scoop area during reefed inflation. Reefed opening shock, 6600 lb. Disreef opening shock, 7600 lb. |
| 7 | 80 A-2 | 2650 | 5200 | 43.5/120 knots | Determine deployment characteristics without an internal parachute. | Canopy streamed for 3 sec, only partially inflated at disreef, with a poor skirt shape. Indicates necessity for an internal parachute. |

[a]Initial configuration: Pocket bands added to front 21 gores; 10-ft-diameter internal parachute; 36-line bridle to 8-ft guide surface vent parachute; and 14.5 percent reefing.

TABLE III. - EIGHTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA[a] - Continued

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Dynamic pressure, q | | |
| 8 | 80 A-1 | 2605 | 11 100 | 39/120 knots | Evaluate deployment characteristics at design altitude. | Canopy inflated symmetrically, and held a large reefed air-ball. Excellent disreef, steady skirt. |
| 9 | 80 A-1 | 3700 | 10 600 | 120 knots | Determine effect on deployment characteristics caused by increased payload weight at design altitude. | Breakcord on left stabilization panel failed, allowing it to inflate prematurely, causing canopy entanglement and, eventually, suspension line failures. Cause; improper rigging. |
| 10 | 80 A-3 | 3700 | 11 000 | 31.5/120 knots | Determine effect on deployment characteristics of an increased payload weight at design altitude. | Canopy inflated symmetrically, and held a large reefed air-ball. Reefed opening shock, 10 580 lb; disreef opening shock, 8600 lb. Excellent disreef. |

[a]Initial configuration: Pocket bands added to front 21 gores; 10-ft-diameter internal parachute; 36-line bridle to 8-ft guide surface vent parachute; and 14.5 percent reefing.

TABLE III. - EIGHTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA[a] - Continued

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Dynamic pressure, q | | |
| 11 | 80 A-3 | 4750 | 10 600 | 37/120 knots | Determine deployment characteristics at design weight and drop altitude. | Stabilization panel breakcord failed, which caused premature inflation and pulled a large portion of the canopy through two suspension lines. The canopy blew at disreef. |
| 12 | 80 A-4 | 4755 | 10 600 | 46/120 knots | Determine deployment characteristics at design weight and altitude, with hesitation bags on the stabilization panels. | Rapid and symmetrical inflation. Reefed opening shock, 14 100 lb. Disreef opening shock, 13 800 lb. No canopy damage. Hesitation bags were ineffective. |
| 13 | 80 A-5 | 2785 | 10 600 | 49/120 knots | Determine deployment and steady-state characteristics with stabilization panels removed, low weight, at design altitude. | Canopy inflated rapidly. Rapid disreef with evident loss of steady-state stability. Rate of descent was higher by 12 percent than similar test with stabilization panels. |

[a]Initial configuration: Pocket bands added to front 21 gores; 10-ft-diameter internal parachute; 36-line bridle to 8-ft guide surface vent parachute; and 14.5 percent reefing.

TABLE III. - EIGHTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA[a] - Continued

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Dynamic pressure, q | | |
| 14 | 80 A-6 | 4755 | 10 300 | 73/120 knots | Determine effect of zero reefing stabilization panels independent of skirt. Test conducted at design altitude and weight. | Canopy damaged during reefed inflation, caused by improper suspension of line 37. Two gores split in the front pressure area. |
| 15 | 80 A-7 | 4755 | 11 000 | 65/137 knots | Same as previous test. Remove irregular suspension line from previous drop. | Reefing line failed before disreef. Reefed opening shock, 16 100 lb. Disreef opening shock, 16 600 lb. Damage limited to one split gore in the pressure area. |

[a]Initial configuration: Pocket bands added to front 21 gores; 10-ft-diameter internal parachute; 36-line bridle to 8-ft guide surface vent parachute; and 14.5 percent reefing.

TABLE III. - EIGHTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA[a] - Continued

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Dynamic pressure, q | | |
| 16 | 80 A-8 | 4755 | 10 700 | 71/137 knots | Determine effect of zero reefing stabilization panels independent of skirt. Replace reefing line with a 1500-lb tubular line. Reinforcing tapes added to the crown area. | Canopy inflated to a large reefed airball. Reefed opening shock, 15 000 lb. Disreef opening shock, 11 000 lb. Two panels blew in the pressure area during reefed inflation. Burn damage caused by canopy ties and internal parachute riser not being sleeved. |
| 17 | 80 A-9 | 4750 | 10 600 | 80 | Object of this test was to increase the deployment dynamic pressure to 80 lb/ft$^2$ at design weight and altitude. 1500-lb braided nylon reefing line. 12.4 percent reefing. | Canopy inflated slightly smaller than seen in earlier tests. Rapid and even disreef. Limited damage, two blown panels, and minor burns. |

[a]Initial configuration: Pocket bands added to front 21 gores; 10-ft-diameter internal parachute; 36-line bridle to 8-ft guide surface vent parachute; and 14.5 percent reefing.

TABLE III. - EIGHTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA[a] - Concluded

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Dynamic pressure, q | | |
| 18 | 80 A-9 | 4750 | 10 600 | 31.5 | Test was conducted to determine the rate of turn of the canopy with one row of turn slots closed. | Reefed opening shock, 11 500 lb. Disreef opening shock, 13 700 lb. Turn rate, 12 deg/sec. |
| 19 | 80 A-9 | 4750 | 10 600 | 80 | Test was conducted to determine performance of canopy when deployed at design altitude, dynamic pressure, and weight. | Good reefed shape with rapid disreef. Reefed opening shock, 14 400 lb. Disreef opening shock, 12 600 lb. Damage limited to one blown number 10 panel and several small burns. |
| 20 | 80 A-9 | 4750 | 10 600 | 50 | Test was conducted to determine the rate of turn with both rows of turn slots closed. | Reefed opening shock, 12 000 lb. Disreef opening shock, 14 750 lb. Turn rate, 19 deg/sec. Damage limited to light burns in the crown area. |

[a]Initial configuration: Pocket bands added to front 21 gores; 10-ft-diameter internal parachute; 36-line bridle to 8-ft guide surface vent parachute; and 14.5 percent reefing.

TABLE IV. - SEVENTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Deployment, knots | | |
| 1 | 70-1 | 4750 | 10 600 | 140 | To study the inflation and gliding characteristics of the advanced version. | Deployment and inflation processes appeared very orderly. Indication of tuck in the right front of canopy. The canopy exhibited a continuous turn to its left at 9 deg/sec. |
| 2 | 70 A-4 | 4738 | 10 600 | 154 | To study inflation and gliding characteristics at design conditions. | Reefed opening shock, 16 250 lb; disreef opening shock, 16 000 lb; R/D, 30.1 ft/sec; and L/D, 1.05. Very stable descent. Light seam strain in 2.25-oz nylon ripstop material. |

TABLE IV. - SEVENTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA - Continued

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Deployment, knots | | |
| 3 | 70 A-4 | 4738 | 10 600 | 105 | To study turn characteristics. Turn line was shortened 78 in. | R/T, 50 deg/sec; and R/D, 43.1 ft/sec; caused by banking during turn. Two vent lines were damaged, and several small burns were sustained. |
| 4 | 70 A-4 | 4750 | 10 600 | 105 | To determine what reduction in L/D could be effected with both turn lines shortened 78 in. | During steady-state descent the canopy was very stable. L/D, 0.9 to 1.0, and R/D, 27 ft/sec. Three vent lines were damaged. |
| 5 | 70 A-5 | 4747 | 10 600 | 154 | To study the inflation and glide characteristics of the 70-ft Para-Sail with the taffeta material replaced by a similar weight ripstop material. 80 q. | Reefed opening shock, 15 600 lb; disreef opening shock, 15 800 lb; R/D, 25.4 ft/sec; and L/D, 1.2. Very stable descent. |

TABLE IV. - SEVENTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA - Continued

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Deployment, knots | | |
| 6 | 70 A-5 | 4747 | 10 600 | 167 | To study the deployment and inflation characteristics at a dynamic pressure of 95 lb/ft$^2$. | Very stable descent. Reefed opening shock, 17 500 lb; disreef opening shock, 15 500 lb; R/D, 26.3 ft/sec; and L/D, 1.1. Air damage was negligible. |
| 7 | 70 A-5 | 4747 | 10 600 | 181 | To study the deployment and inflation characteristics at a dynamic pressure of 110 lb/ft$^2$ | Very good deployment. Reefed opening shock, 22 000 lb; disreef opening shock, 17 250 lb; R/D, 23.7 ft/sec. One blown lateral scoop; no other visible damage. |
| 8 | 70 A-5 | 4760 | 10 600 | 192 | Ultimate strength test. | Reefed opening shock, 22 000 lb; disreef opening shock, 16 600 lb; R/D, 26.1 ft/sec. No damage. |

TABLE IV. - SEVENTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA - Continued

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Deployment, knots | | |
| 9 | 70 A-5 | 4760 | 10 600 | 190 | Ultimate strength test. | Some burns and seam strains. Reefed opening shock, 20 800 lb; disreef opening shock, 16 800 lb; and R/D, 26.9 ft/sec. Canopy was considered to have passed this ultimate strength test. |
| 10 | 70 A-5 | 4760 | 10 600 | 190 | Ultimate strength test. | Light seam strain in panels 10, 11, and 12. Three blown panels. All vent parachute lines burned 3 in. below the skirt. Four small tears; several small burn holes. Reefed opening shock, 21 400 lb; disreef opening shock, 15 400 lb; R/D, 27.1 ft/sec; and L/D, approximately 1.0. Successful ultimate strength test. |

TABLE IV. - SEVENTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA - Continued

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Deployment, knots | | |
| 11 | 70 A-5 | 4747 | 10 600 | 105 | To determine the turn characteristics, with the turn line pulled down 48 in. | Negligible air damage. Reefed opening shock, 10 600 lb; and full opening shock, 15 500 lb R/T, 23.7 deg/sec, turn-line force, 90 lb. |
| 12 | 70 A-5 | 4747 | 10 600 | 105 | To investigate further the turn characteristics, with the turn line pulled down 60 in. | No canopy damage. R/T 25.4 deg/sec, turn-line force, 150 lb. |
| 13 | 70 A-5 | 4747 | 10 600 | 105 | To determine what reduction in L/D could be achieved, with both turn lines pulled down 60 in. | The canopy made a stable descent, with negligible deployment damage. R/D, 28.6 ft/sec; and L/D, approximately 0.95, turn-line force, 120 to 150 lb. |

TABLE IV. - SEVENTY-FOOT PARA-SAIL TESTS, EL CENTRO, CALIFORNIA - Concluded

| Test | Canopy configuration | Vehicle weight, lb | Launch conditions | | Purpose | Results |
|---|---|---|---|---|---|---|
| | | | Altitude, ft | Deployment, knots | | |
| 14 | 70 A-5 | 4747 | 2500 | 105 | To demonstrate deployment at pad abort conditions. | The internal parachute inflated during the reefed state outside of the skirt of the main canopy. Did not adversely affect deployment. |

TABLE V. - LANDING ROCKETS[a]

| Number of tests | Objectives | Test conditions, °F | Results |
|---|---|---|---|
| 3 | Thrust-time verification, statistical. | Fired at 60°. | Satisfactory |
| 3 | Thrust-time verification, statistical. | Fired at 125°. | Satisfactory |
| 2 | Environmental testing, then fired to verify thrust time. | Cycled between -20 and +125° then shock tested to 5g laterally and longitudinally. One motor fired at -20°. The other fired at +125°. | Satisfactory |
| 2 | Environmental testing. | Vibrated at 60°. ±9g from 6 to 20 cps for 1 minute and ±0.05g to ±0.25g at 6 to 20 cps for 4 hours. | Satisfactory |
| 2 | Verify case burst pressure. | Hydrostatically loaded to destruction. | Satisfactory |

[a]Verification of the thrust-time characteristics after the rocket motors were subjected to drop-test environmental conditions.

TABLE VI. - LANDING GEAR

| Number of tests | Objectives | Test conditions | Results |
|---|---|---|---|
| 7 | Determine the pressure required to deploy the gear. | The gear was pumped down with a high-pressure nitrogen supply. | Nose gear, 50 lb/in.$^2$, average; left main, 900 lb/in.$^2$, average; right main, 2700 lb/in.$^2$, average. |
| 3 | Verify the pyrotechnic deployment of all three gears. | The vehicle was suspended in the flying attitude with stowed gear. The actuators were then remotely fired. | Satisfactory |
| 1 | Verify the deployment with only one actuator. | Same. | Satisfactory |
| 12 | Determine the impact attenuation capabilities over a range of vehicle attitudes and velocities. | The vehicle was suspended in the desired attitude (-18.7 to -8°) with deployed gear, then released from a height to furnish the desired impact velocity (7.2 to 12.0 ft/sec). | Satisfactory |

TABLE VII. - CONTROL SYSTEM

| Number of tests | Objectives | Test conditions | Results |
|---|---|---|---|
| 100 | Insure that the motors met test requirements. | Statically cycled at varying loads (6 to 120 lb). | Necessary to modify braking system and cable guides, for example, the electrical system. |
| 75 | Verify the modifications. | Statically cycled at varying loads (6 to 120 lb).<br>Continuous cycling, 50 cycles, each motor. | Satisfactory, with minor cable difficulties. |
| 5 | Verify the turn-cable fatigue tests. | Continuous cycling of cable over a short radius. | Changed to high-strength stainless-steel cable. |
| 50 | Verify the motors. | Continuous cycling at varying loads, 50 cycles. | Satisfactory, but difficult to adjust. |

TABLE VIII.- ALTITUDE SENSORS

| Number of tests | Objectives | Test conditions | Results |
|---|---|---|---|
| 12 | Verify the interim sensor deployment. | The interim sensor housing was mounted and released to deploy. | Satisfactory. |
| 37 | Verify the interim sensor activation. | The interim sensor was impacted at various horizontal and vertical velocities and impact attitudes. Match squibs fired through the sensor microswitches on closure. | Last 35 satisfactory. |
| 6 | Verify the deHavilland sensor deployment. | The sensors were mounted on the suspended vehicle, then pyrotechnically deployed. | Four satisfactory. Two malfunctions due to wrong boom guides inadvertently installed by manufacturer. |
| 4 | Verify the deHavilland sensor activation. | The deHavilland sensors were impacted at a range of vertical velocities of 20 to 51 ft/sec, with match squibs to denote microswitch closure. | Satisfactory. All microswitches jammed closed at impact. |

TABLE IX. - VISUAL REFERENCE SYSTEM TESTS

| Number of tests | Objectives | Test description | Results |
|---|---|---|---|
| 18 | Preliminary investigation of resolution and view requirements. | Film analyses plus helicopter descents simulating the Para-Sail. A pilot viewed the ground through various reticles and, for example, commanded the helicopter to turn. | Resolution, unity or better; field of view, 30° forward of straight down; approximately 90° field. |
| 30 | Determine the optical system and visual requirements for using the Para-Sail in a land landing. | Air drop tests of a 1/3-scaled model system with radio-command-actuated control system. Various optical systems were used for reference. Motion pictures through fiber optics bundle. Onboard television system with various lenses and settings. Pilot controls vehicle by watching television monitor. | Simple, uncluttered view of unity power with cross hairs to determine relative motion over ground. Field of view, approximately 90°. Visual control can be accomplished from 10 000 ft. Wind drift can be determined beginning at 6000 ft. |

TABLE IX. - VISUAL REFERENCE SYSTEM TESTS - Concluded

| Number of tests | Objectives | Test description | Results |
| --- | --- | --- | --- |
| 20 | Determine the pilot display requirements, effect of wind on landing accuracy, finalize field of view, and determine results when several pilots fly the system under realistic mission simulation. | Air drops of the 1/3-scaled model with radio-command-actuated control system and onboard television. Pilots flew the vehicle solely from the television monitor. Simulated missions were conducted, for example, cloud cover. | Field of view; 30° forward, 10° aft, and 30° to either side.<br>Unity power with cross hairs.<br>Display requirements; altimeter and compass.<br>General; the system can be maneuvered to a safe land landing in winds up to 30 ft/sec. |

TABLE X. - ONE-THIRD-SCALE SYSTEM DROP TESTS

| Number of tests | Objectives | Test description | Results |
|---|---|---|---|
| 18 | To investigate stability and dynamic motion of the system, including deployment, attitude change, and steady-state with turn. | The 375-lb vehicle was dropped from altitudes of 900 to 4000 ft at velocities up to 30 knots. The existing Gemini spacecraft structure and planned Para-Sail operation was simulated. A radio-command-actuated control system allowed inflight maneuvering. Parachute riser loads, turn-line loads, and accelerations were collected. These tests utilized a slotted front Para-Sail and single panel turn slots. Control-line travel was 21 in. Four cameras were mounted in the test vehicle. | The Para-Sail-Gemini vehicle is compatible during deployment attitude change and steady state. Oscillations about all three axes were less than 5° during level flight. The parachute remains oriented and stable during attitude change. Foreshortening riser couples does not change angle of attack of the parachute (modulating L/D), but does change suspension angle of the vehicle. R/D, approximately 18 ft/sec; L/D, 1.2; and R/T, 12 deg/sec; with angular accelerations of 4 deg/sec$^2$. The system preselected area, and faced into the wind for impact. |

TABLE XI. - ONE-THIRD-SCALE SYSTEM ROCKET-SIMULATION LANDING TESTS[a] - Concluded

| Number of tests | Objectives | Test description | Results |
|---|---|---|---|
| 26 | To determine the attenuation and stability characteristics of the landing rocket-landing gear combination. | A high-pressure, cold-gas system programed to furnish scaled thrust-time characteristics was installed. The thrust acted through nozzles simulating rocket location. The vehicle was again dropped at a range of vertical and horizontal velocities and pitch and yaw angles. Hard-packed soil covered with canvas was used for the landing surface. | System stability was established over the following range; horizontal velocity, 0 to 30 ft/sec; vertical velocity (initial), 30 ft/sec; pitch angle (13° nose down), ±5°; and yaw angle, ±15°. Impact accelerations were on the order of 2g to 3g. |

[a]Determination of the vehicle dynamics during rocket firing and associated impact and post-impact accelerations over a wide range of conditions.

TABLE XII. - FULL-SCALE SYSTEM TESTS

| Test | Objectives | Test description | Results |
|---|---|---|---|
| 1 | Water landing<br>To verify the drop method; the evaluation of R and R canister drogue and separation system, Para-Sail deployment from separated R and R canister, the system dynamics during attitude change, and the vehicle suspension attitude and stability. | Onboard systems include an 18-ft ringsail drogue; R and R canister separation mechanism; attitude-change mechanism; and an 80-ft Para-Sail. | Satisfactory. Deployment and attitude change accomplished with no difficulties. No discernible oscillation in steady-state descent with horizontal suspension attitude. |
| 2 | Water landing<br>To investigate the dynamic behavior during inflight maneuvering. | 18-ft ringsail drogue; R and R canister separation and attitude-change mechanisms; 80-ft Para-Sail; turn motors (21-in. travel) 84-ft ringsail emergency parachute; and an impact-switch disconnect system. | The Para-Sail reefing line failed just prior to disreef, and the premature disreef caused the canopy to fail. The emergency parachute was radio commanded, and recovered the vehicle with no damage. |

TABLE XII. - FULL-SCALE SYSTEM TESTS - Continued

| Test | Objectives | Description | Results |
|---|---|---|---|
| 3 | Water landing<br>To repeat test 2. | Same as test 2, with heavier reefing line. | All systems functioned correctly. The turn system furnished rates of turn of 10 to 12 deg/sec. The vehicle was maneuvered to the target and faced into the wind prior to impact. The missed distance was approximately 50 yd. |
| 4 | Water landing<br>To continue the investigation of the dynamic behavior and incorporation of the interim altitude sensor, with flashbulbs simulating the rockets. | 22-ft ringslot drogue; R and R canister separation and attitude-change mechanisms; 80-ft Para-Sail; turn motors (21-in. travel); 84-ft ringsail emergency parachute; interim altitude sensor; and an impact-switch disconnect system. | All systems functioned correctly. The forward Y-bridle, leg-load cell failed, causing a 20° nose-down attitude. The altitude sensor deployed and armed. Turn system repeated the performance of previous test. The vehicle was maneuvered to the target and faced into the wind prior to impact. The flashbulbs fired at |

TABLE XII. - FULL-SCALE SYSTEM TESTS - Continued

| Test | Objectives | Description | Results |
|---|---|---|---|
| 4 | Water landing | | altitude-sensor closure. The missed distance was approximately 40 yd. |
| 5 | Water landing<br>To continue the system evaluation. Turn-motor travel doubled. | Same as previous test. | All systems functioned correctly; except, that the right turn line failed structurally at attitude change. Left turn rate was 20 deg/sec. Altitude sensor deployment and arming were satisfactory, and the flashbulbs fired on closure. Using the left turn; the vehicle was maneuvered into the landing area and faced into the wind prior to impact. The missed distance was approximately 90 yd. |

TABLE XII. - FULL-SCALE SYSTEM TESTS - Continued

| Test | Objectives | Description | Results |
|---|---|---|---|
| N/A | Full-scale crane drop<br>To investigate impact dynamics under closely controlled conditions. Prior to incorporating the rockets into the drop testing, a crane drop was conducted with the rockets, predeployed gear, and the altitude sensor. | The vehicle was suspended in the correct landing attitude and sufficient height to furnish nominal Para-Sail descent velocity at altitude-sensor closure. The vehicle was then released and allowed to fall free. The impact surface was 1/2-in. steel plate over concrete. | The vehicle accelerated rapidly; then, the altitude sensor closed, and the rockets fired, decelerating the vehicle to approximately 7 ft/sec at impact. The gear stroked at approximately 2/3 of the maximum. |
| 6 | Water landing<br>To incorporate landing rockets. | 22-ft ringslot drogue; R and R canister separation and attitude-change mechanisms; 80-ft Para-Sail; turn motors (42-in. travel); 84-ft ringsail emergency parachute; interim altitude sensor; blast deflector, jettison mechanism, and recovery system; landing rockets; and a salt-water-activated disconnect system. | All systems functioned correctly, with the exception of the turn motors. Investigation later proved that the turn lines were too short and overloaded the motors, causing the fuses to blow. The system made a long gliding descent, impacting approximately 2 mi from the target. |

TABLE XII. - FULL-SCALE SYSTEM TESTS - Continued

| Test | Objectives | Description | Results |
|---|---|---|---|
| 7 | Water landing<br>To test the incorporation of the improved 70-ft version of the Para-Sail, and the deHavilland altitude sensor. | 22-ft ringslot drogue; R and R canister separation and attitude-change mechanisms; 70-ft Para-Sail; turn motors (42-in. travel); 84-ft ringsail emergency parachute; two deHavilland altitude sensors; blast deflector, jettison mechanism, and recovery system; landing rockets; and a salt-water disconnect switch. | All systems functioned correctly with the following exceptions: one deHavilland sensor bent and trailed 15° aft, and both turn lines failed structurally at attitude change. The system made a spiral descent and impacted approximately 200 yd from the target area. |
| 8 | Water landing<br>To change the test order, incorporating two tests to correct the difficulties encountered with the turn lines. | 22-ft ringslot drogue parachute, R and R canister separation and attitude-change mechanisms, 70-ft Para-Sail, turn motors (42-in. travel), 84-ft ringsail emergency parachute and an impact-switch disconnect system. | All systems functioned correctly. The canopy had a built-in left turn. Turn rates were 40 deg/sec left and 15 deg/sec right. The vehicle was maneuvered into the general target area and faced into the wind prior to impact. The missed |

TABLE XII. - FULL-SCALE SYSTEM TESTS - Continued

| Test | Objectives | Description | Results |
|---|---|---|---|
| 8 | Water landing | | distance was approximately 250 yd. Analysis of the test data shows the turn lines were still too short, and only half the travel could be used before stalling the motors. |
| N/A | Full-scale crane drop 2<br>To continue the impact dynamics study, to verify the stiffened motor mounts, and to study the effect of rocket erosion on a sod surface. | The vehicle was suspended in the correct landing attitude, and at a sufficient height to furnish nominal Para-Sail descent velocity at altitude-sensor closure. The landing surface was grassy sod. | All systems functioned correctly. The rockets fired on sensor closure, and the landing gear stroked approximately 2/3 the maximum. Rocket exhaust eroded a hole 132 in. wide, 96 in. long, and 28 in. deep. |
| 9 | Water landing<br>To conduct a second test for correcting turn-line length and the stowage problem. | 22-ft ringslot drogue parachute, R and R canister separation and attitude-change mechanisms, 70-ft Para-Sail turn motors (42-in. travel), 84-ft ringsail emergency | All systems functioned correctly. Drop was made in very high winds. Canopy had a 12 to 15 deg/sec built-in left turn. Forty percent takeup on right motor would counteract |

TABLE XII. - FULL-SCALE SYSTEM TESTS - Continued

| Test | Objectives | Description | Results |
|---|---|---|---|
| 9 | Water landing | parachute, and an impact-switch disconnect system. | built-in turn. Obtained rates of 45 to 50 deg/sec left turn and 6 to 8 deg/sec right turn. Poor exit point. The system impacted approximately 1000 yd downwind. |
| 10 | First land landing test<br>To test the complete system. | 22-ft ringslot drogue parachute; R and R canister separation and attitude-change mechanisms; 70-ft Para-Sail; turn motors (42-in. travel); 84-ft ringsail emergency parachute; two deHavilland altitude sensors; blast deflector, jettison mechanism, and recovery system; landing rockets; Gemini landing gear, retention system, and deployment system; television and video-tape systems; and a gear-stroke switch disconnect system. | All systems functioned correctly down to 1400 ft, at which time the right turn line failed. Spacecraft was in excellent target position at this point. Vehicle went into uncontrolled left turn, which lasted through impact. Vehicle landed and rolled over onto left side. Approximately 8g, peak. One nose-gear bracket sheared. |

TABLE XII. - FULL-SCALE SYSTEM TESTS - Continued

| Test | Objectives | Description | Results |
|---|---|---|---|
| 11 | Second land landing test<br>To test turn motors modified to prevent previous malfunction mode. | 22-ft ringslot drogue parachute; R and R canister separation and attitude-change mechanisms; 70-ft Para-Sail; modified turn motors (37-in. travel); 84-ft ringsail emergency parachute; two deHavilland altitude sensors; blast deflector, jettison mechanism, and recovery system; landing rockets; Gemini landing gear, retention system, and deployment system; television and video-tape systems; and a gear-stroke switch disconnect system. | Vehicle pitched up at attitude change, fouling one riser on the nose-gear torque arm. Riser failed, effectively cutting six suspension lines and rendering turns impossible. Decision made to stop sequence and stay with Para-Sail for recovery. Rockets fired 3.5 sec after impact, due to double malfunction in the deHavilland altitude sensors. The nose gear and television camera were severely damaged, and the vehicle suffered a minor structural deformation. |

TABLE XII. - FULL-SCALE SYSTEM TESTS - Concluded

| Test | Objectives | Description | Results |
|---|---|---|---|
| 12 | Third land landing test<br>To allow acceleration to design deployment q, the time on the drogue parachute was extended. Roll-bar fairing added to prevent riser entanglement. DeHavilland sensors replaced with interim sensor. | 22-ft ringslot drogue, R and R canister separation and attitude-change mechanisms; 70-ft Para-Sail; modified turn motors (37-in. travel); 84-ft ringsail emergency parachute; interim altitude sensor; blast deflector, jettison mechanism, and recovery system; landing rockets; Gemini landing gear, retention system, and deployment system; television and video-tape systems; and a gear-stroke switch disconnect system. | All systems functioned correctly. The turn system furnished rates of turn of 12 to 15 deg/sec. The vehicle was maneuvered approximately 2 mi crosscountry to the primary target area. The wind was light and variable (1 to 2 knots). Vehicle landed with high forward speed and slid out 59 ft. Erosion trench was 3 ft wide and 3 in. deep. No damage. Peak impact accelerations, approximately 3g. |

NASA-S-66-10305 OCT 24

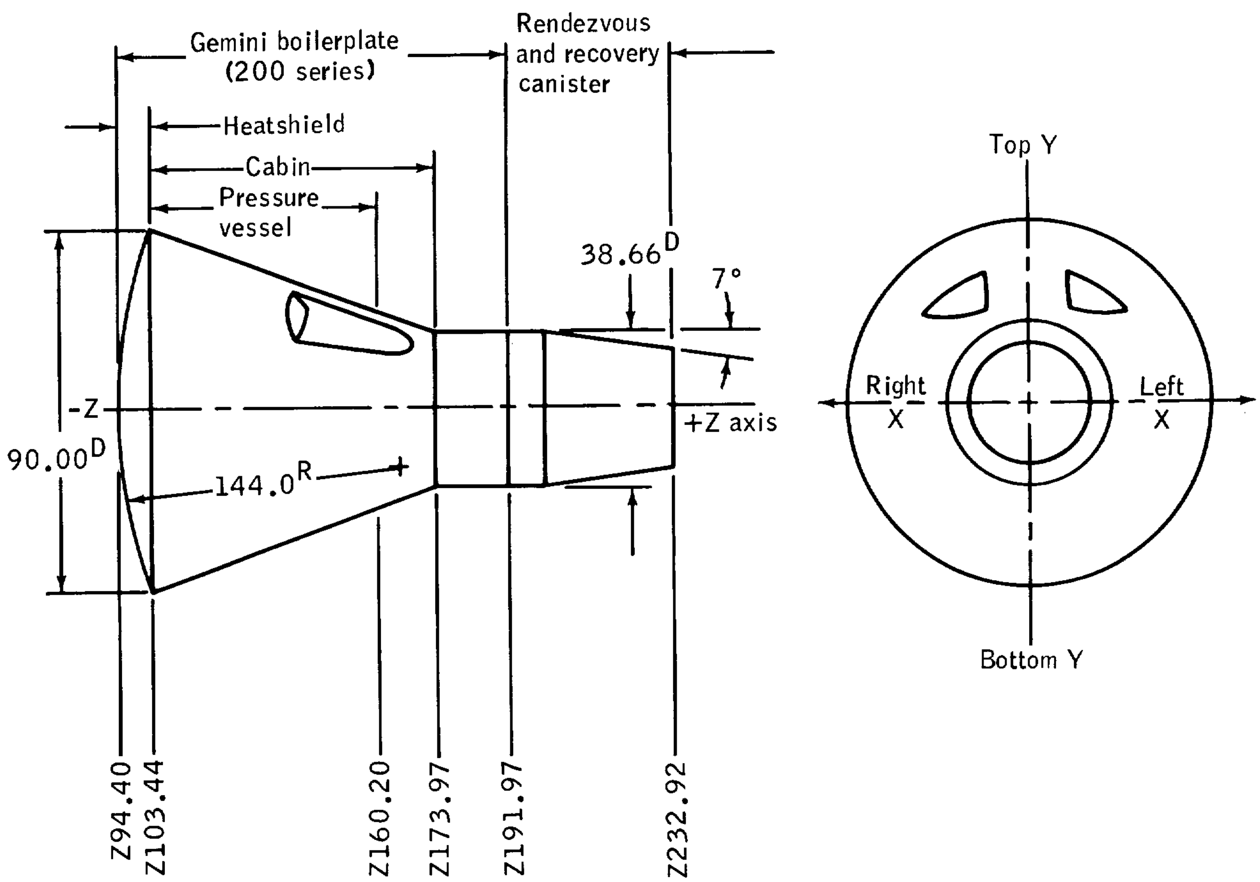



**Figure 1.- Gemini spacecraft boilerplate station and nomenclature diagram (200 series).**

NASA-S-66-10306 OCT 24

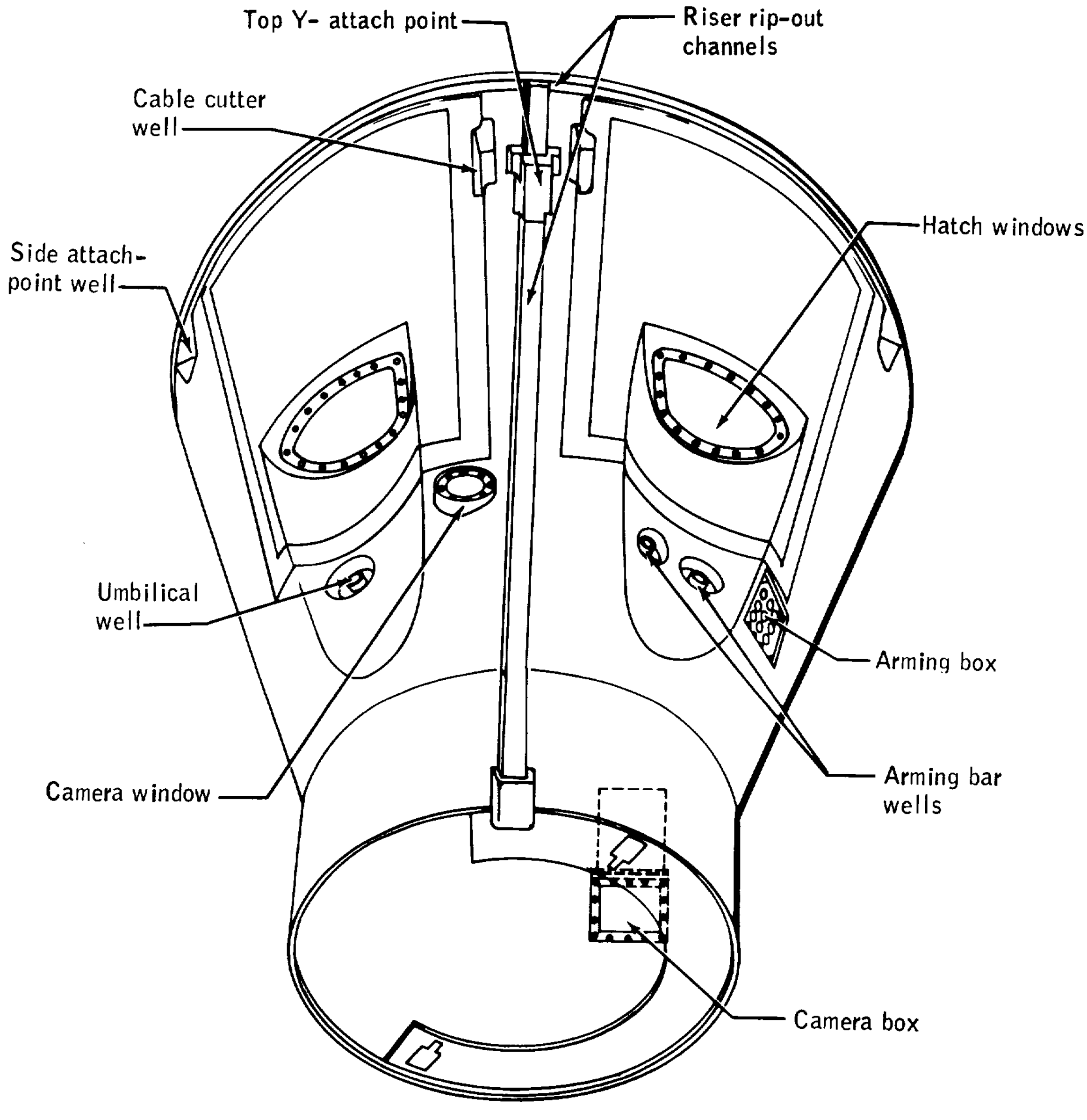



**Figure 2.- Top view of Gemini spacecraft boilerplate (200 series).**

NASA-S-66-10307 OCT 24

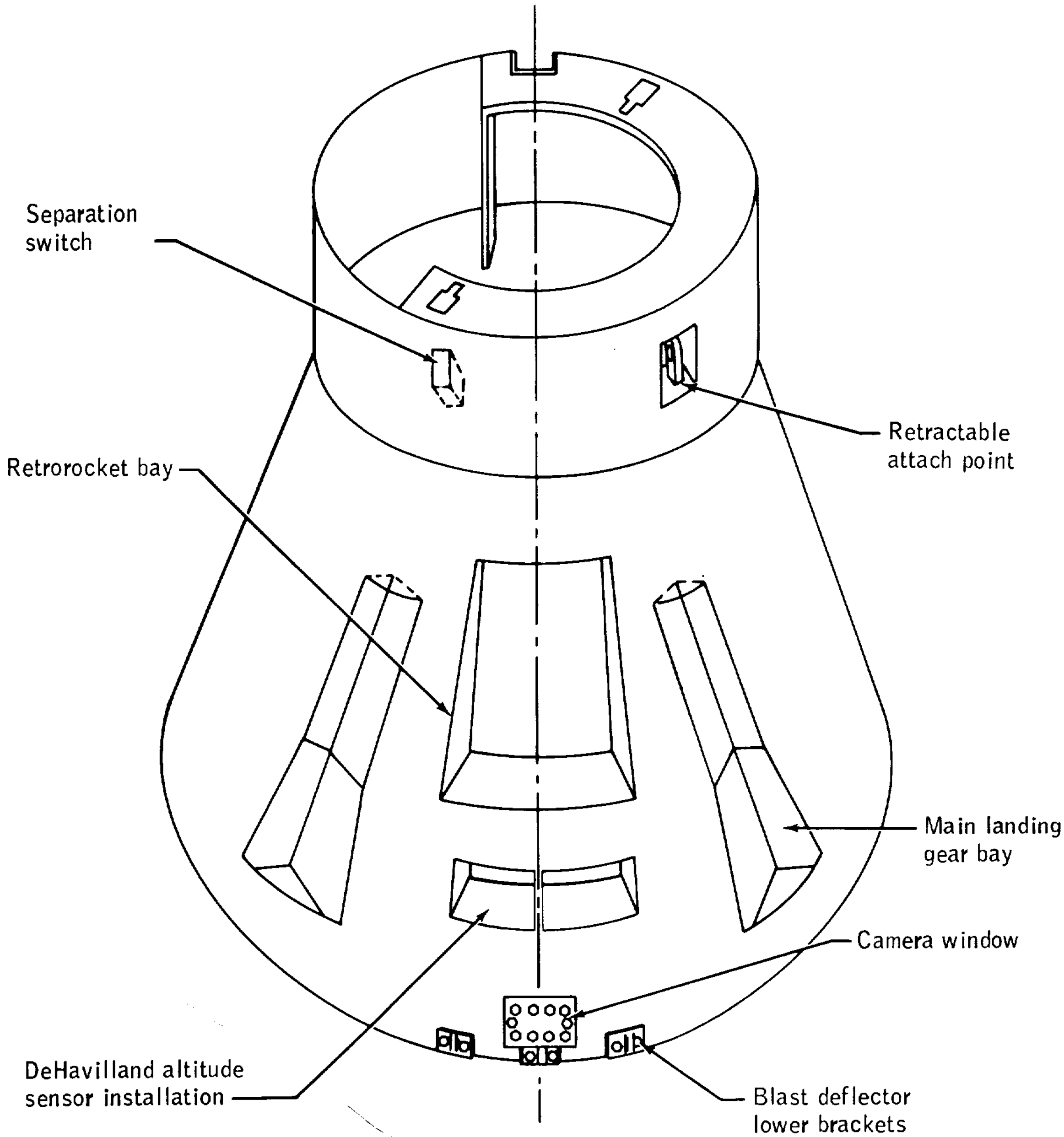



**Figure 3.- Bottom view of Gemini spacecraft boilerplate (200 series).**

NASA-S-66-8846 SEP 19

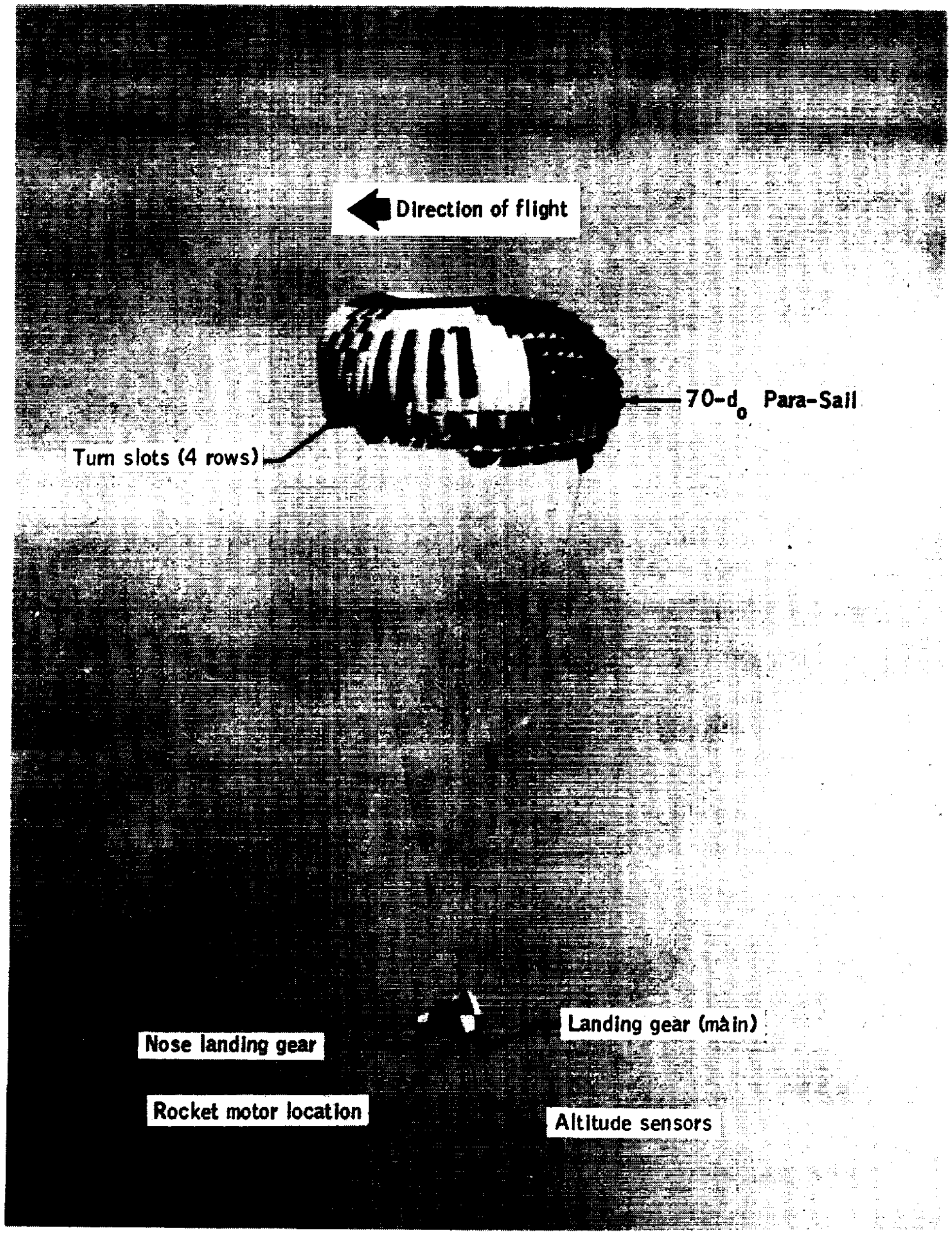



Figure 4.- Land landing system in flight


Figure 5.- Rocket-motor location.

NASA-S-66-8844 SEP 19

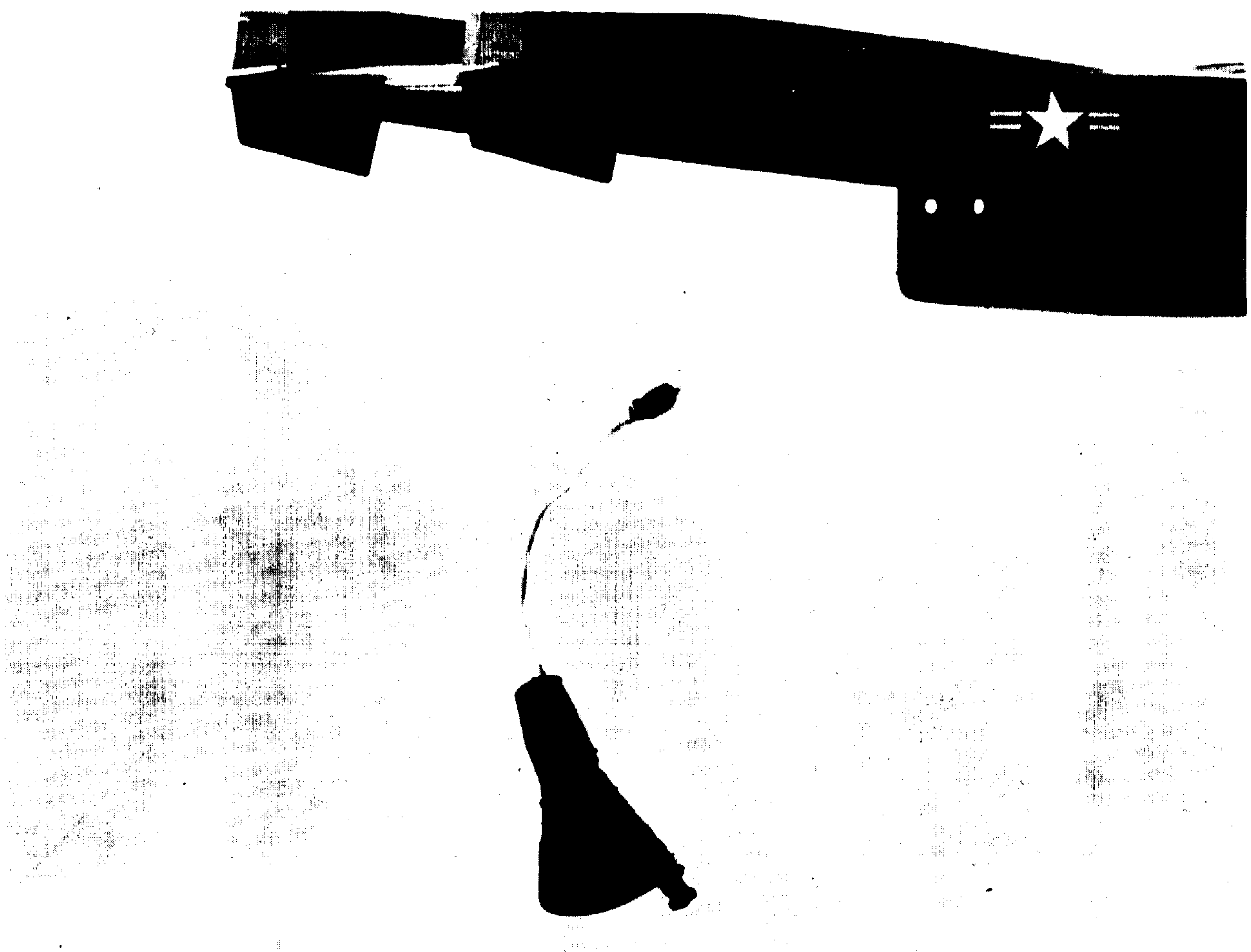


Figure 6.- Launch from the aircraft.

NASA-S-66-8843 SEP 19

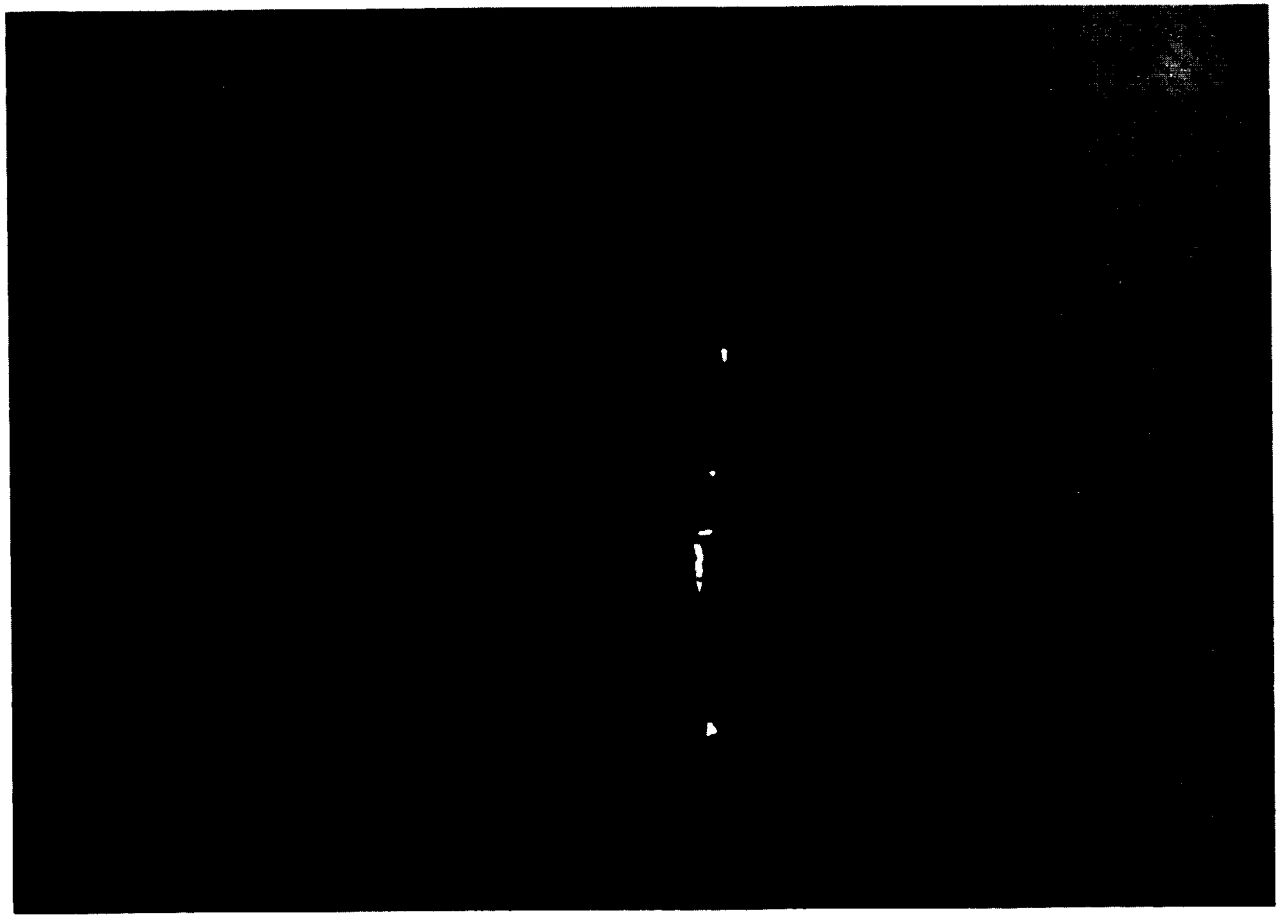


Figure 7.- Parachute deployment.

NASA-S-67-568

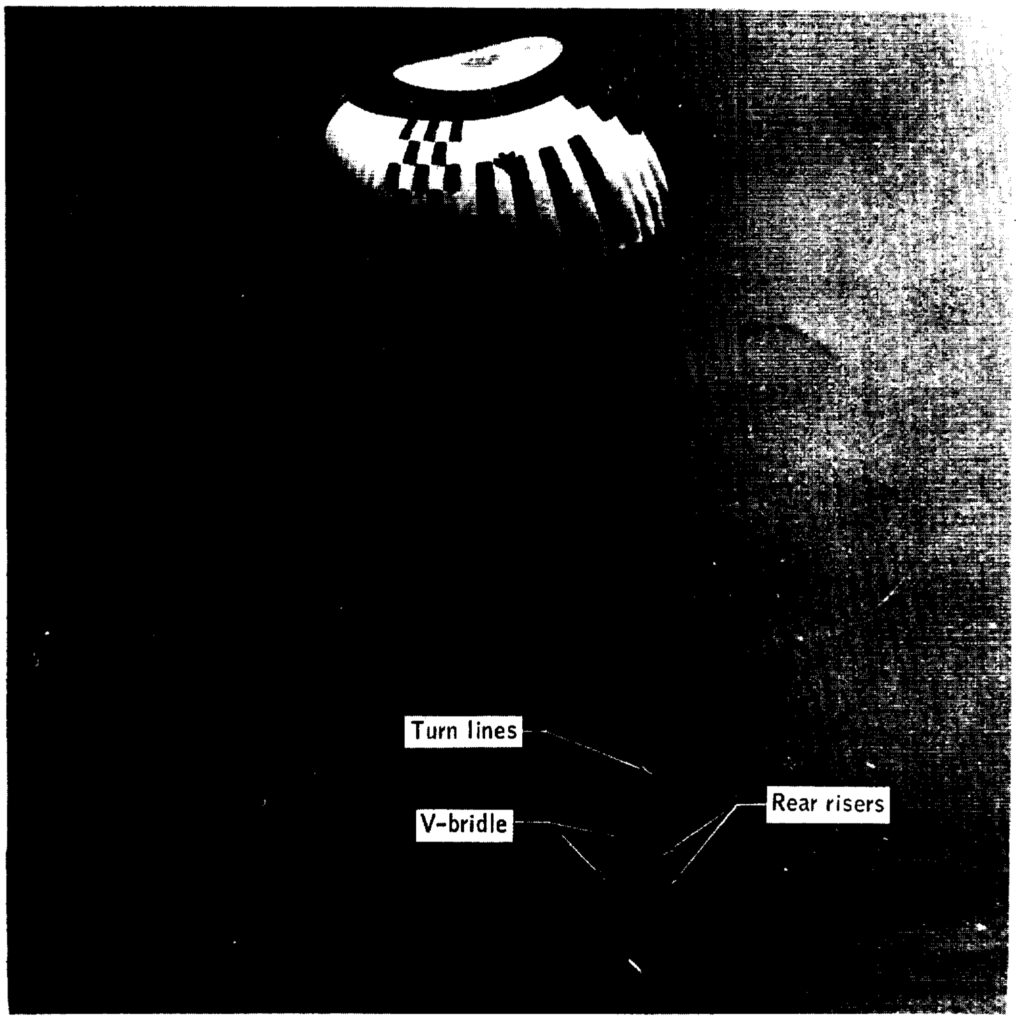



Figure 8.- Flying attitude.

NASA-S-66-8841 SEP 19

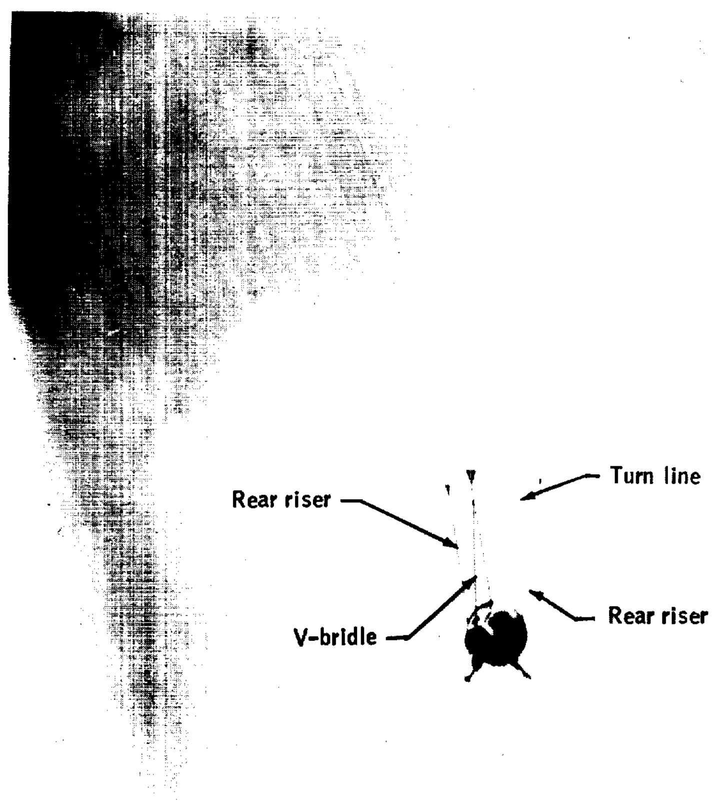



Figure 9.- Suspension system.

NASA-S-66-8840 SEP 19

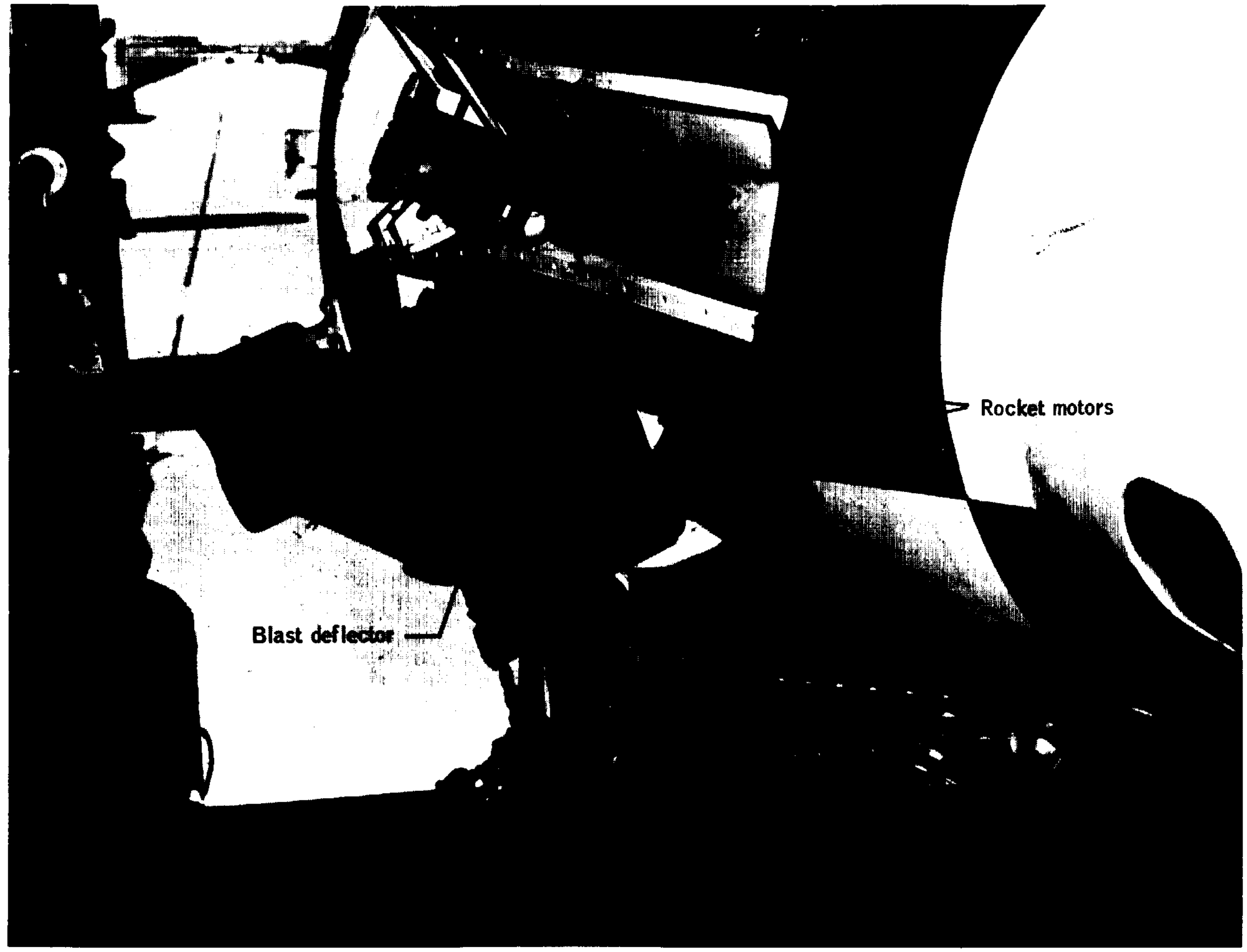



Figure 10.- Blast deflector.

NASA-S-66-8839 SEP 19

Figure 11.- Command console.

NASA-S-66-8838 SEP 19

Figure 12.- Rocket ignition on land.

NASA-S-66-8837 SEP 19

Figure 13.- Slide out.

NASA-S-66-8836 SEP 19

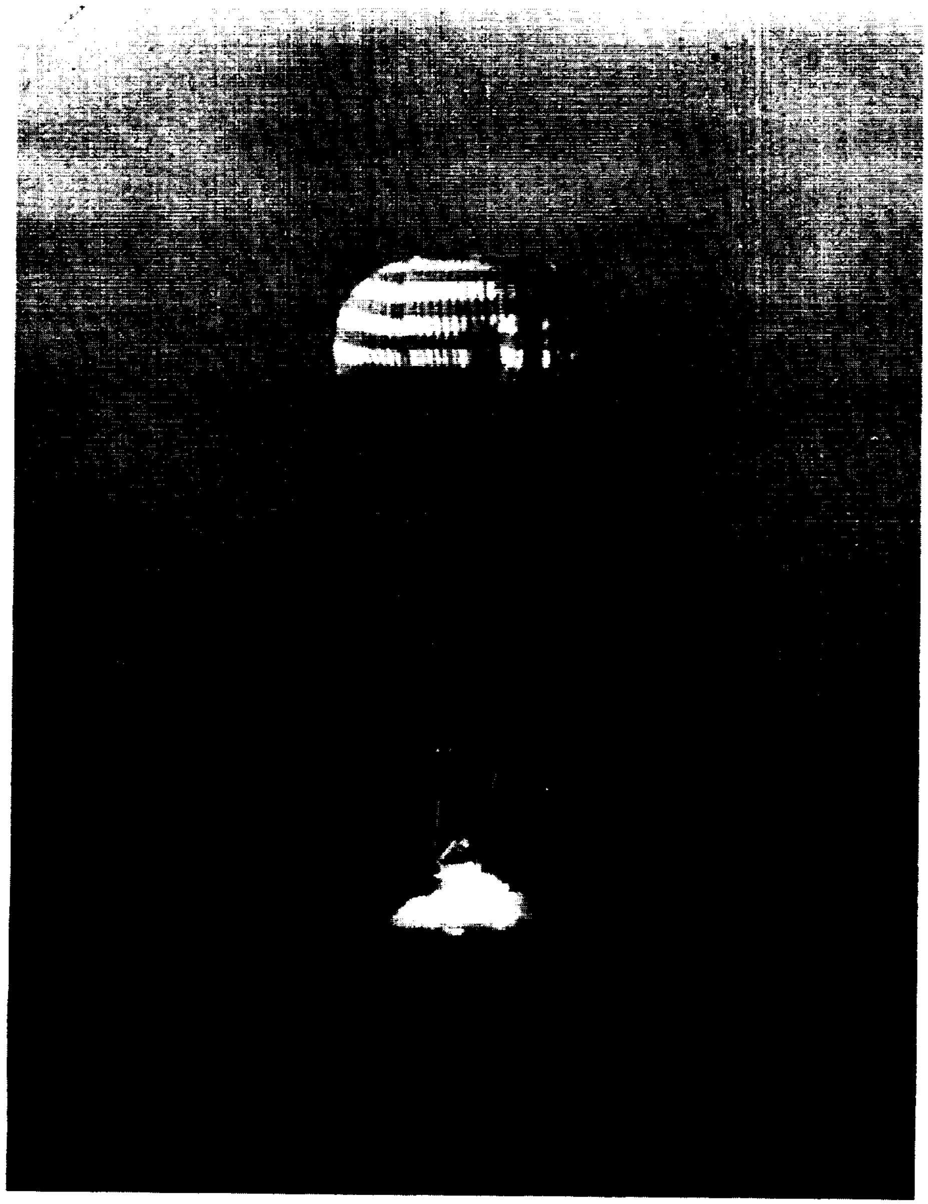


Figure 14.- Rocket ignition on water.

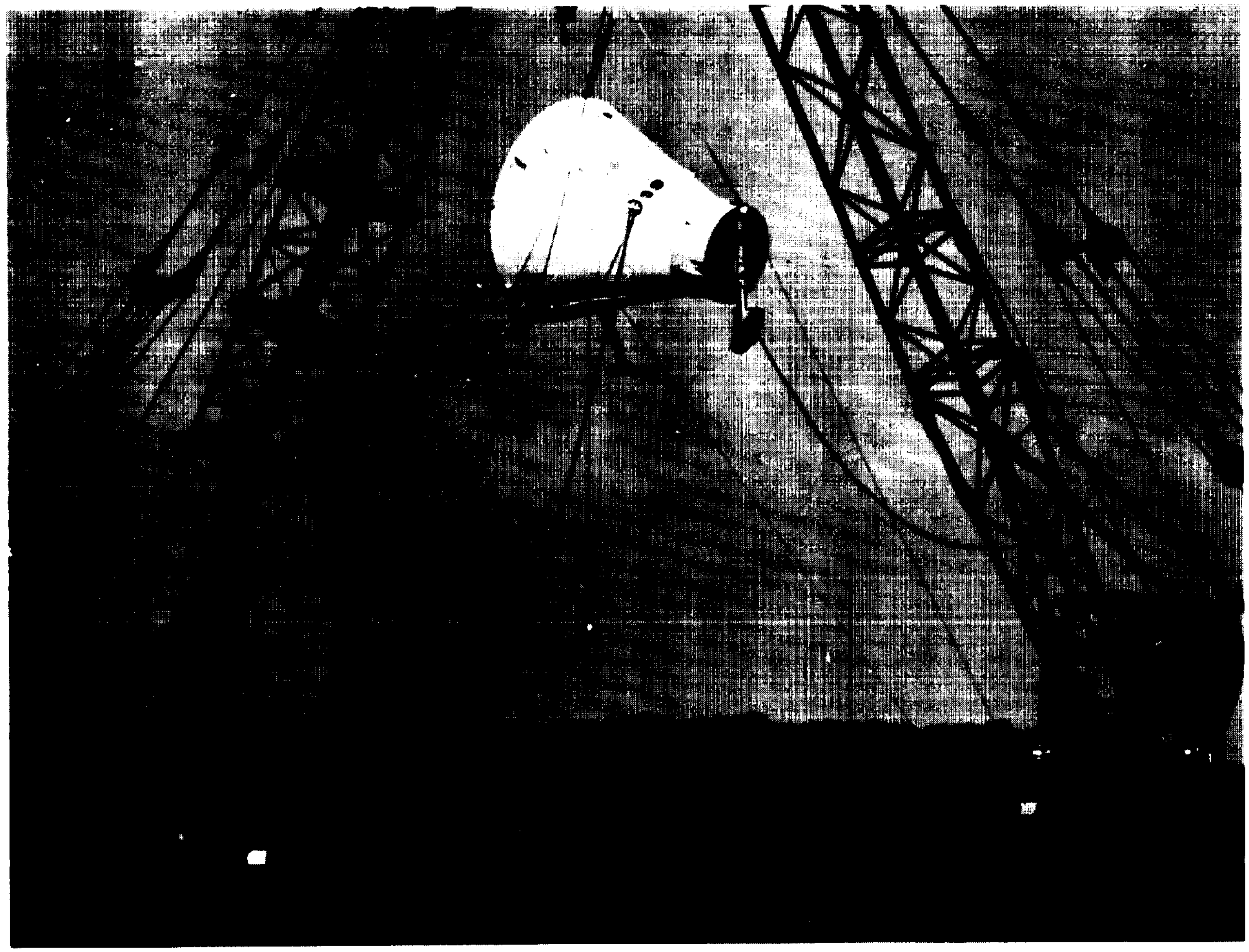
NASA-S-66-8835 SEP 19

Figure 15.- Test 3, impact.

NASA-S-66-8834 SEP 19

Figure 16.- Test 5, in flight.

NASA-S-66-8833 SEP 19



(a) Release.

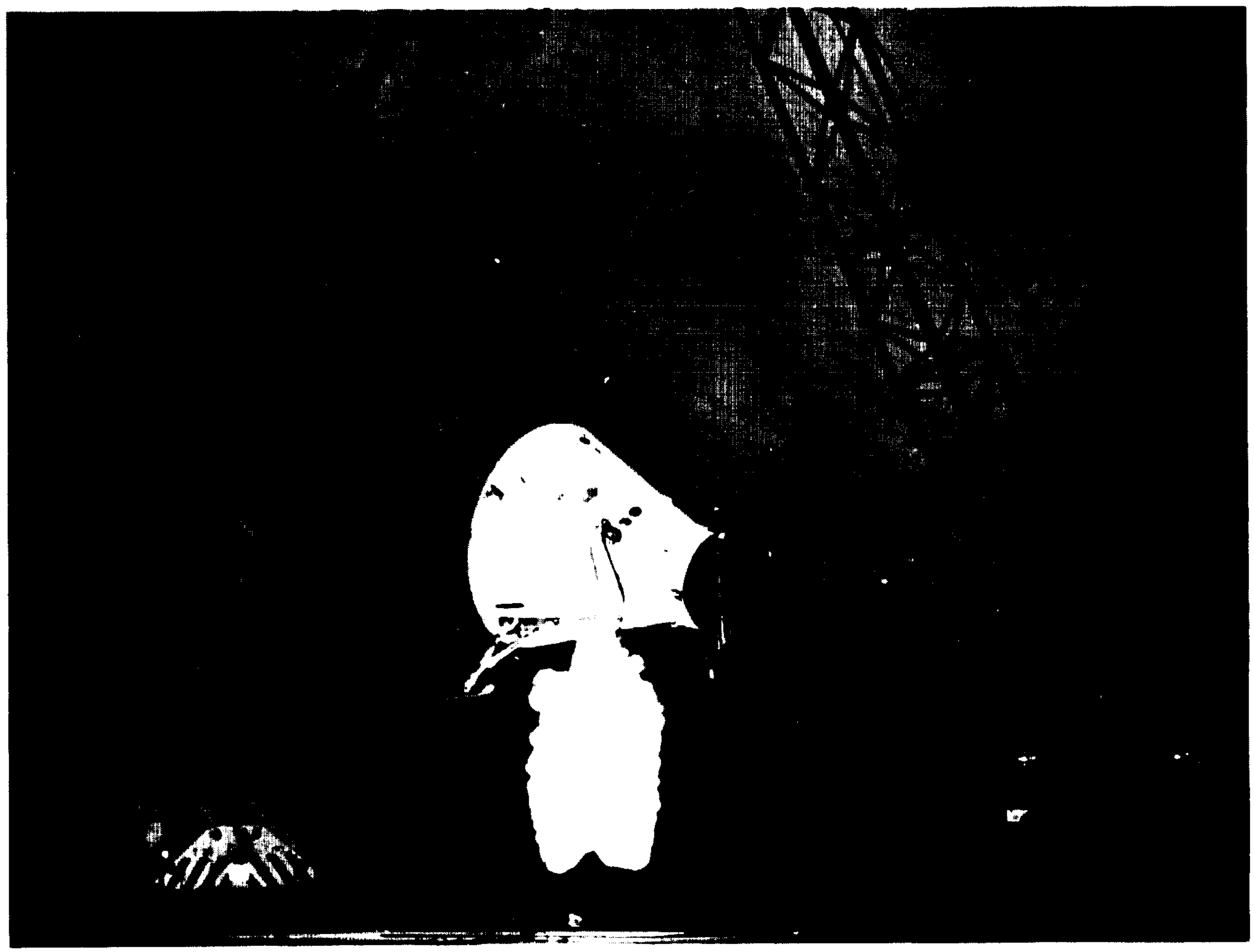
Figure 17.- Crane drop I.

NASA-S-66-8832 SEP 19

(b) Free fall.

Figure 17.- Continued.

NASA-S-66-8831 SEP 19



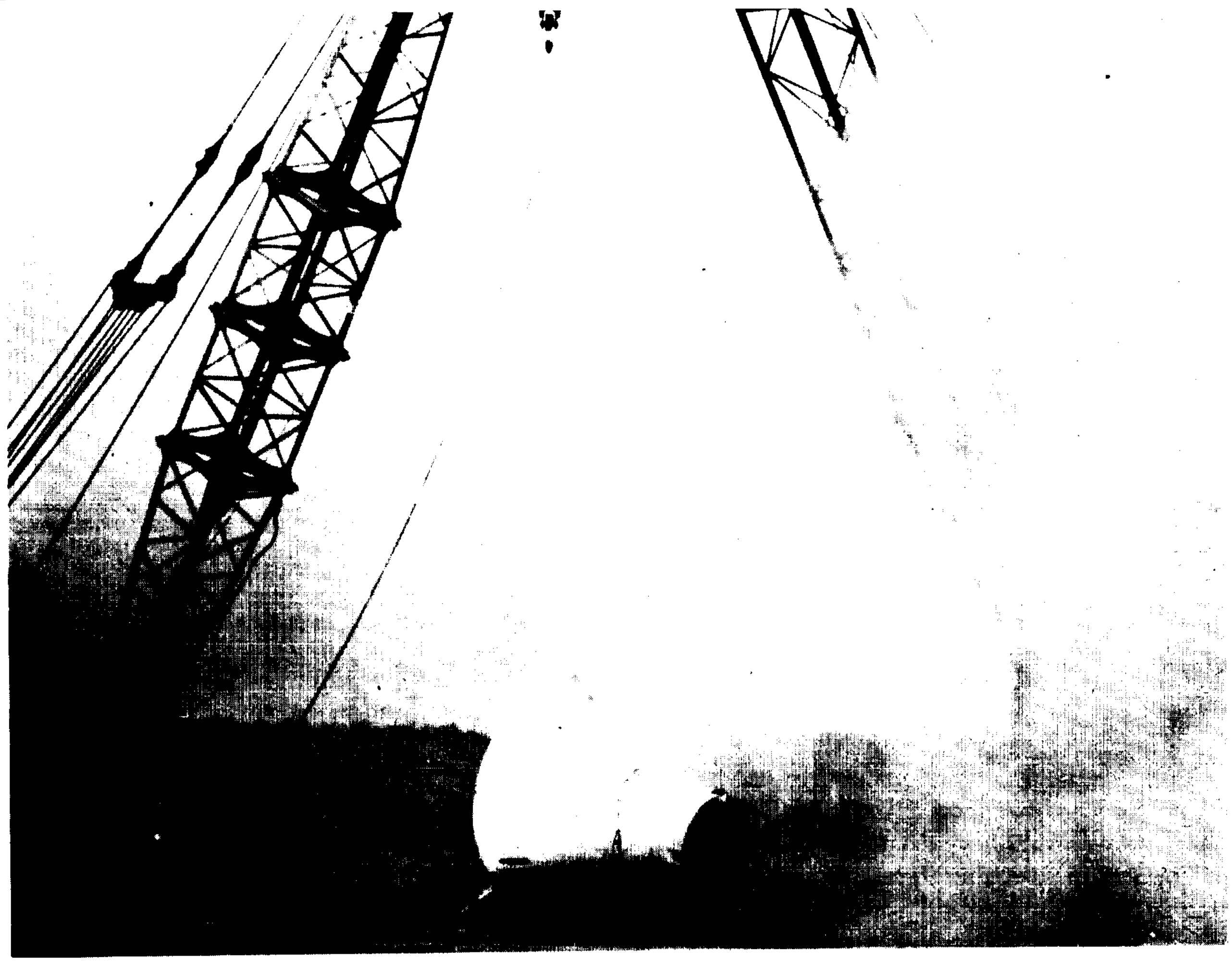
(c) Ignition.

Figure 17.- Continued.

NASA-S-66-8830 SEP 19

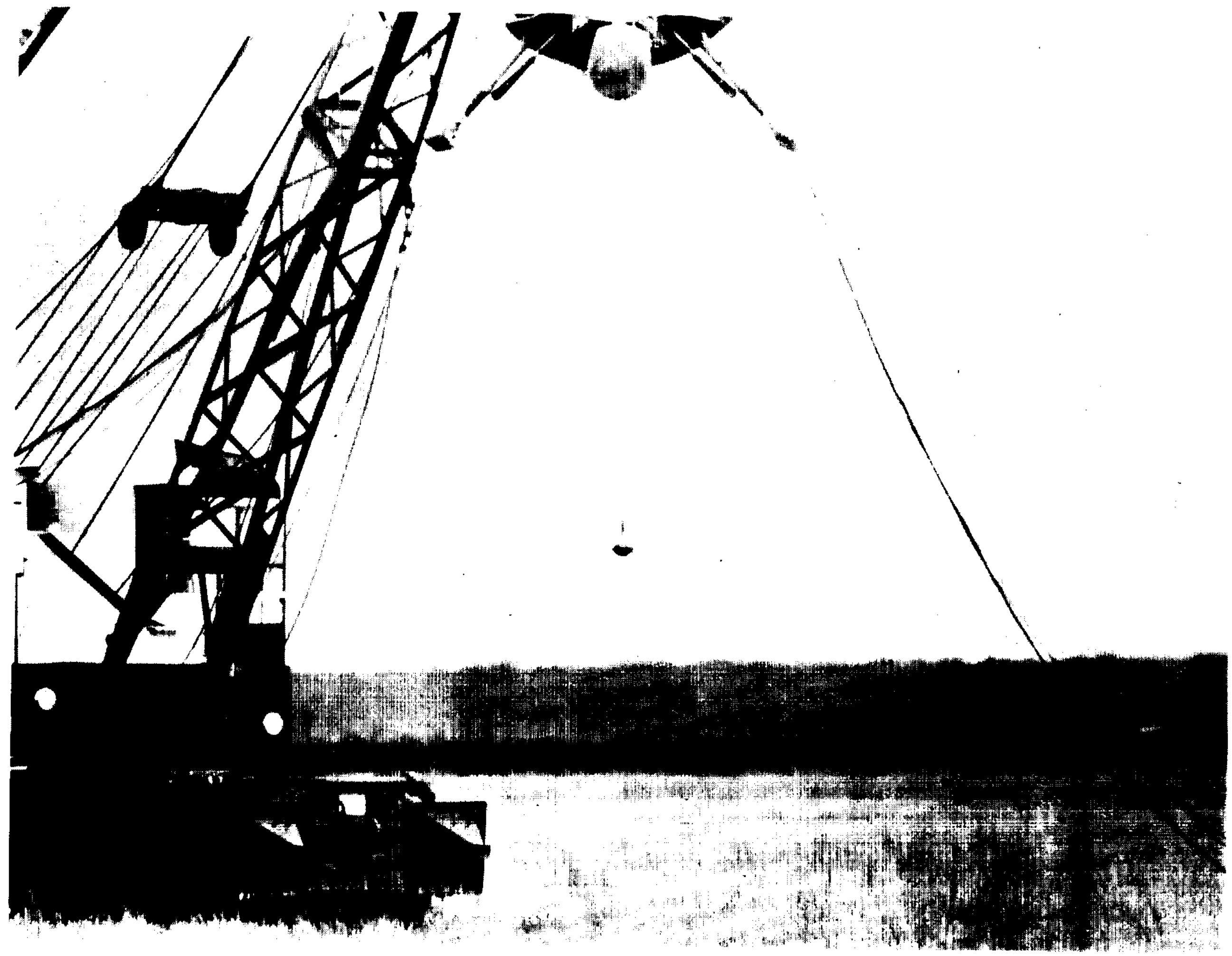
(d) High thrust.
Figure 17.- Continued.

NASA-S-66-8828 SEP 19



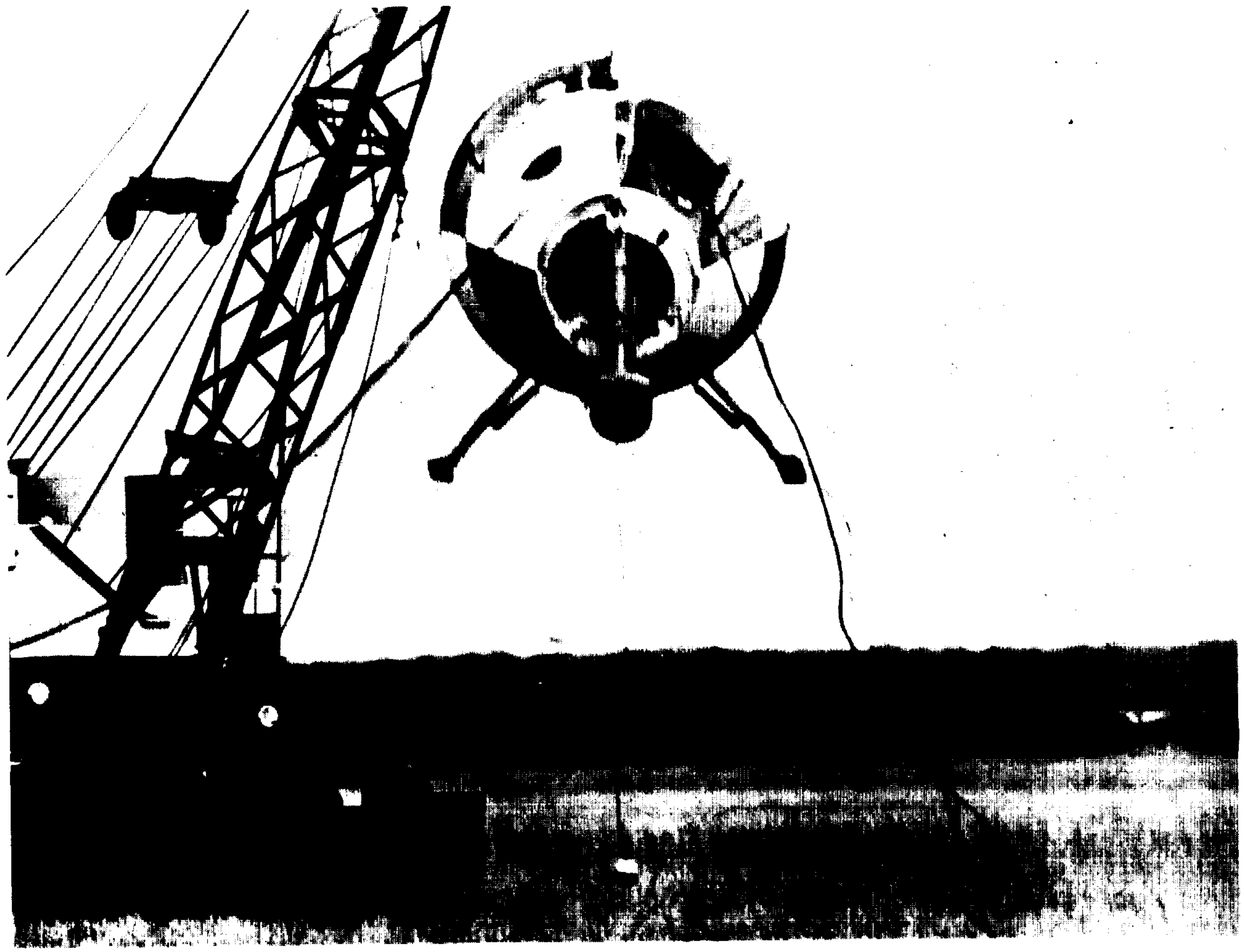
(e) Final landing attitude.
Figure 17.- Concluded.

NASA-S-66-8827 SEP 19



(a) Test set-up.
Figure 18.- Crane drop II.

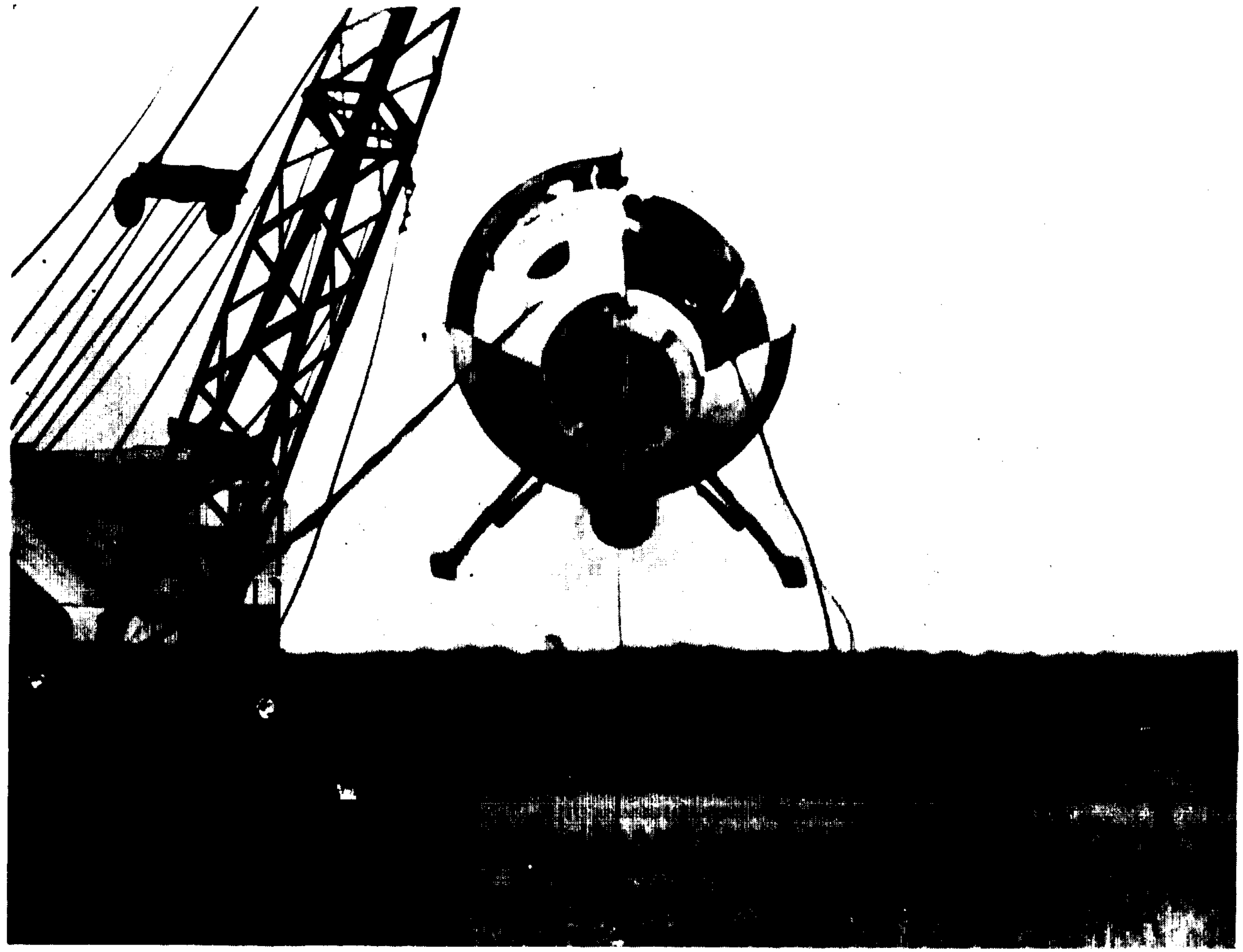
NASA-S-66-8826 SEP 19



(b) Free fall.

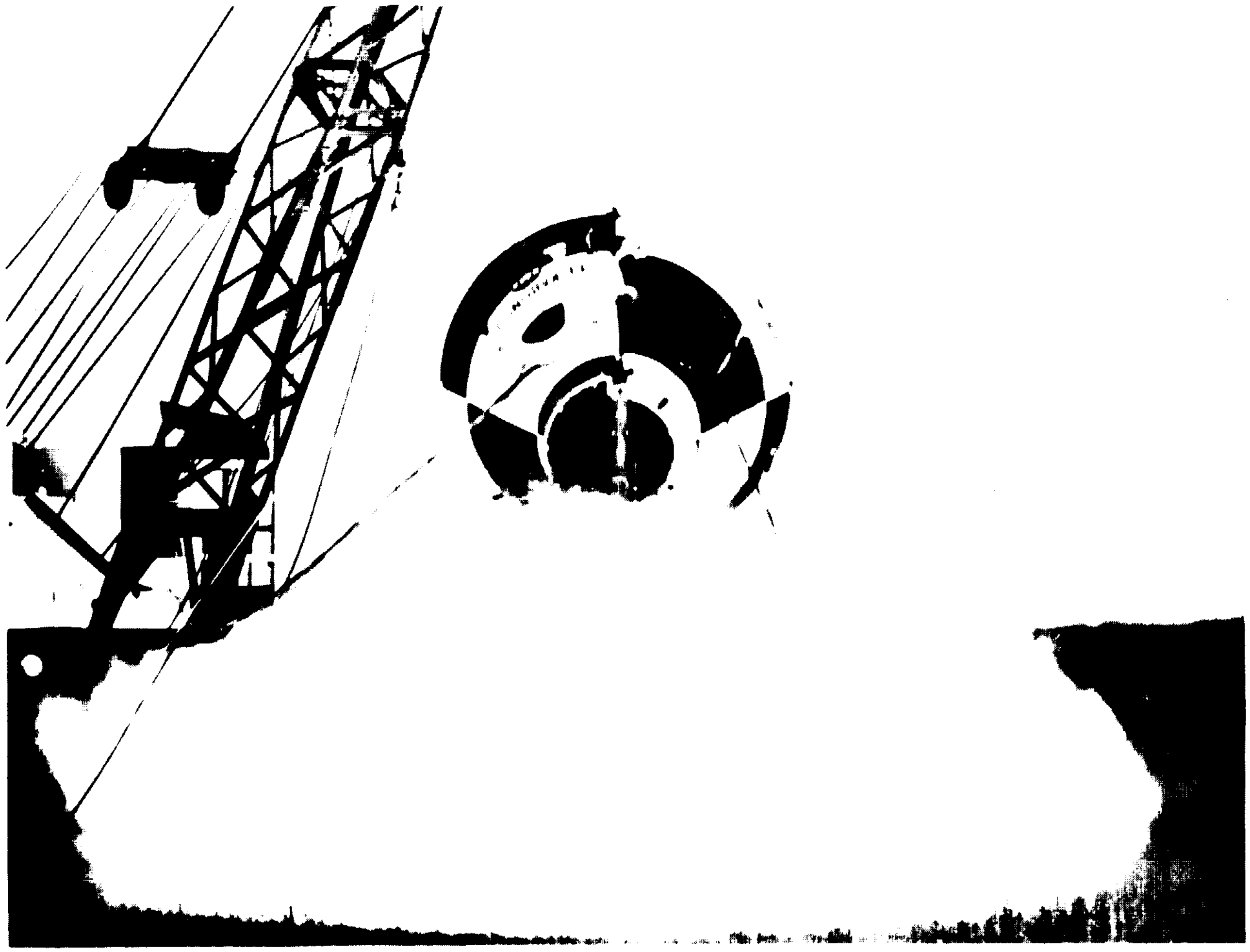
Figure 18.- Continued.

NASA-S-66-8825 SEP 19



(c) Ignition.
Figure 18.- Continued.

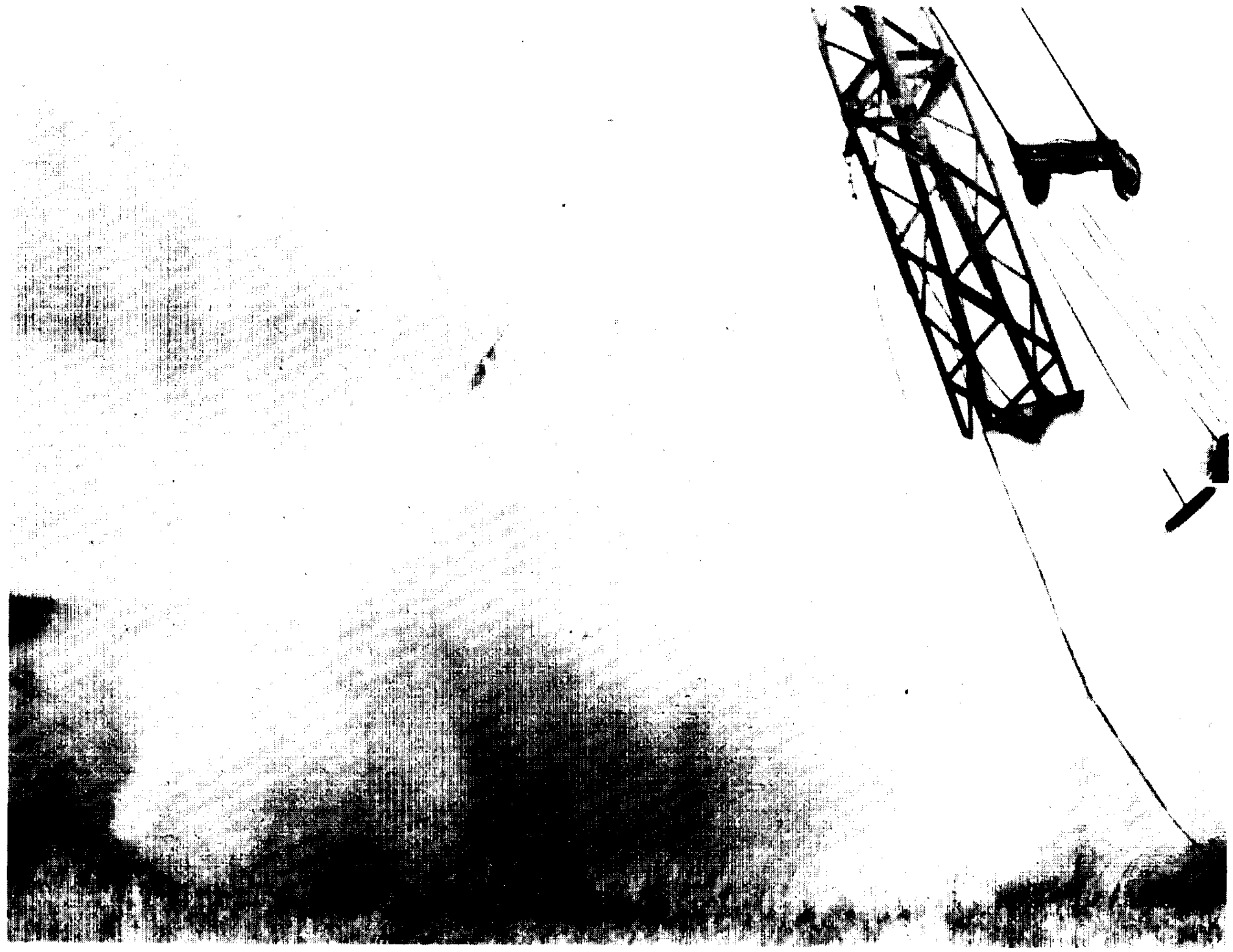
NASA-S-66-8824 SEP 19



(d) Rocket-firing sequence.

Figure 18.- Continued.

NASA-S-66-8823 SEP 19



(e) Rocket-exhaust cloud.
Figure 18.- Continued.

NASA-S-66-8822 SEP 19

(f) Dust cloud.
Figure 18.- Continued.

NASA-S-66-8821 SEP 19

(g) Final vehicle attitude.
Figure 18.- Concluded.

NASA-S-66-8820 SEP 19

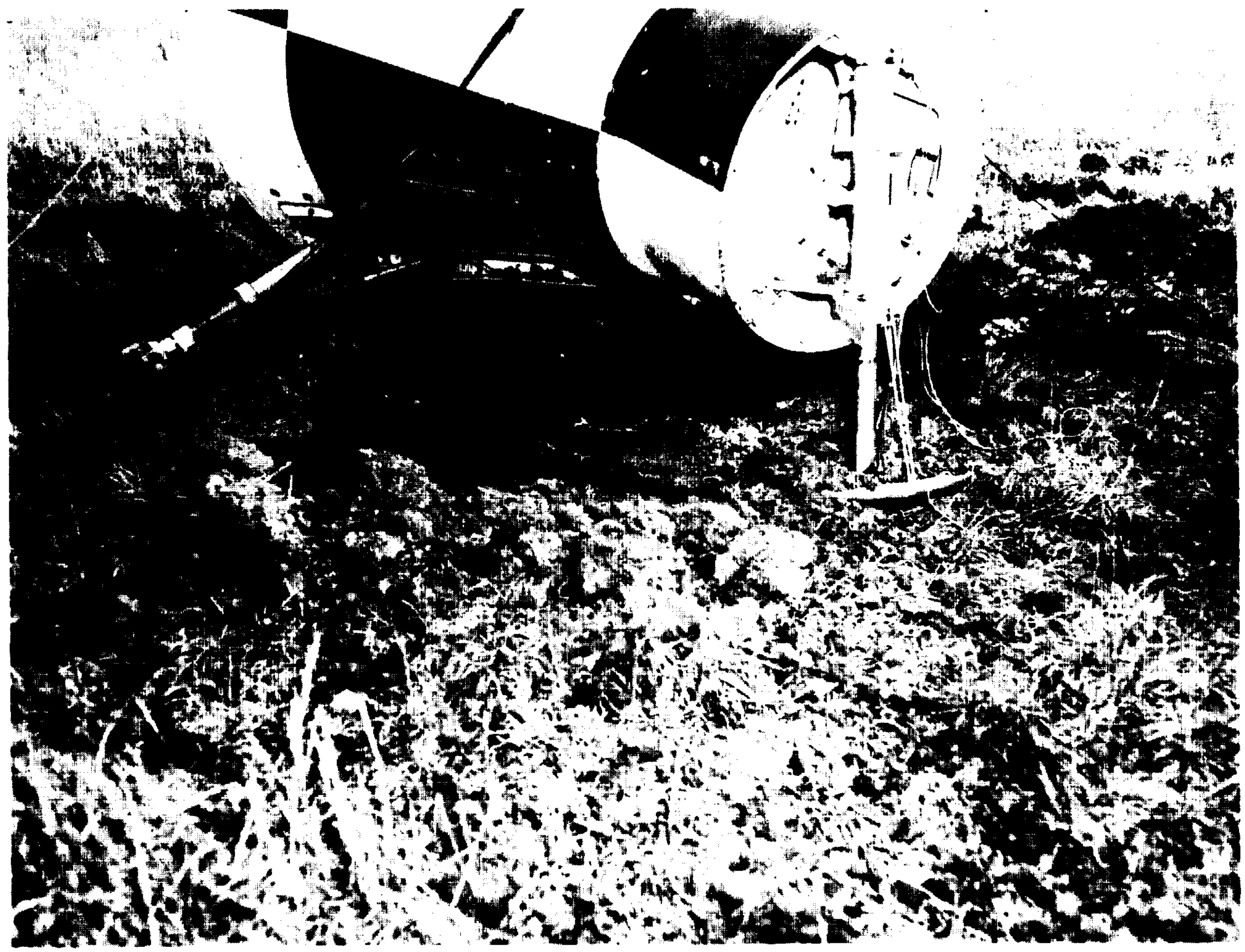


Figure 19.- Crane drop II, impact surface.

NASA-S-66-8819 SEP 19

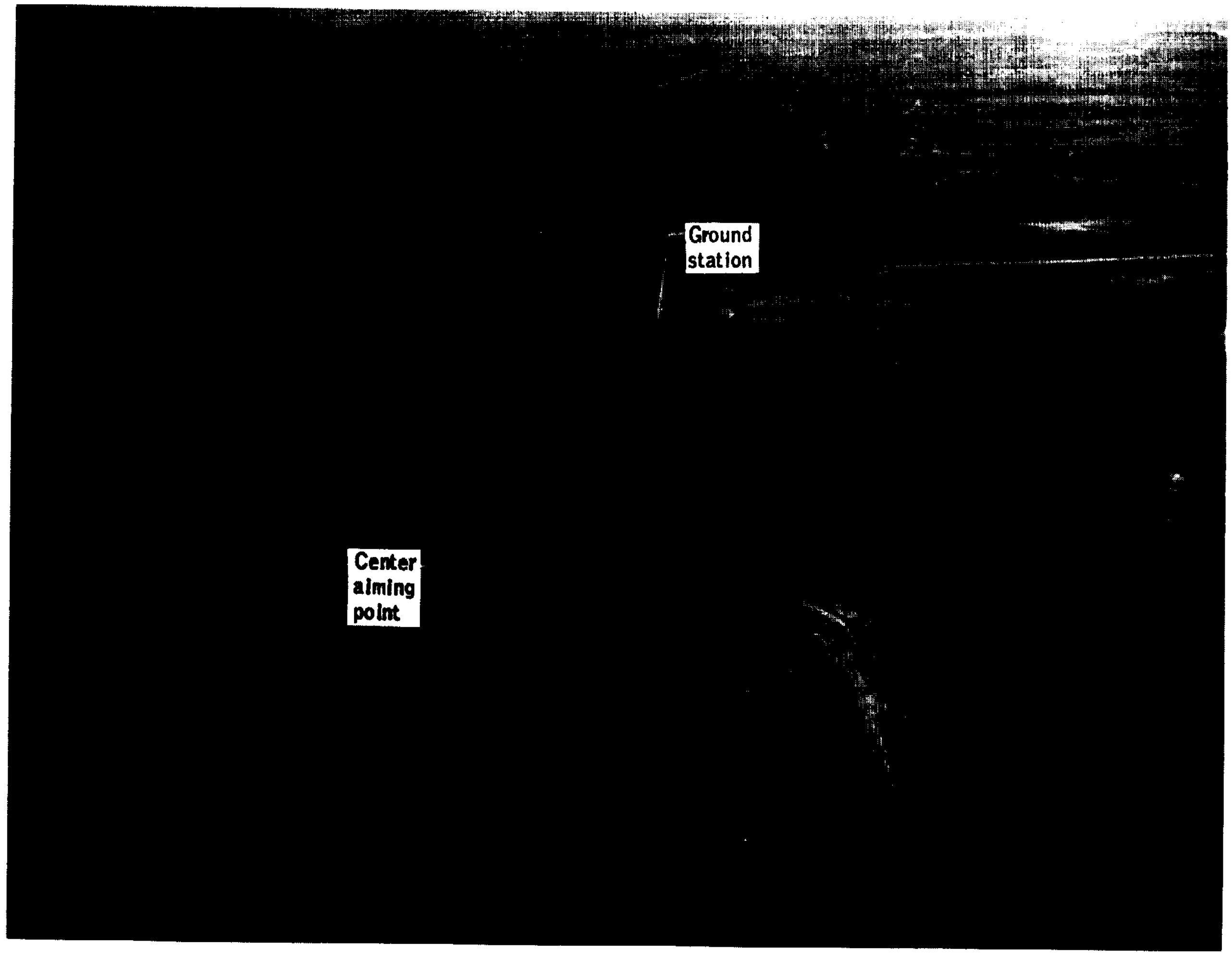
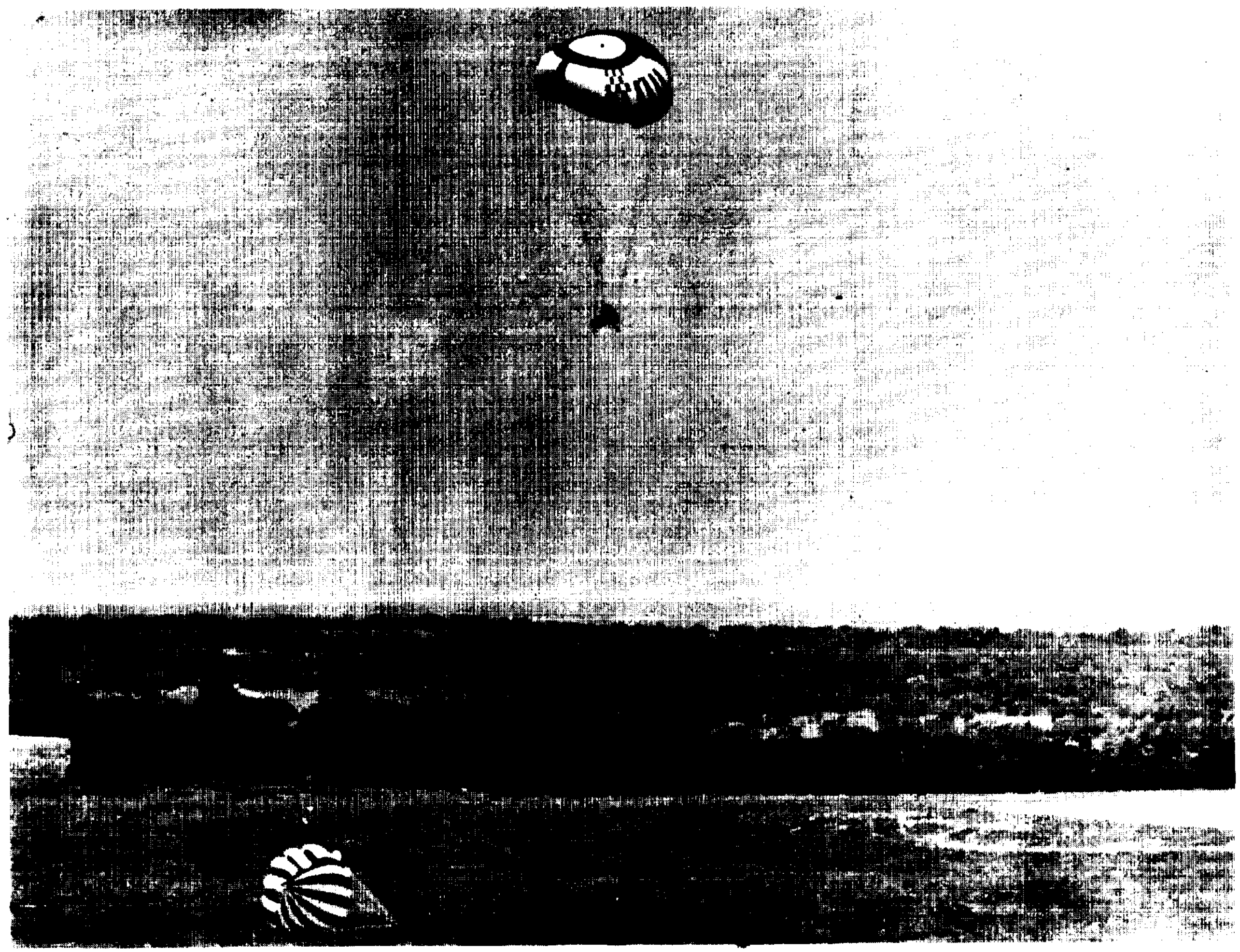


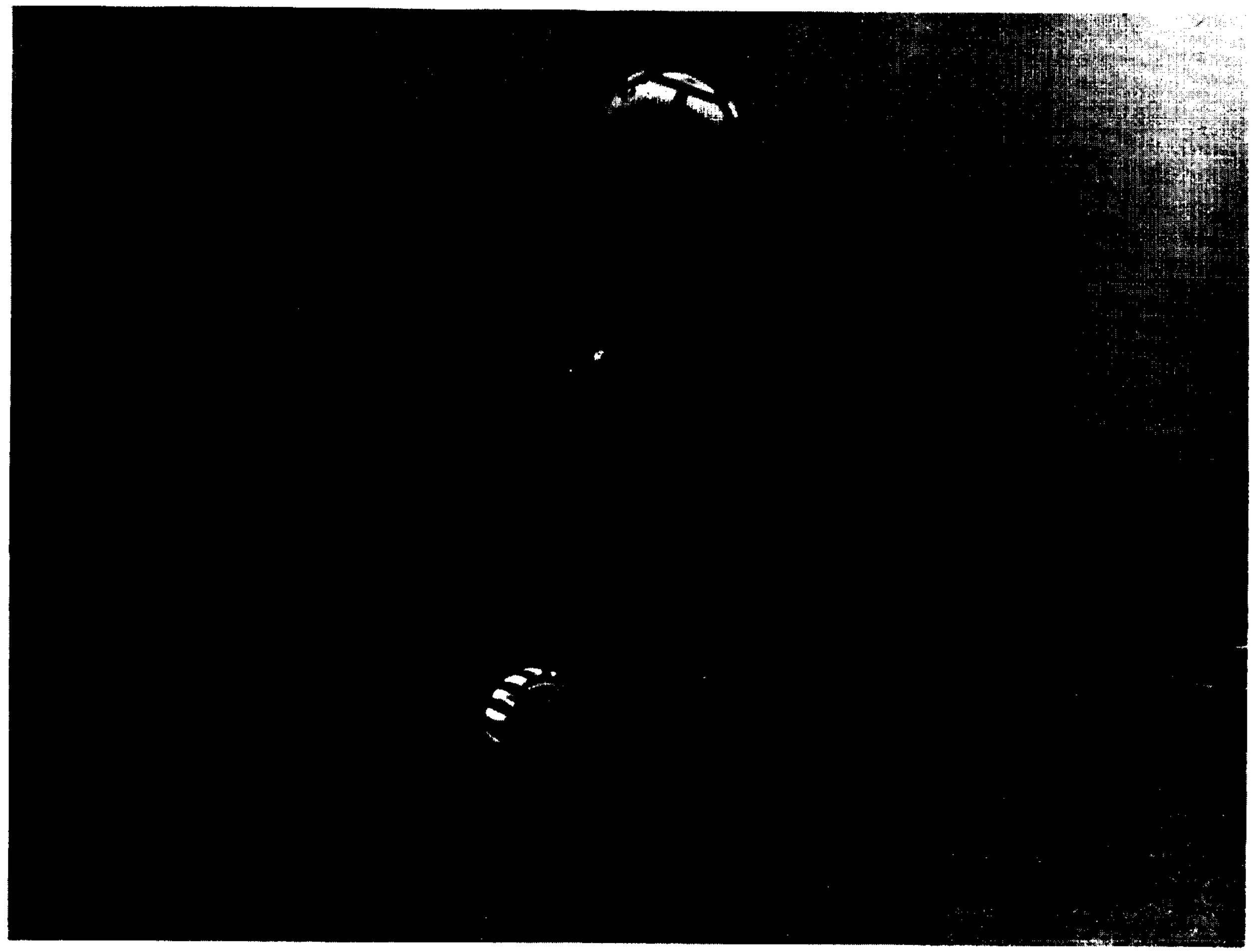

Figure 20.- Primary target area.

NASA-S-66-8818 SEP 19



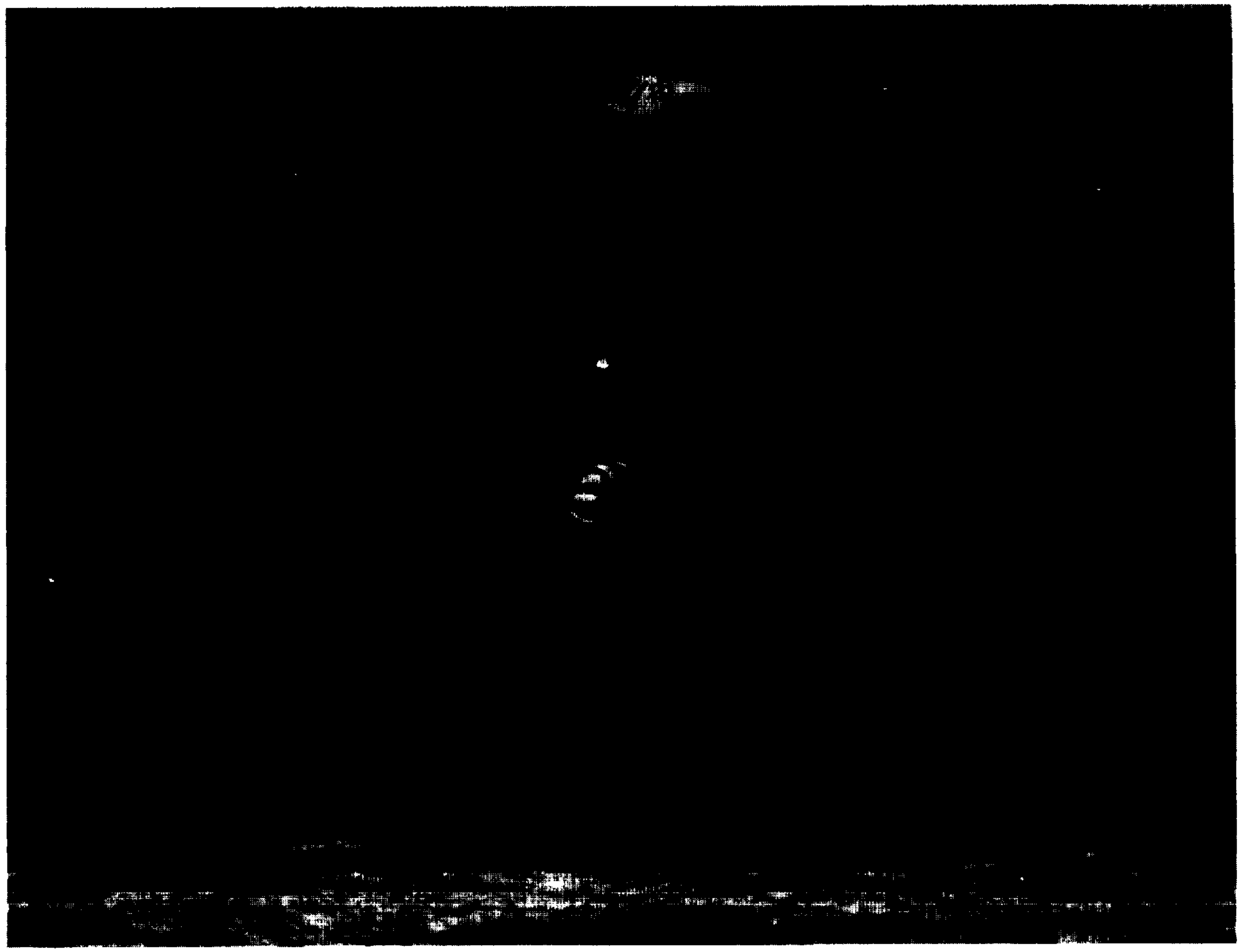
(a) Descent.
Figure 21.- Test 10.

NASA-S-66-8817 SEP 19



(b) Descent.
Figure 21.- Continued.

NASA-S-66-8816 SEP 19



(c) Descent.
Figure 21.- Continued.

NASA-S-66-8815 SEP 19

(d) Ignition.

Figure 21.- Continued.

NASA-S-66-8814 SEP 19

(e) Just prior to impact.

Figure 21.- Continued.

NASA-S-66-8813 SEP 19

(f) Vehicle tumbles.

Figure 21.- Concluded.

Figure 22.- Test 12, landing sequence.

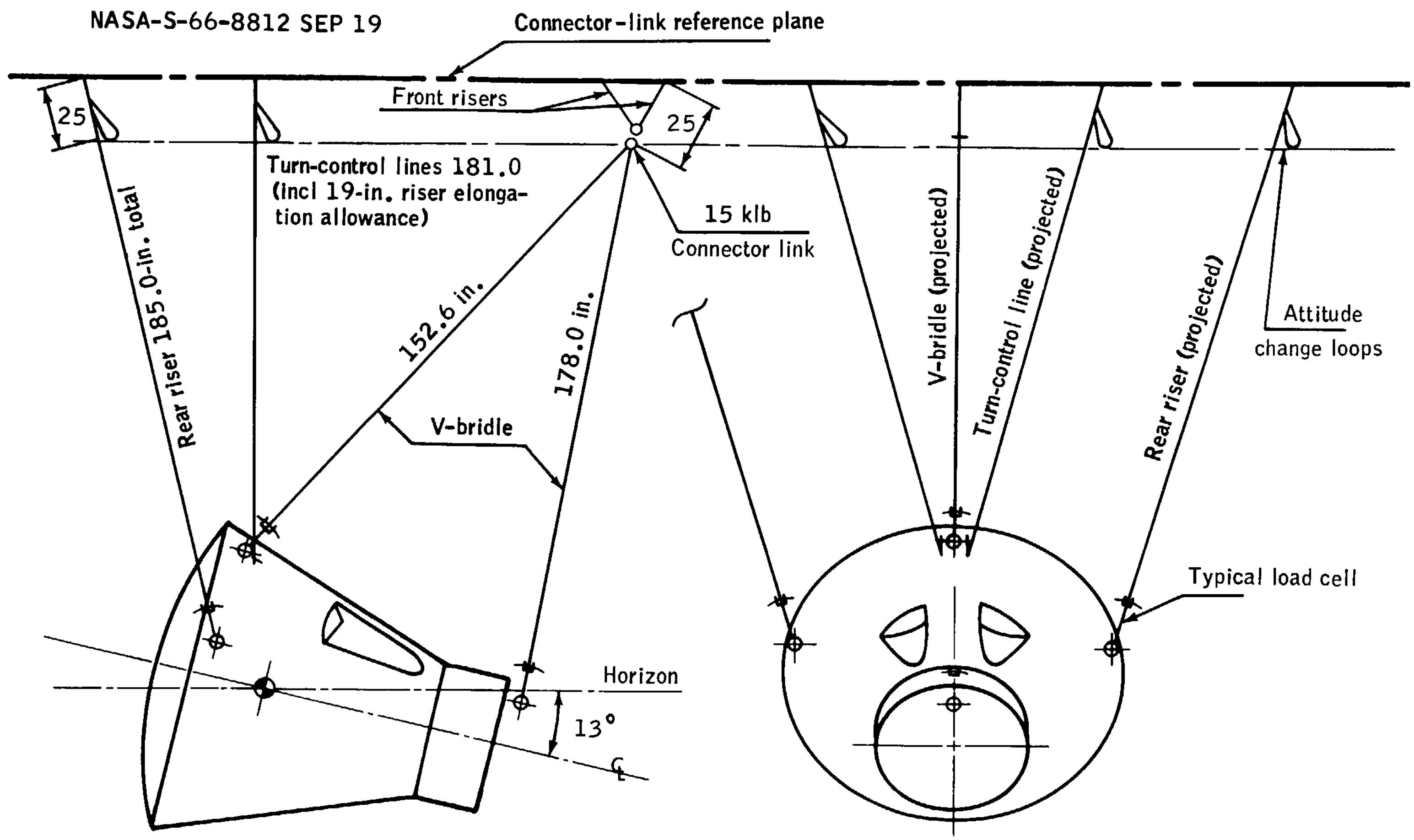



Figure 23.- Bridle geometry.

NASA-S-66-8811 SEP 19

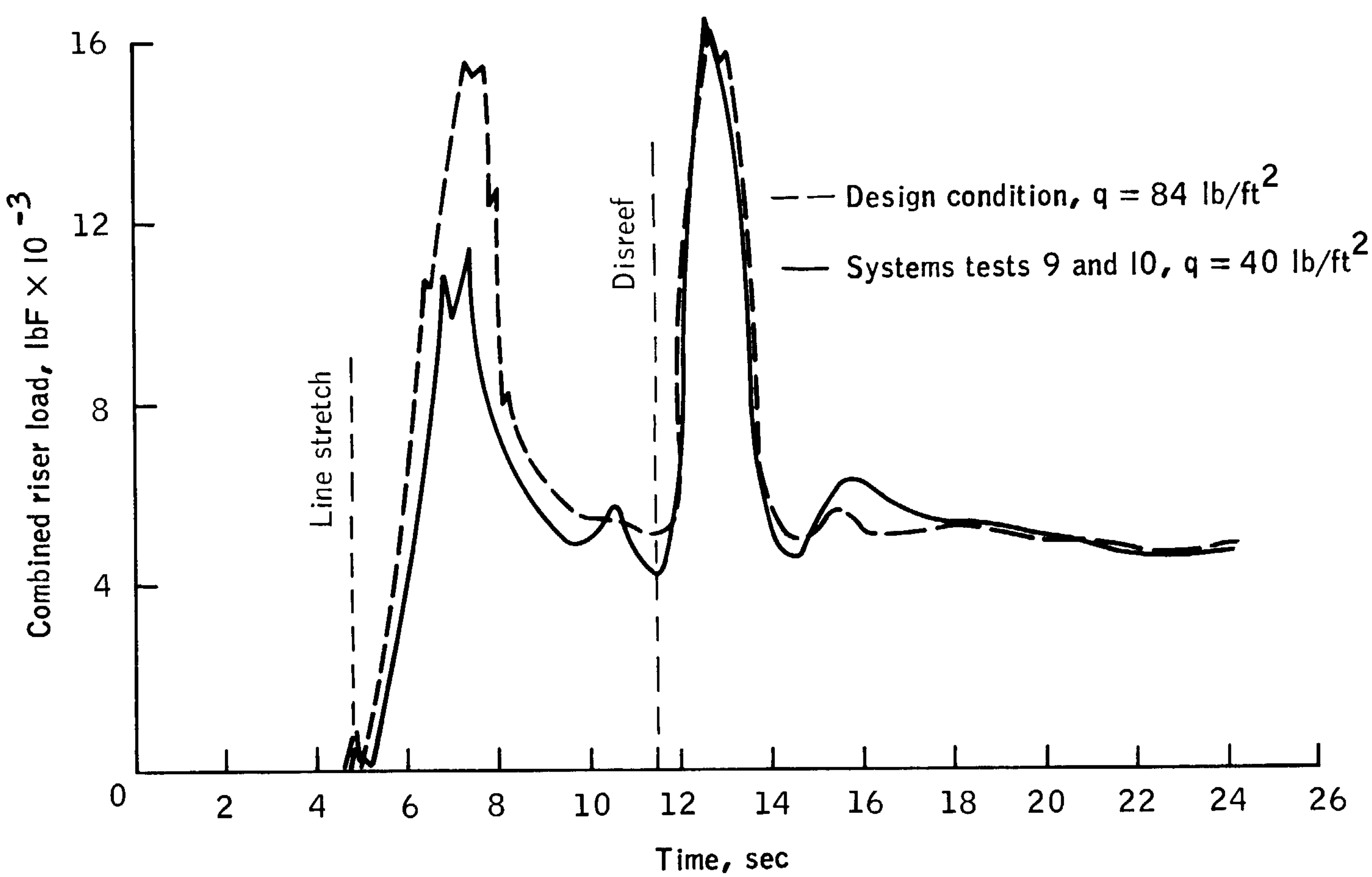



Figure 24.- Parachute opening loads.

NASA-S-66-8810 SEP 19

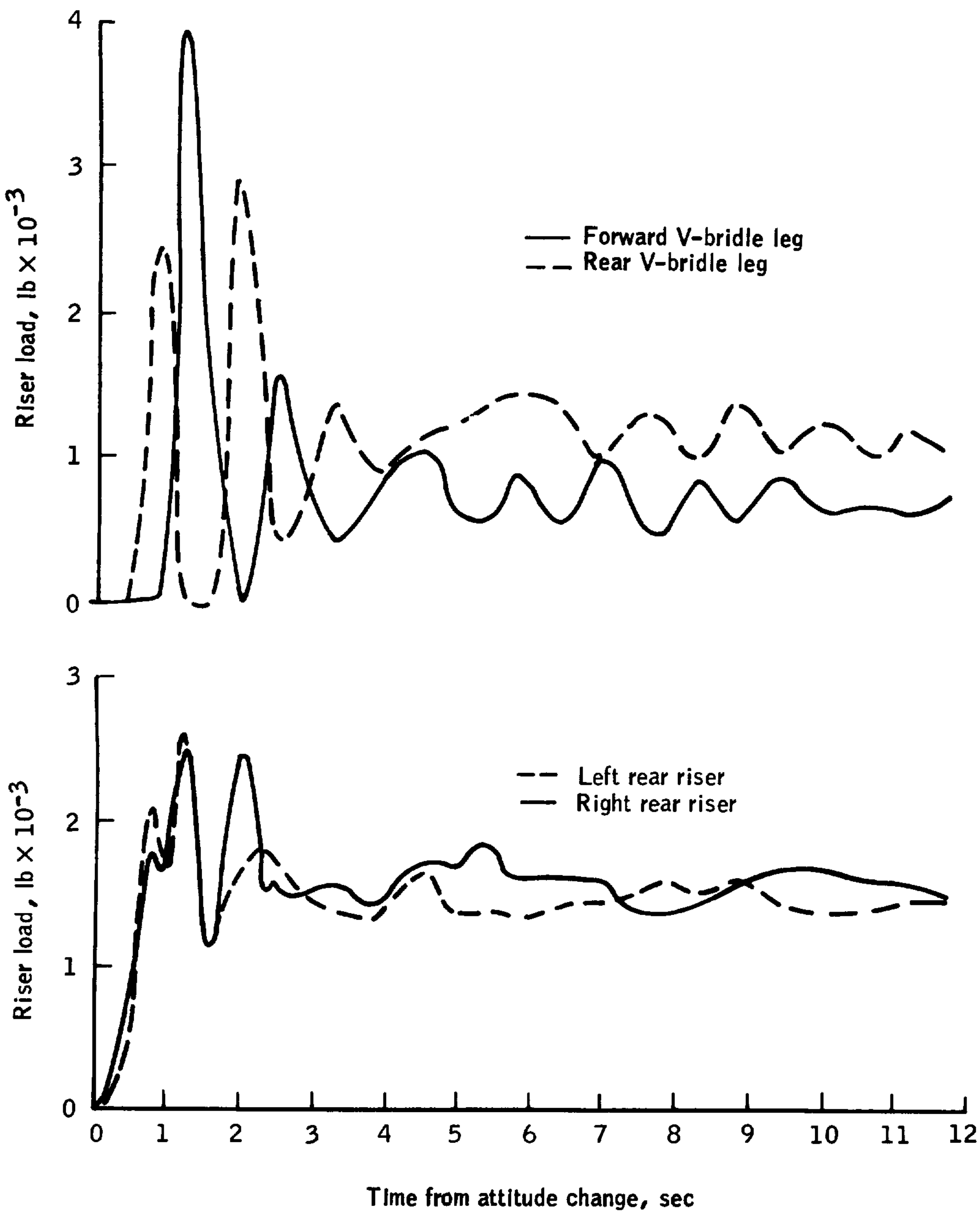



Figure 25.- Attitude-change loads.

NASA-S-66-8809 SEP 19

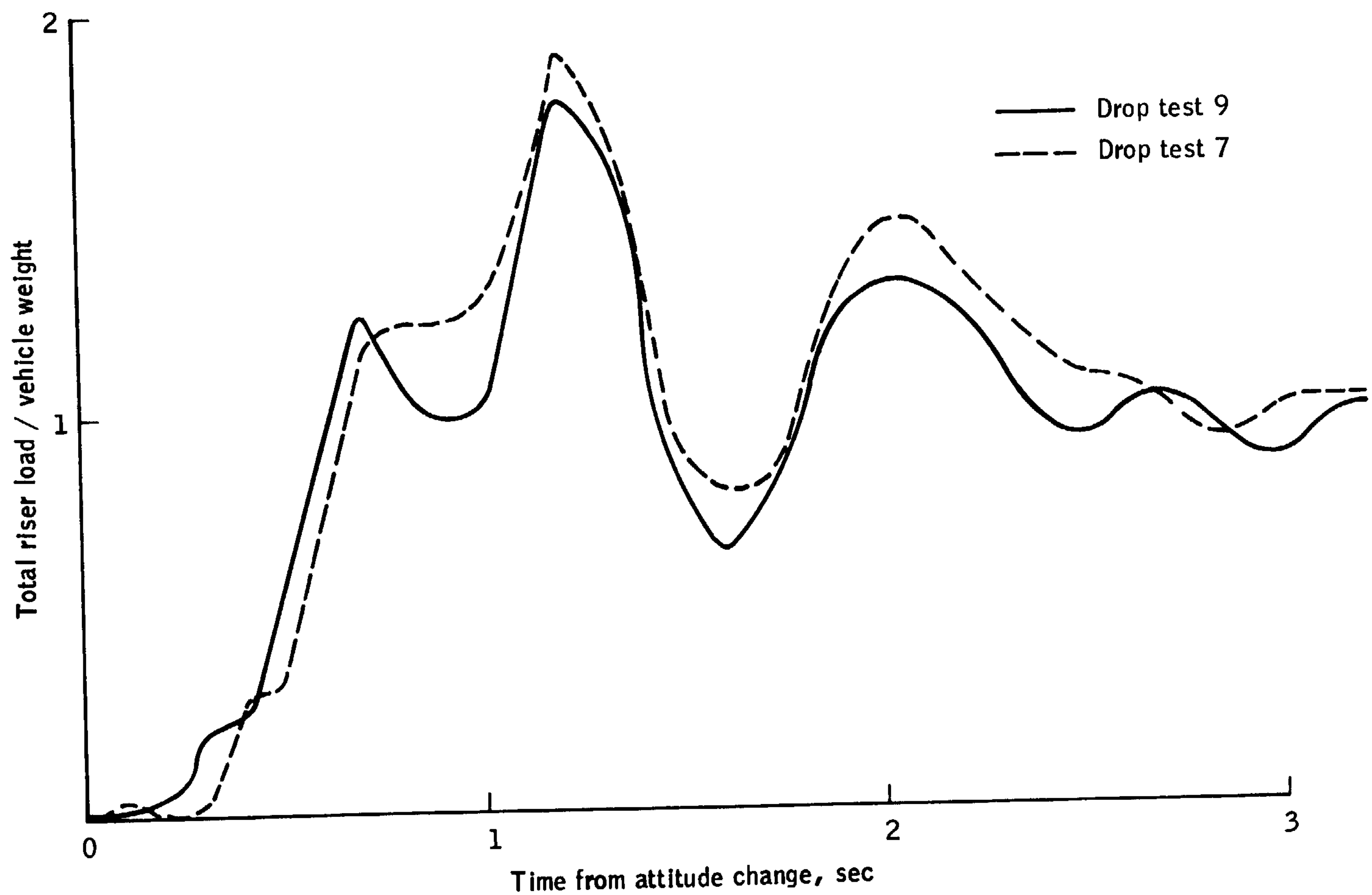



Figure 26.- Attitude-change accelerations.

NASA-S-66-8808 SEP 19

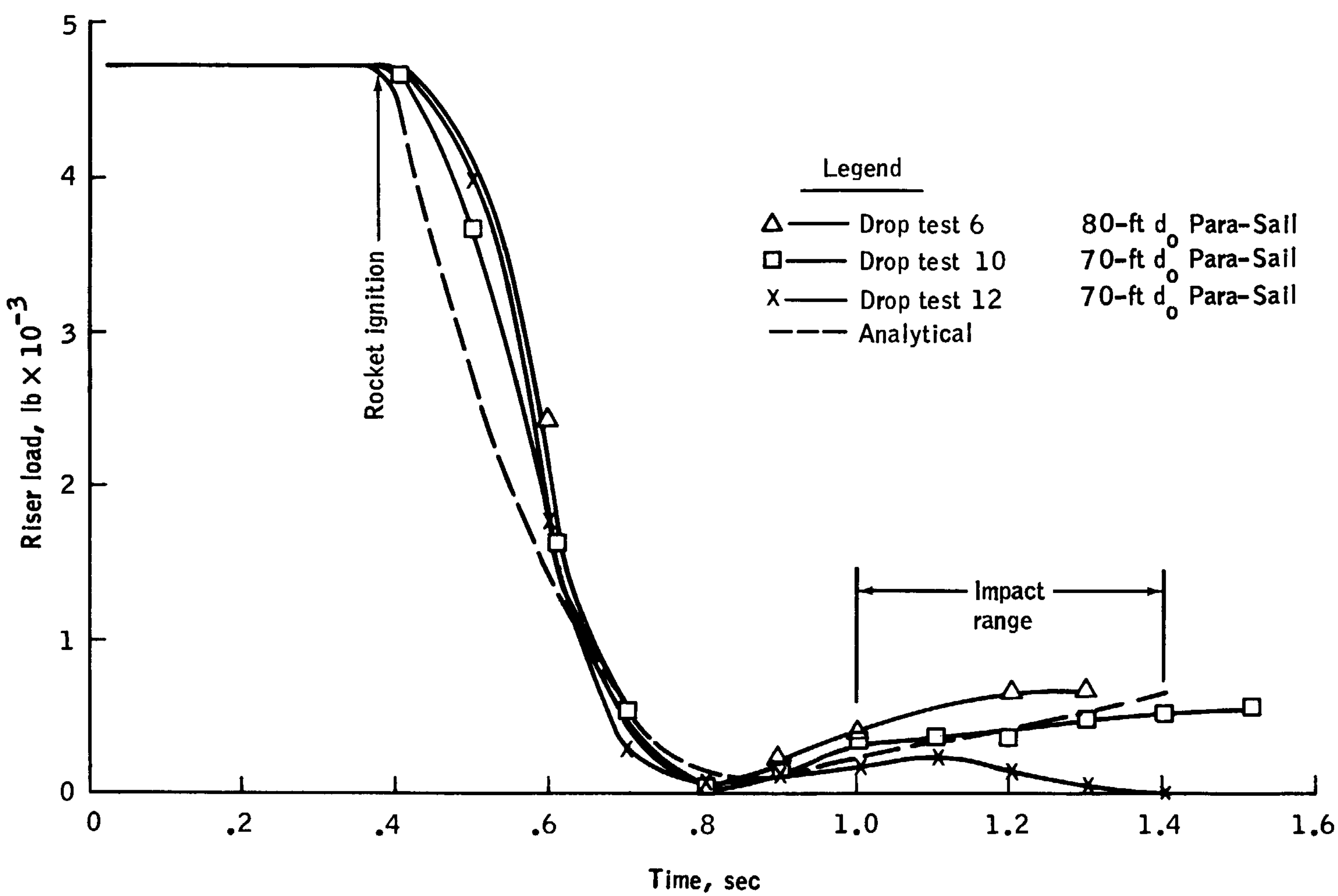



Figure 27.- Riser load during rocket thrust.

NASA-S-66-8807 SEP 19

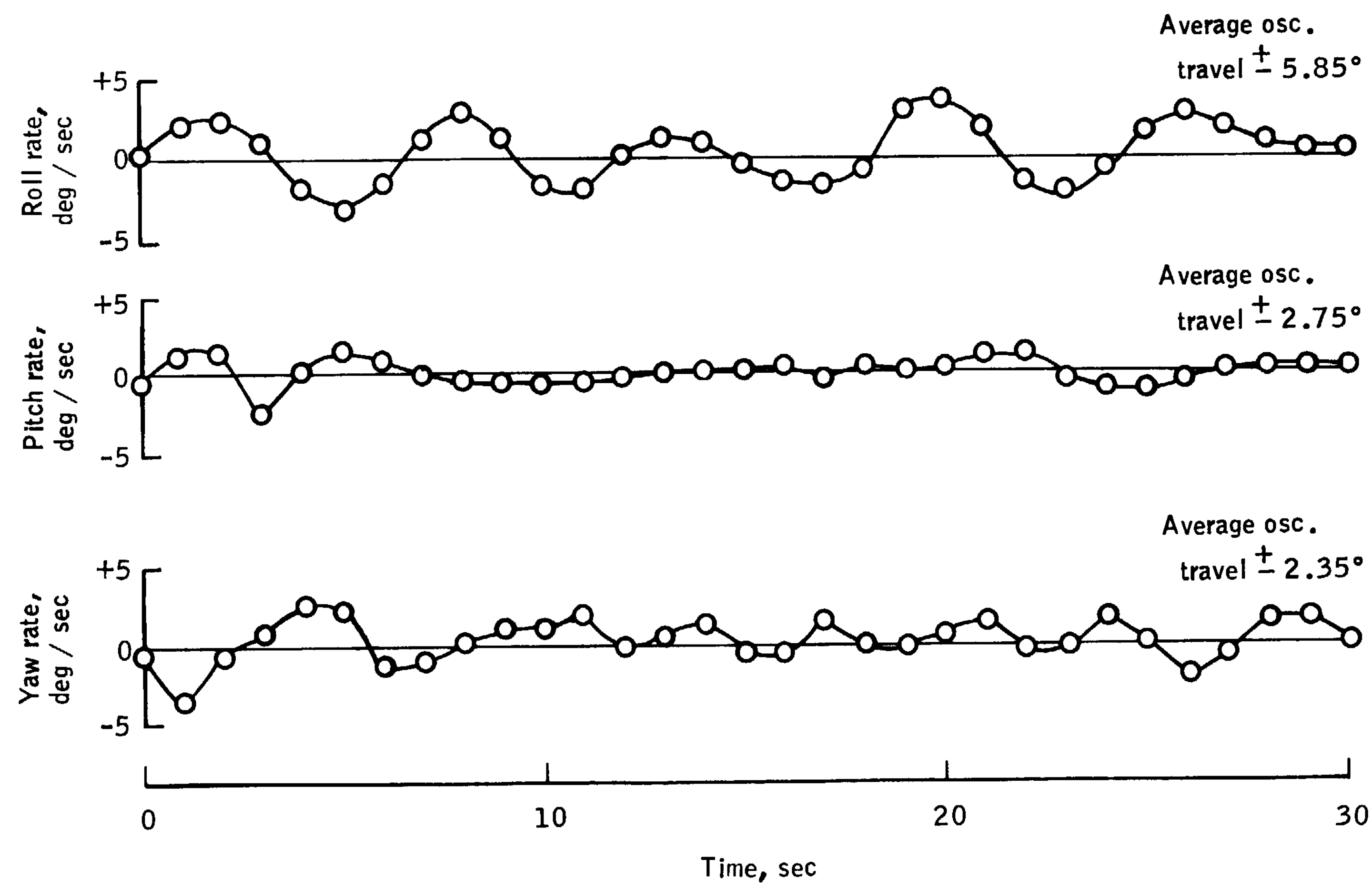



Figure 28.- Descent stability.

NASA-S-66-8806 SEP 19

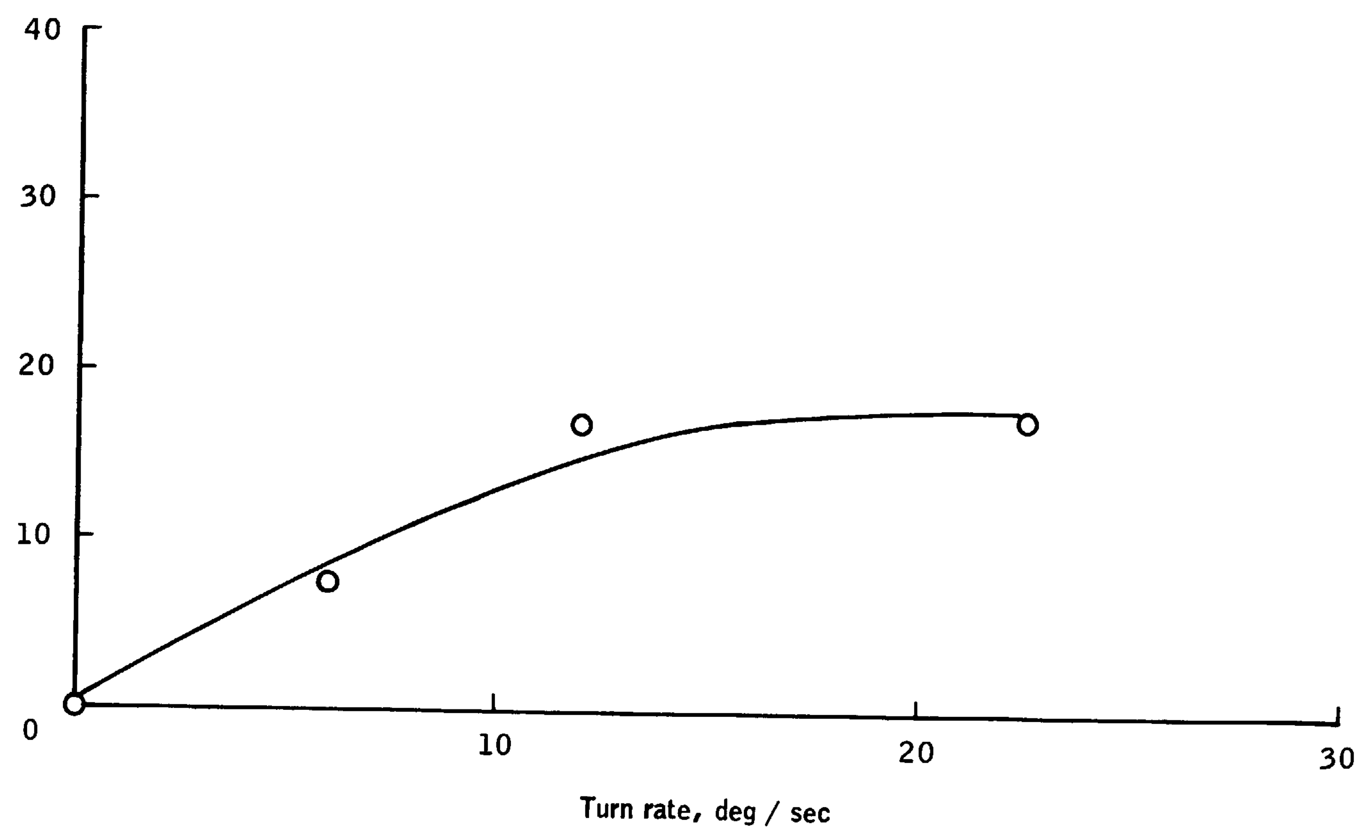



Figure 29.- Bank angle during turn.

NASA-S-66-8805 SEP 19

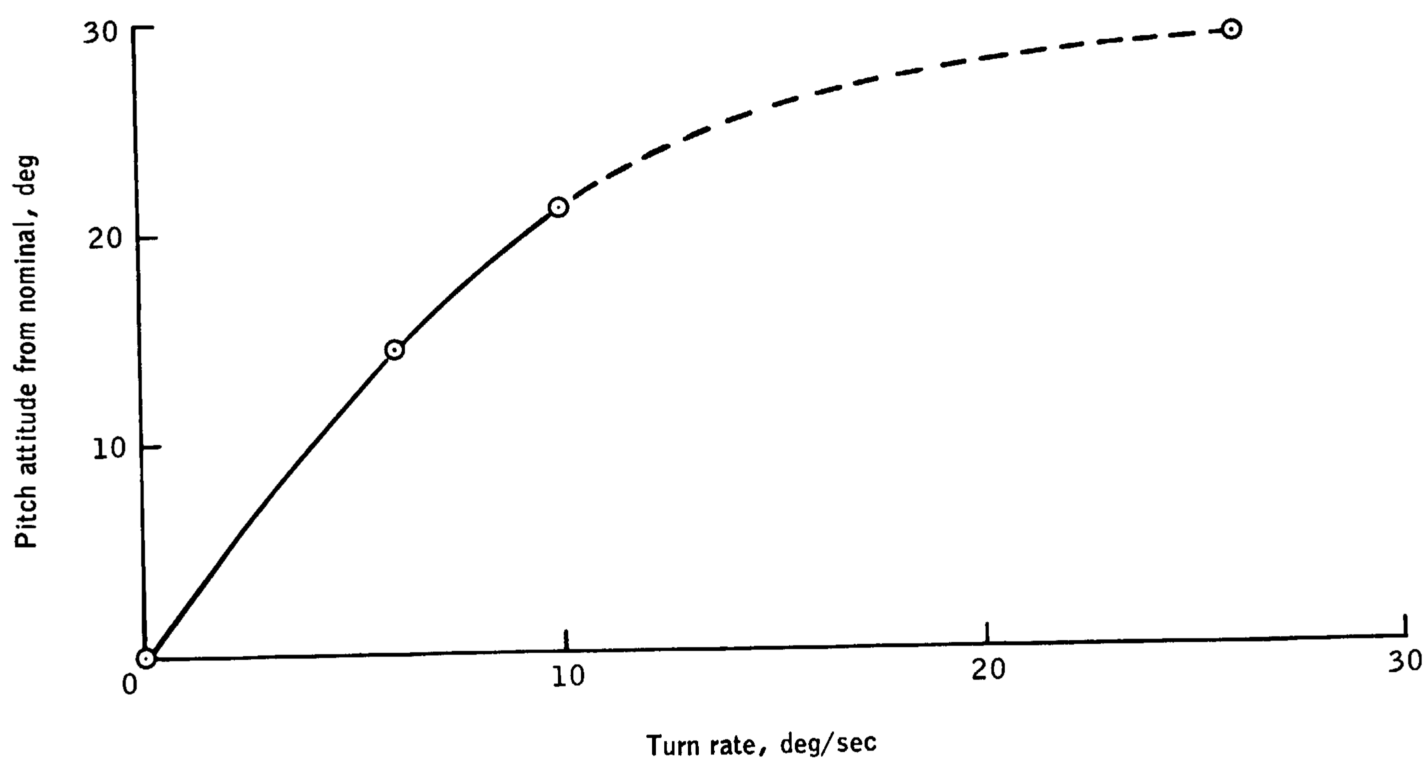



Figure 30.- Pitch angle during turn.

NASA-S-66-8804 SEP 19

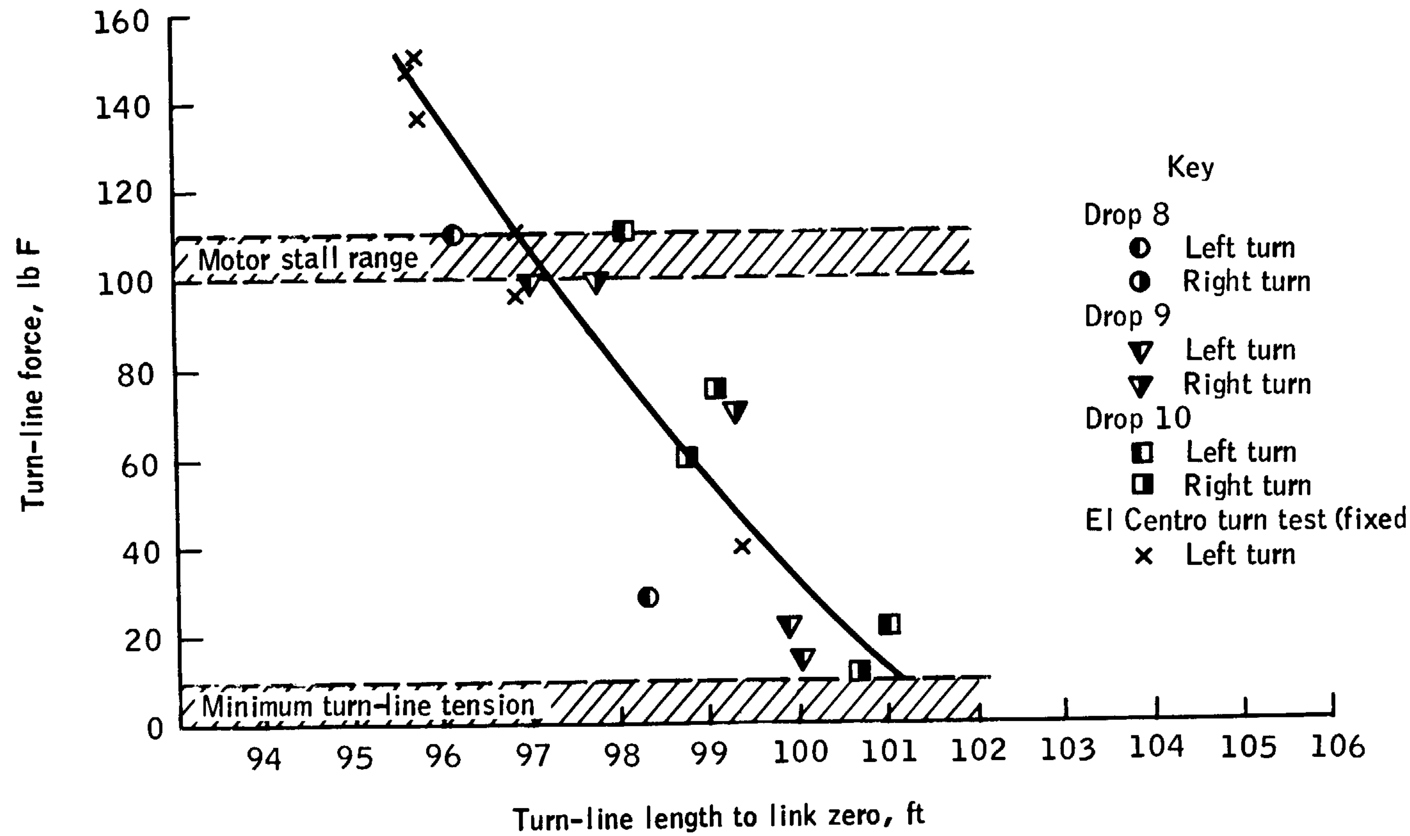



Figure 31.- Turn-line load-stroke characteristics.

NASA-S-66-8803 SEP 19

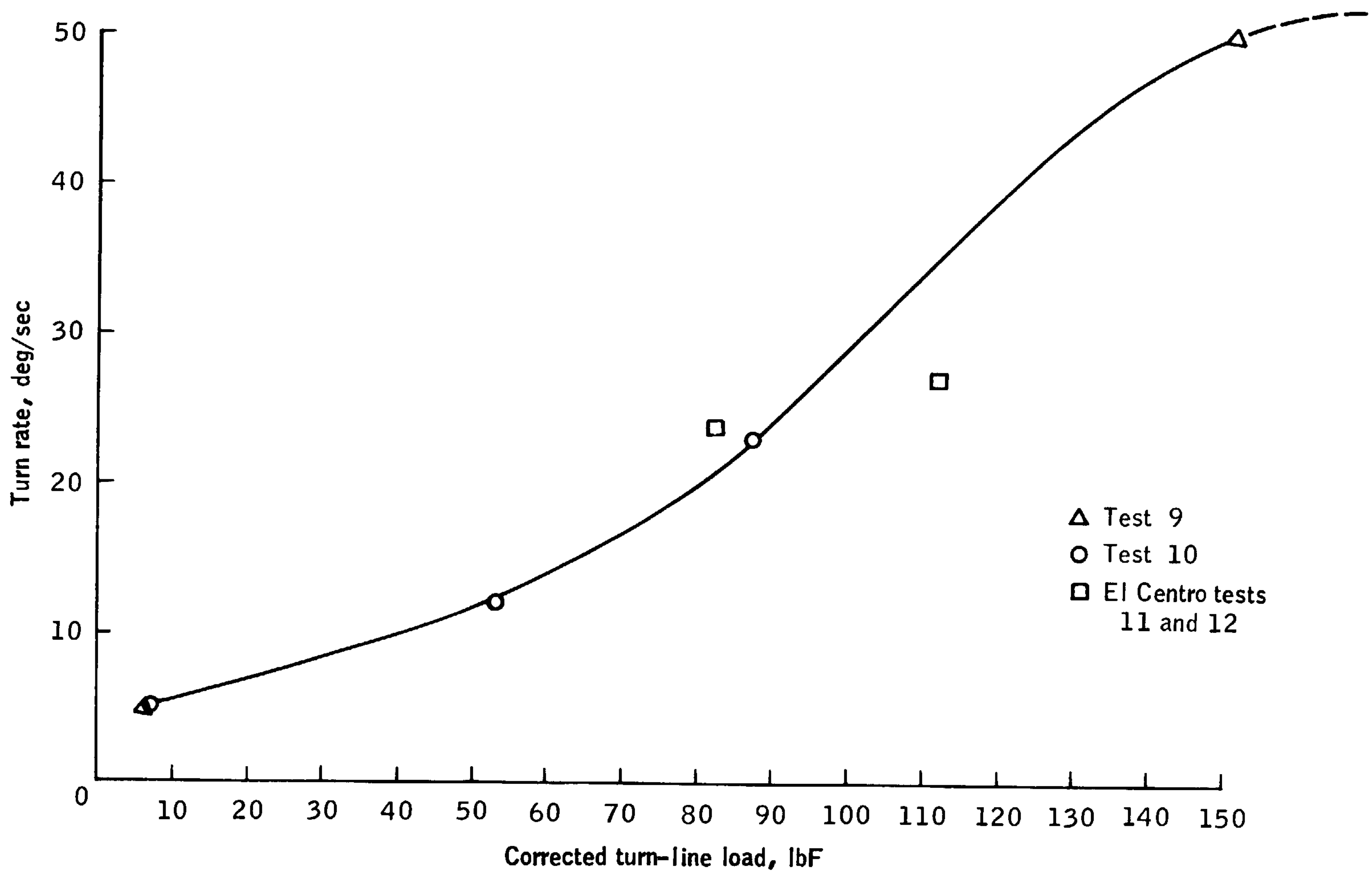



**Figure 32.- Available turn rate.**

NASA-S-66-8802 SEP 19

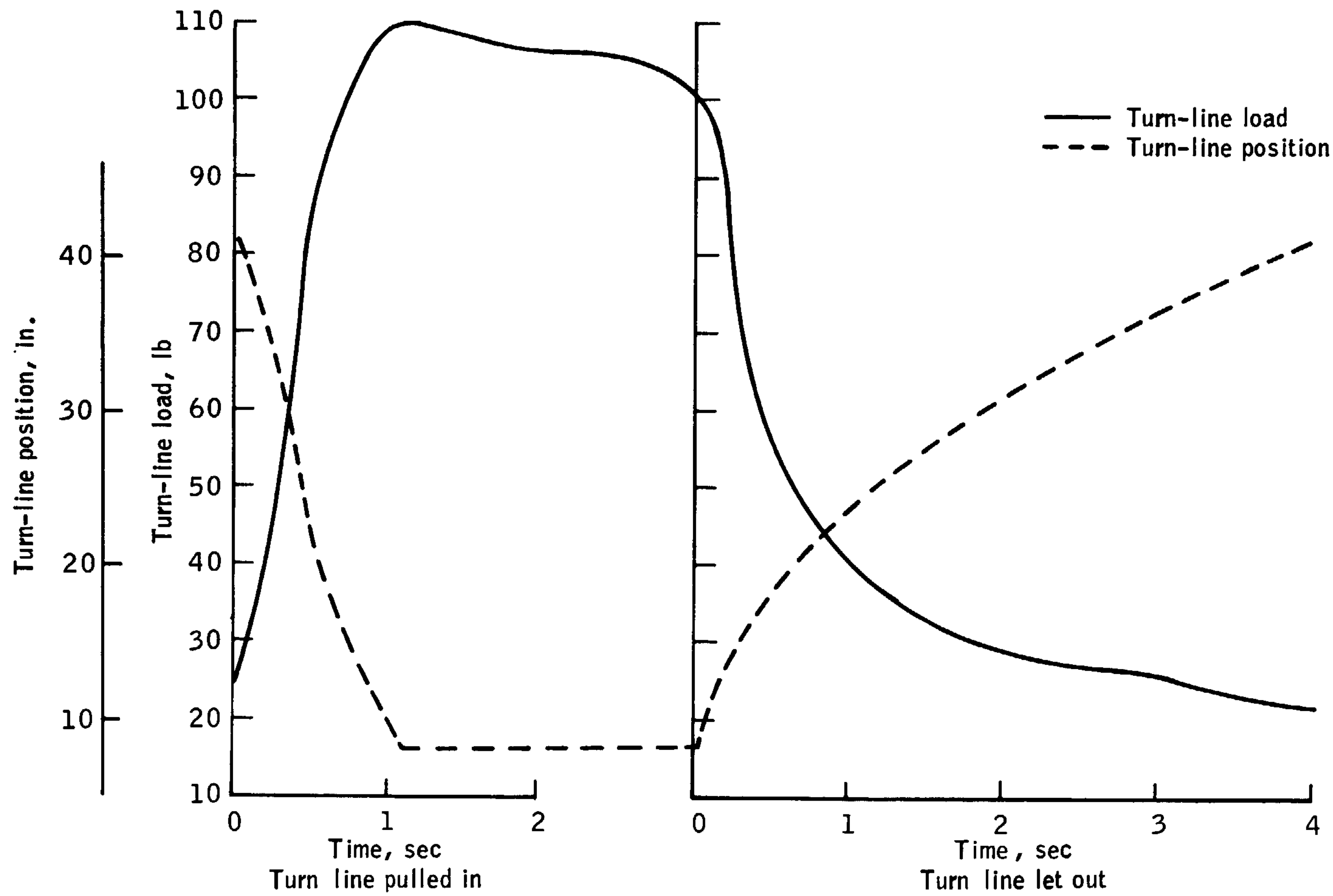



Figure 33.- Force and travel-time characteristics.

NASA-S-66-8801 SEP 19

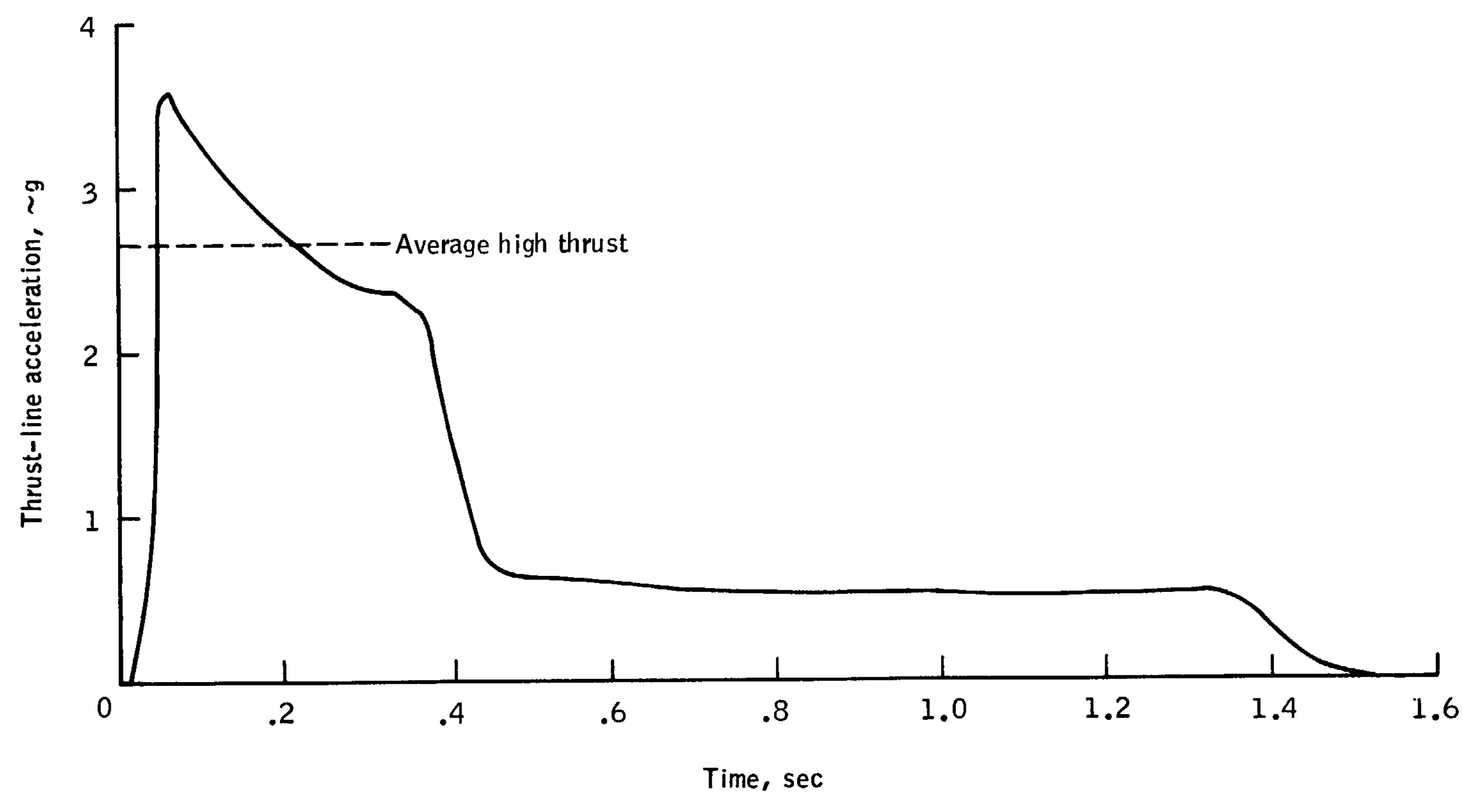



Figure 34.- Nominal rocket-system acceleration history.

NASA-S-66-8800 SEP 19

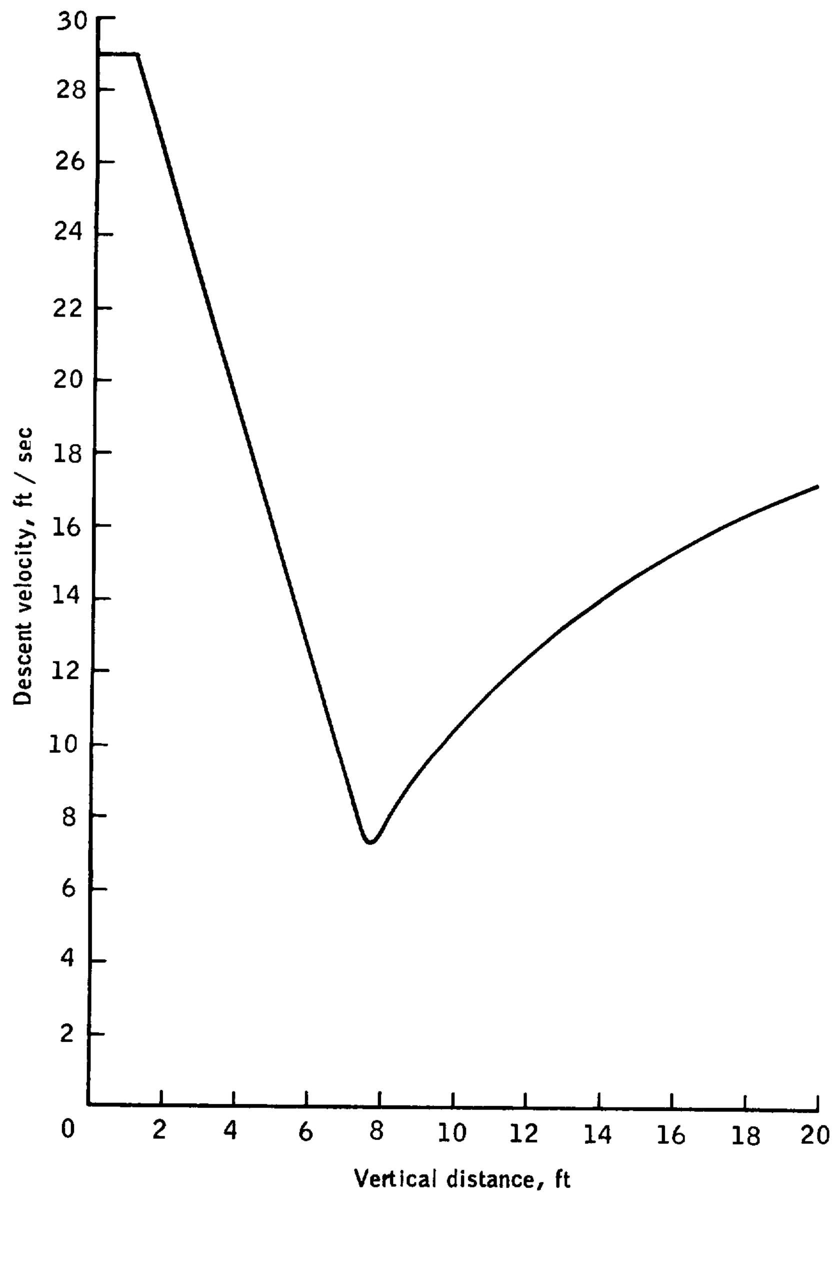



Figure 35.- Nominal velocity decay during rocket fire.

NASA-S-66-8799 SEP 19

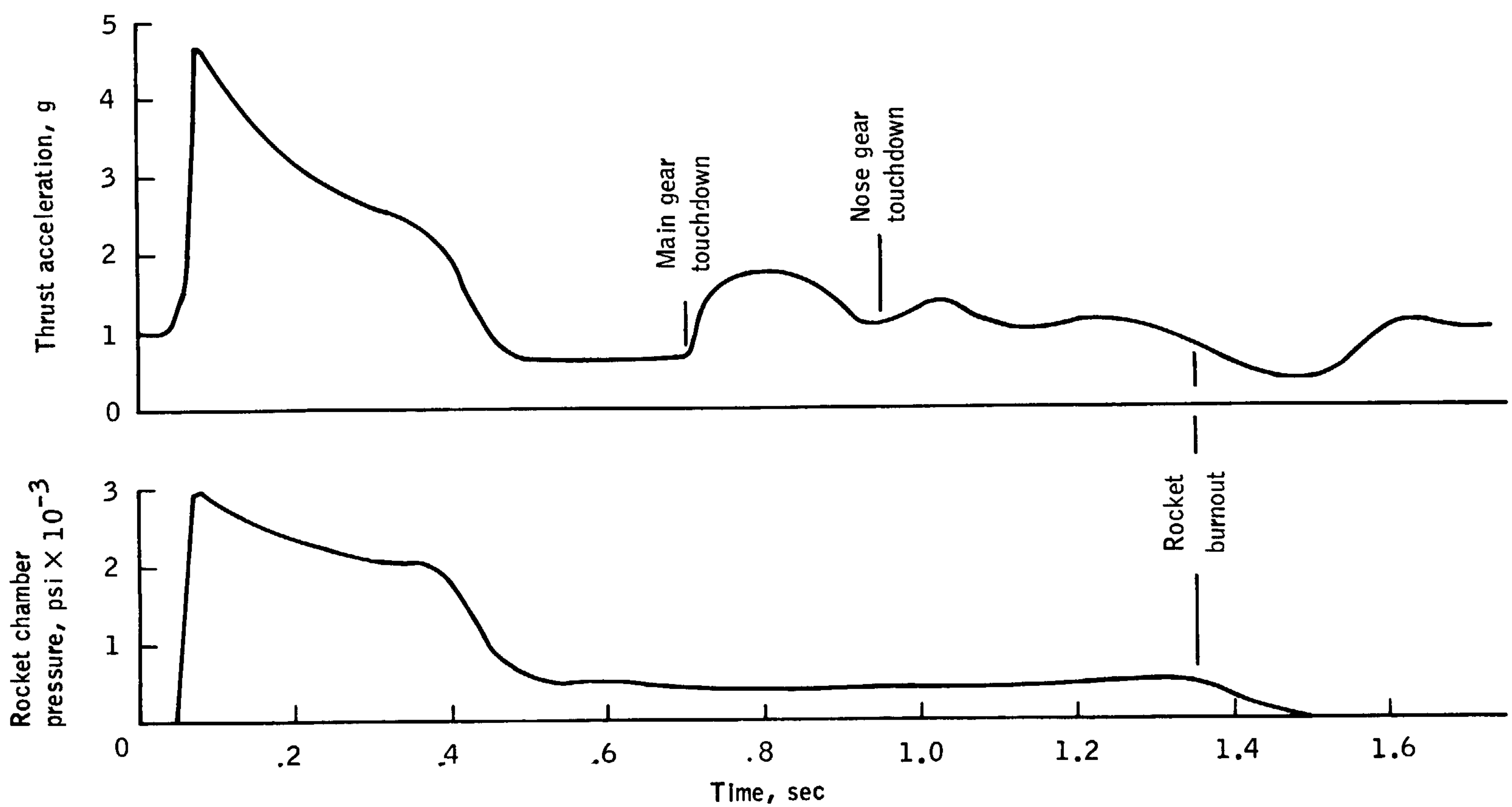



Figure 36.- Test 12, landing accelerations, thrust line.

NASA-S-66-8798 SEP 19

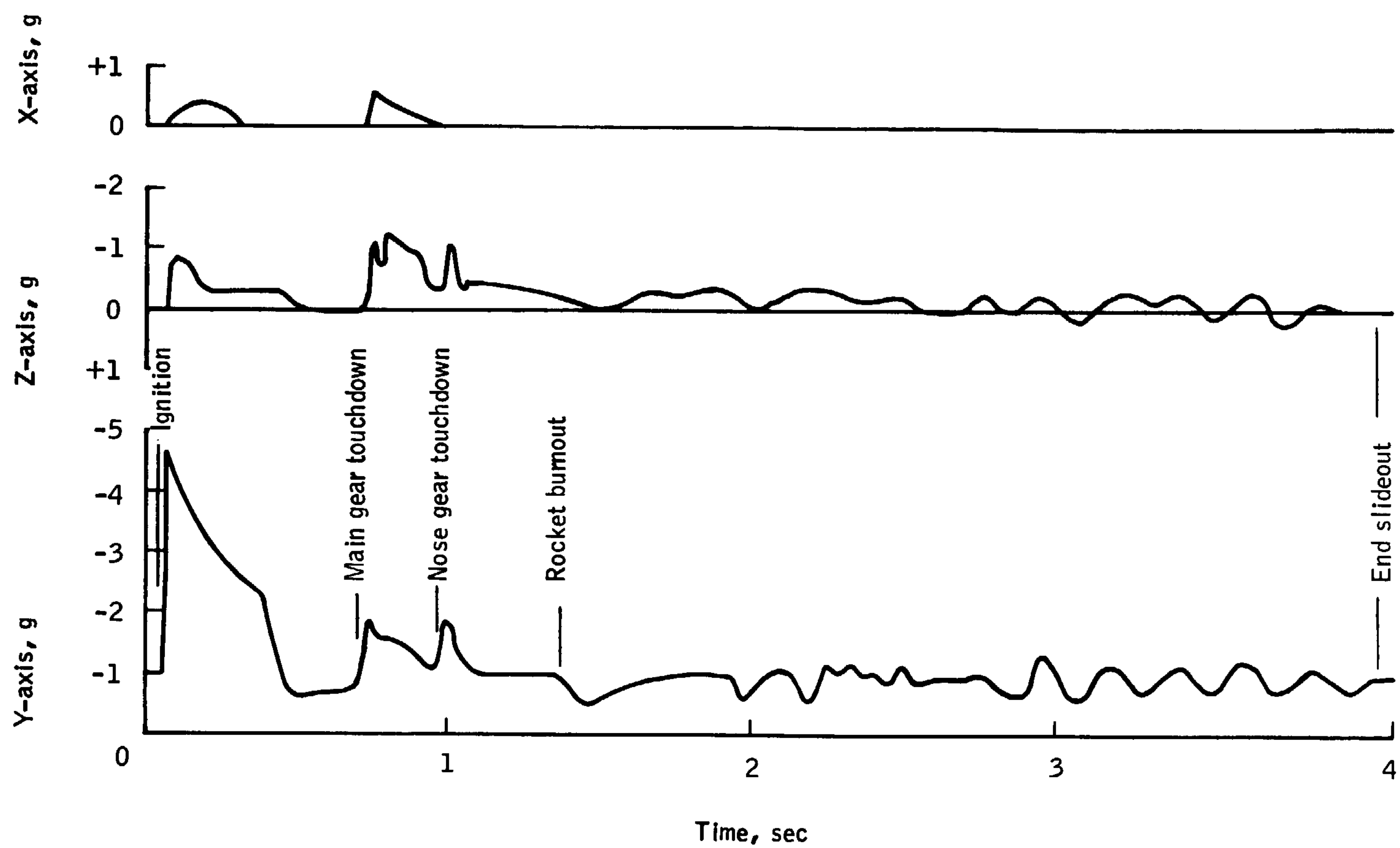



Figure 37.- Test 12, landing accelerations, three axes.

NASA-S-66-8829 SEP 19

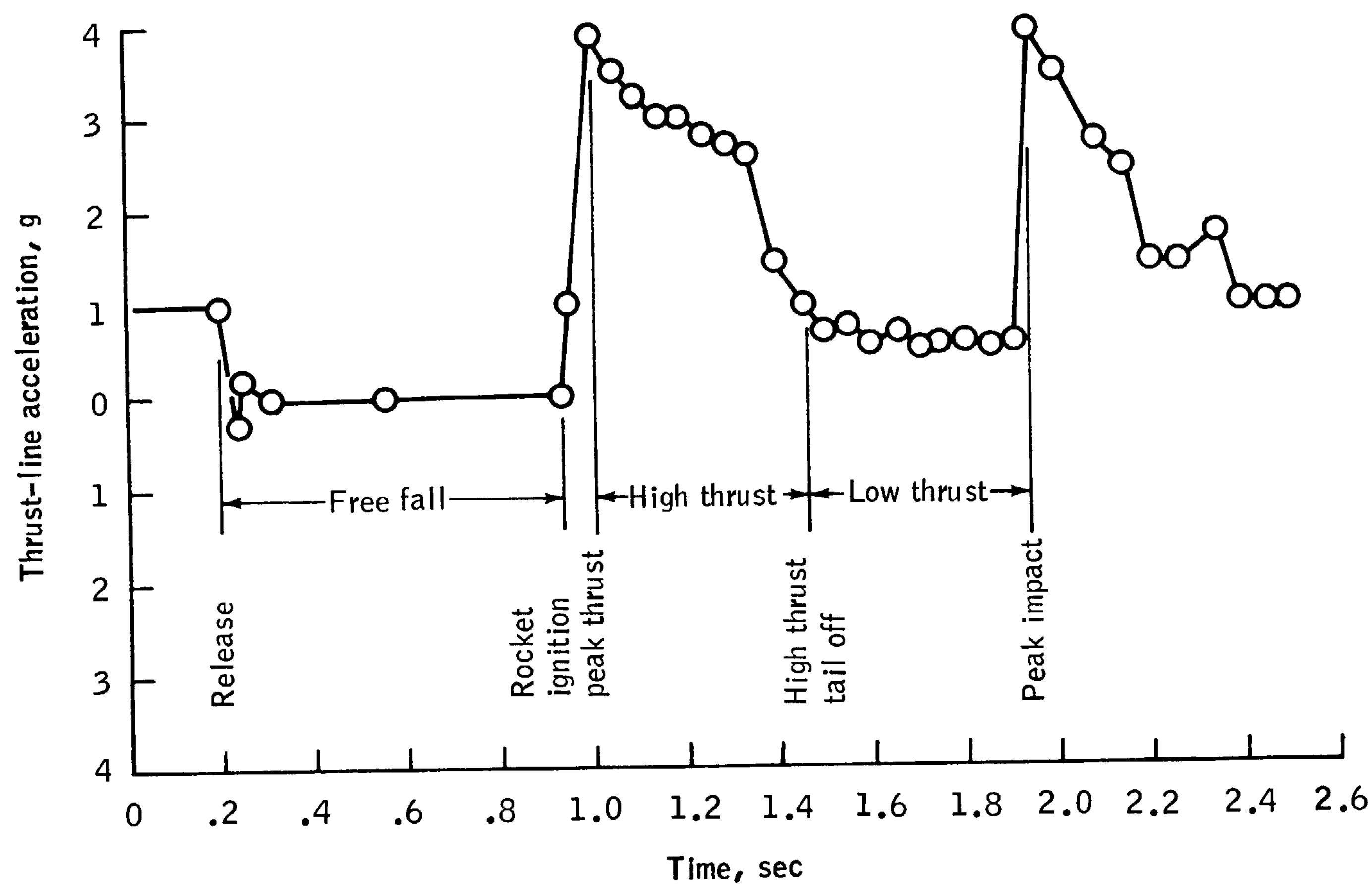



Figure 38.- Crane drop II, landing accelerations, thrust line.

NASA-S-66-8790 SEP 19

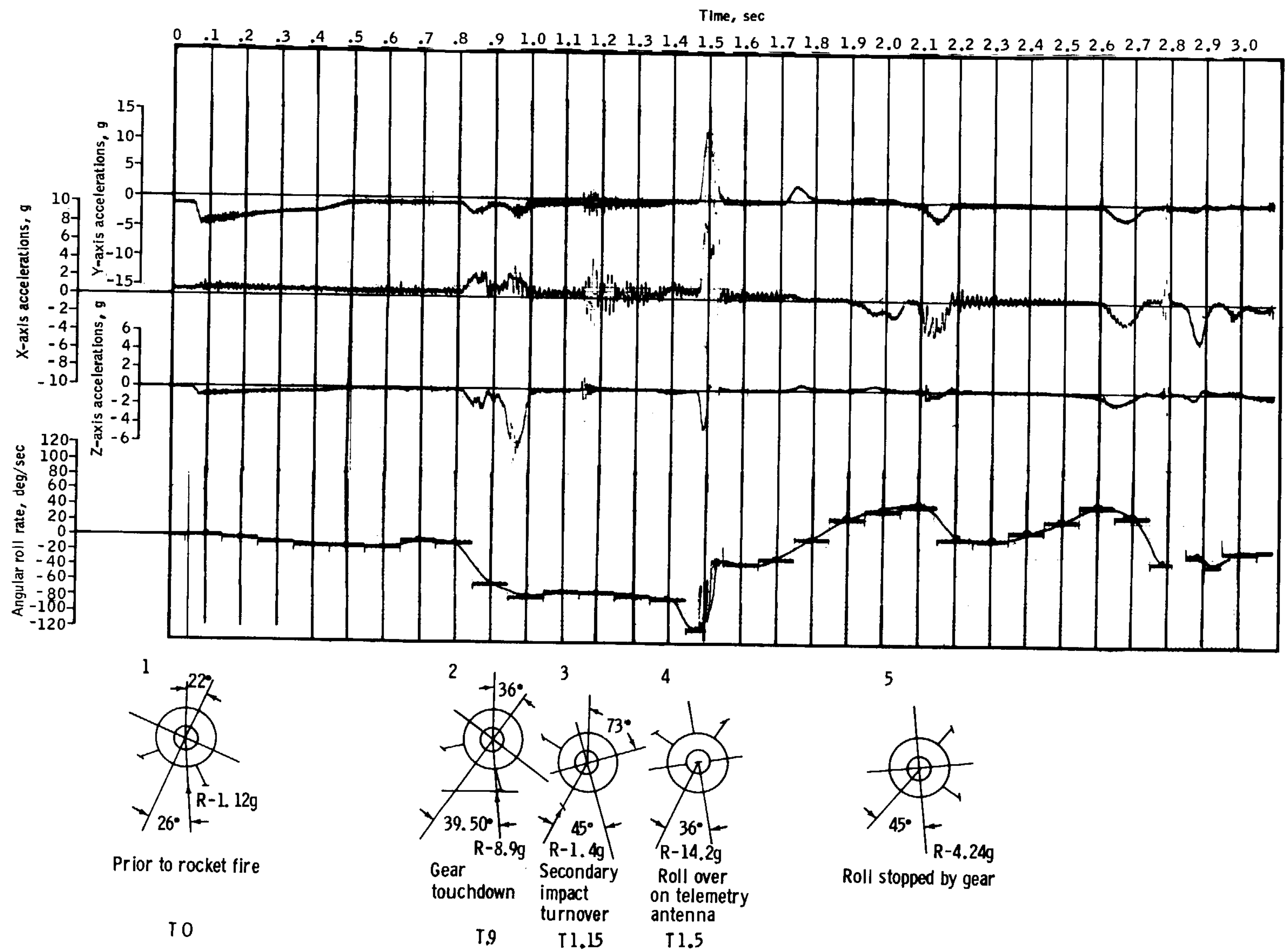



Figure 39.- Test 10, landing accelerations.

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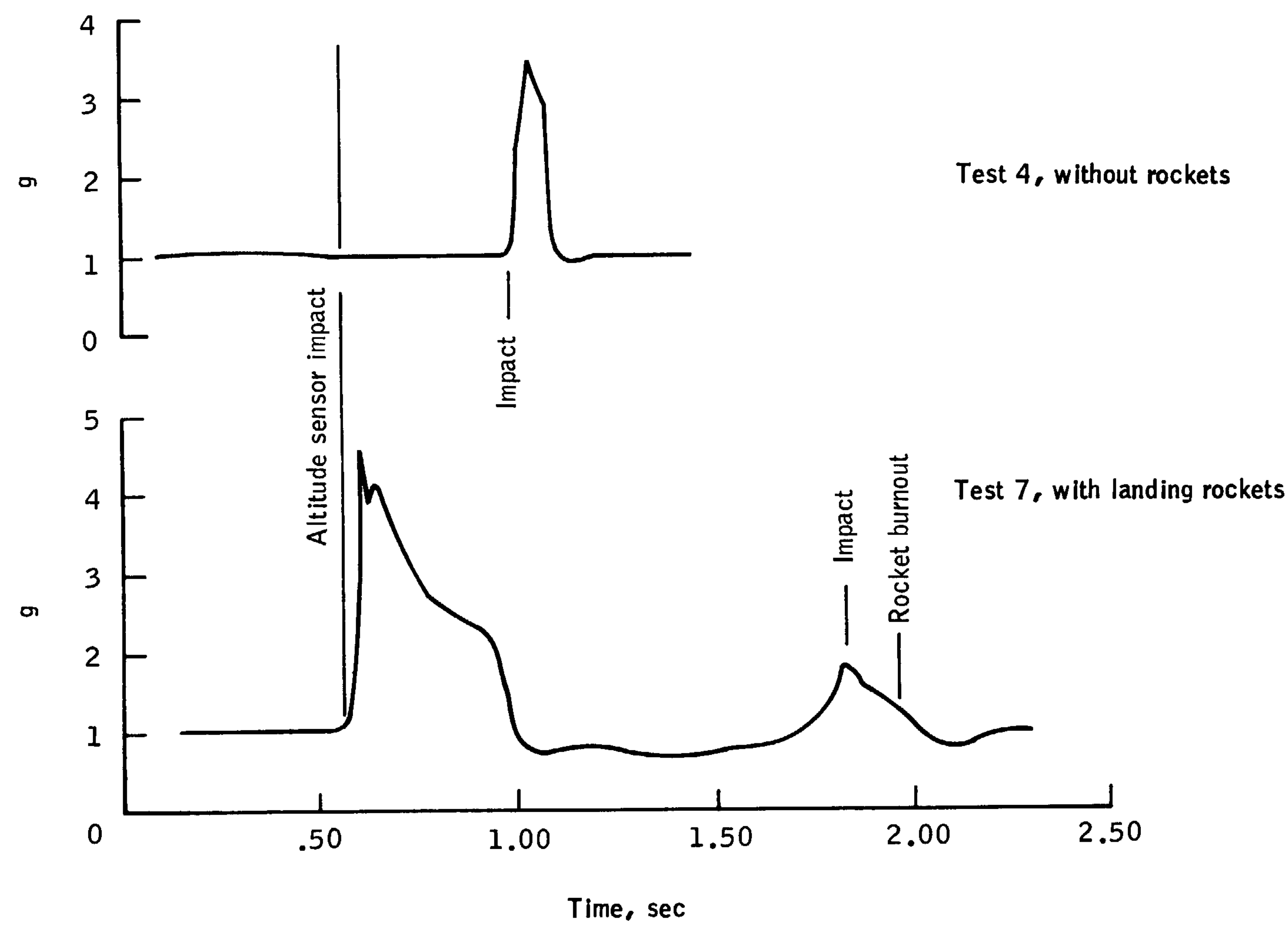


Figure 40.- Landing acceleration, water landings.

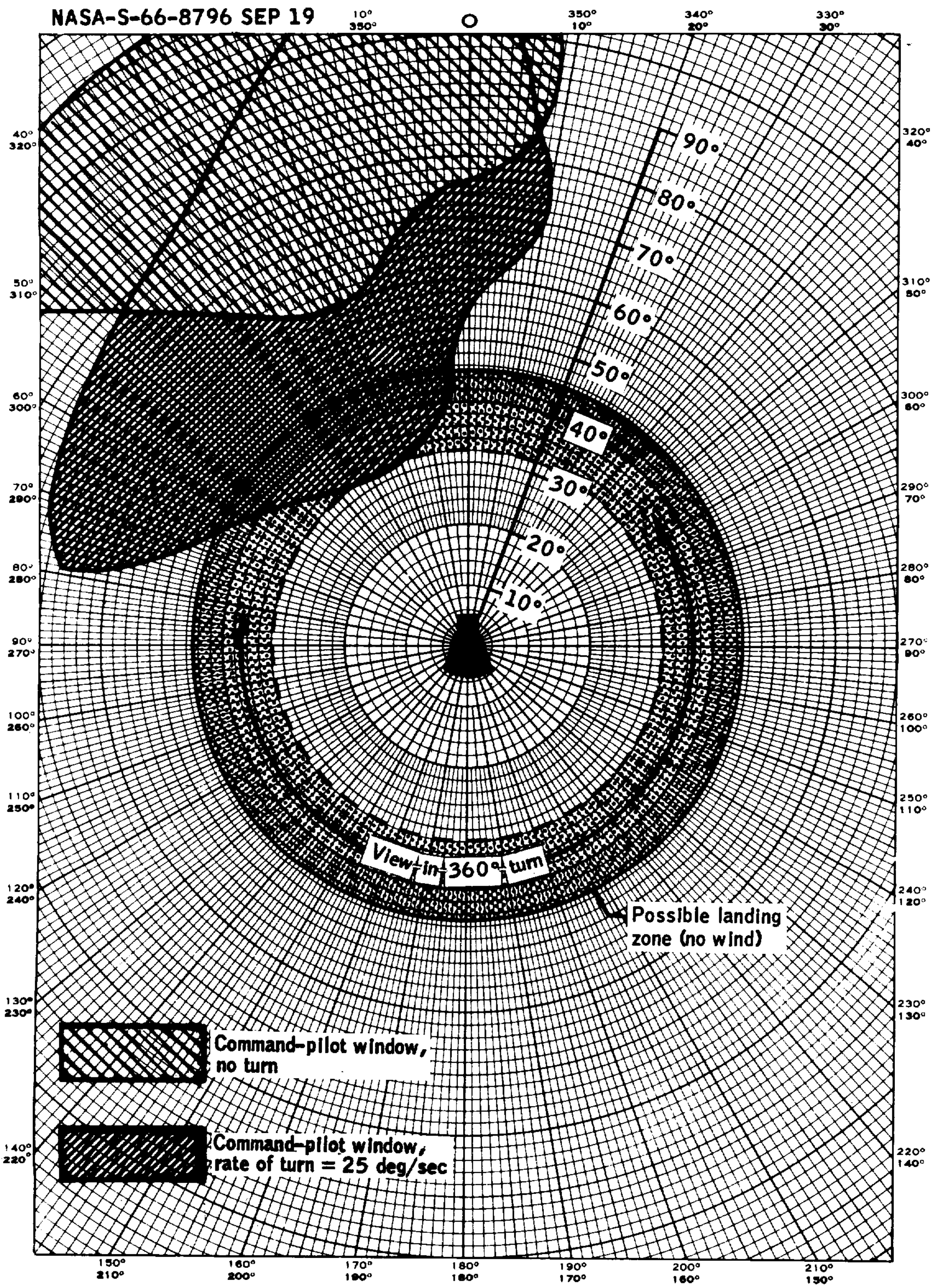


Figure 41.- Command-pilot window, field of view.

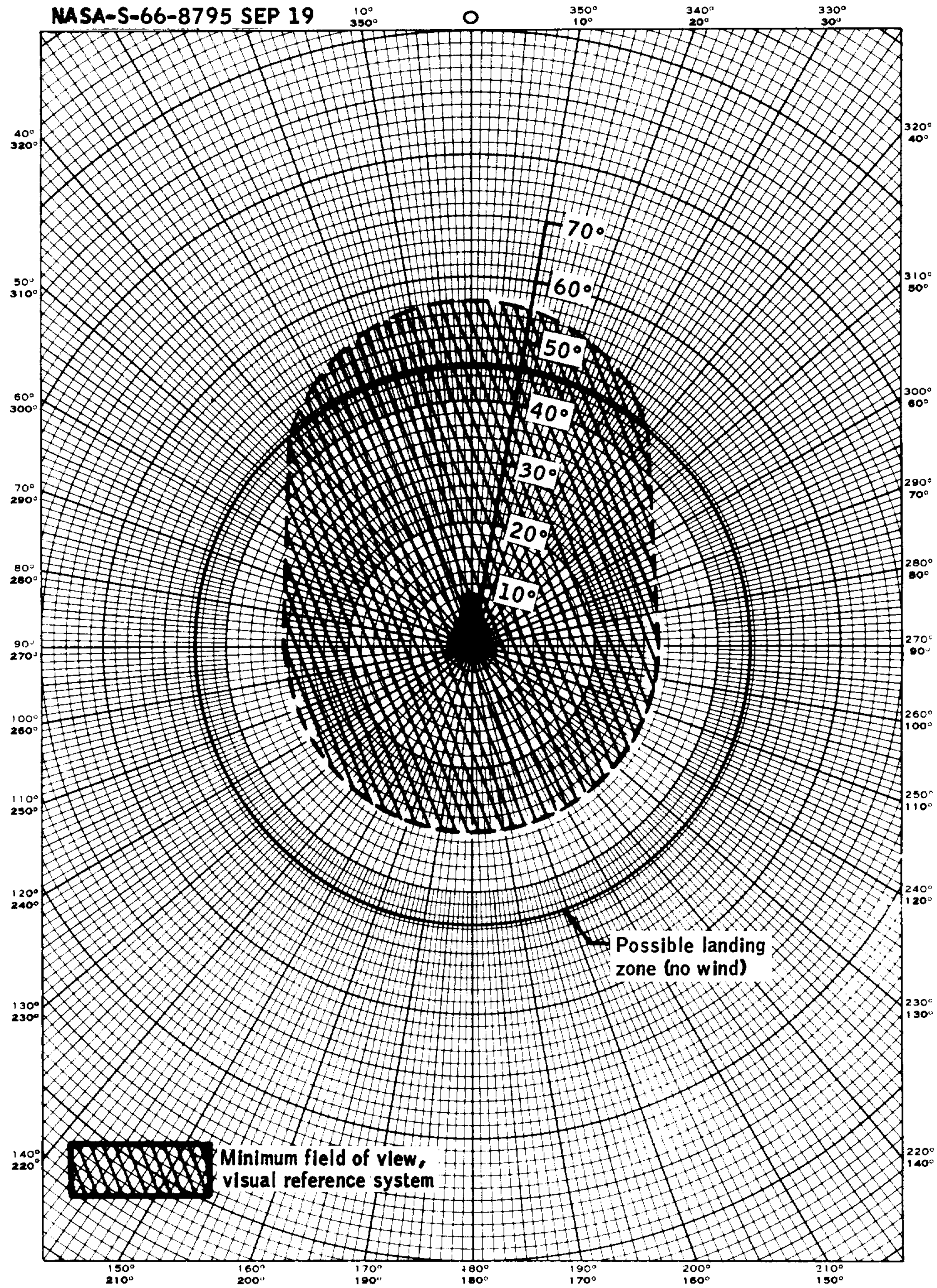


Figure 42.- Visual reference system, field of view.

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Figure 43.- Rocket alinement, exterior view.

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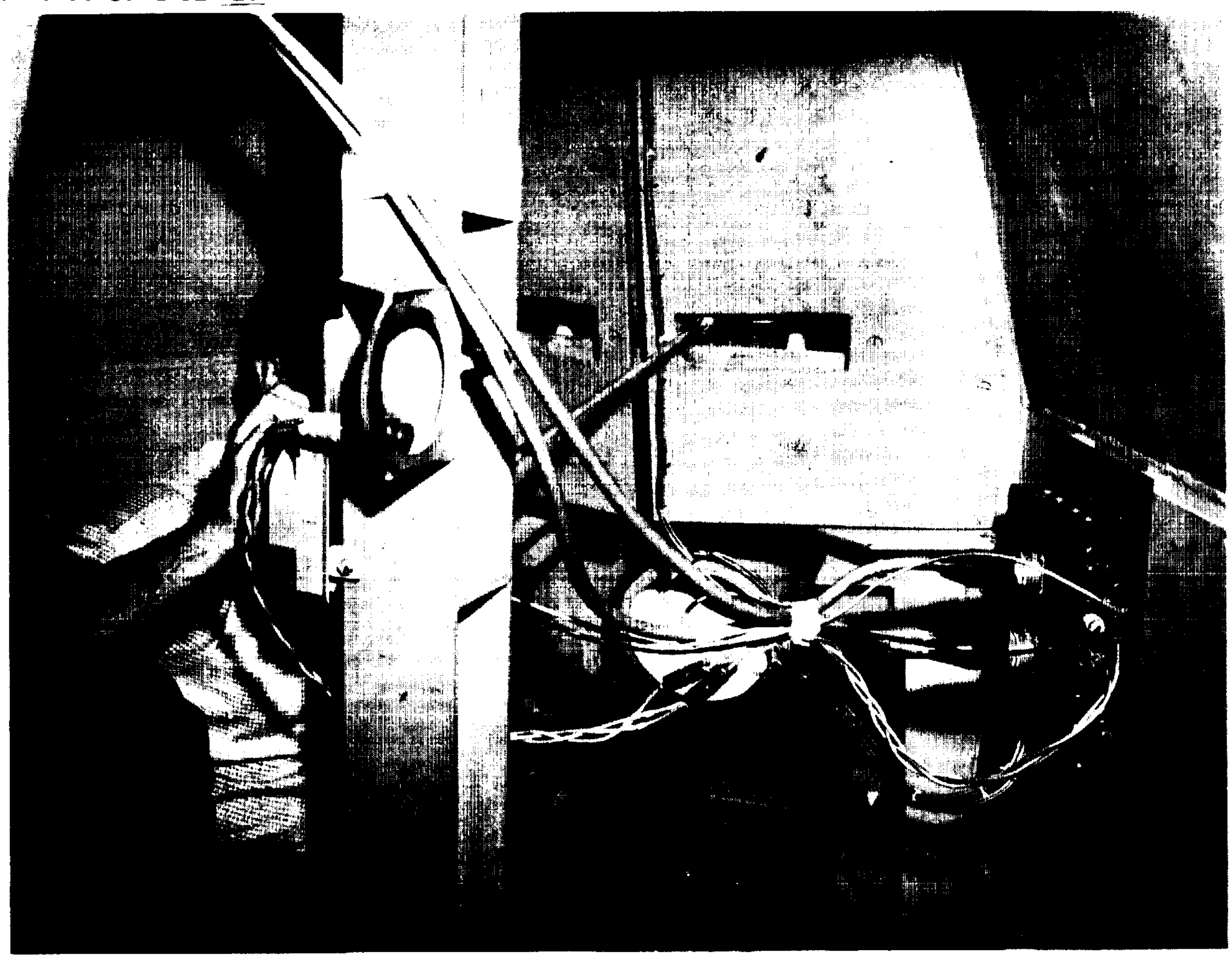

Figure 44.- Rocket alinement, interior view.

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Figure 45.- Crane drop II, erosion crater.

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Figure 46.- Test 12, erosion path.