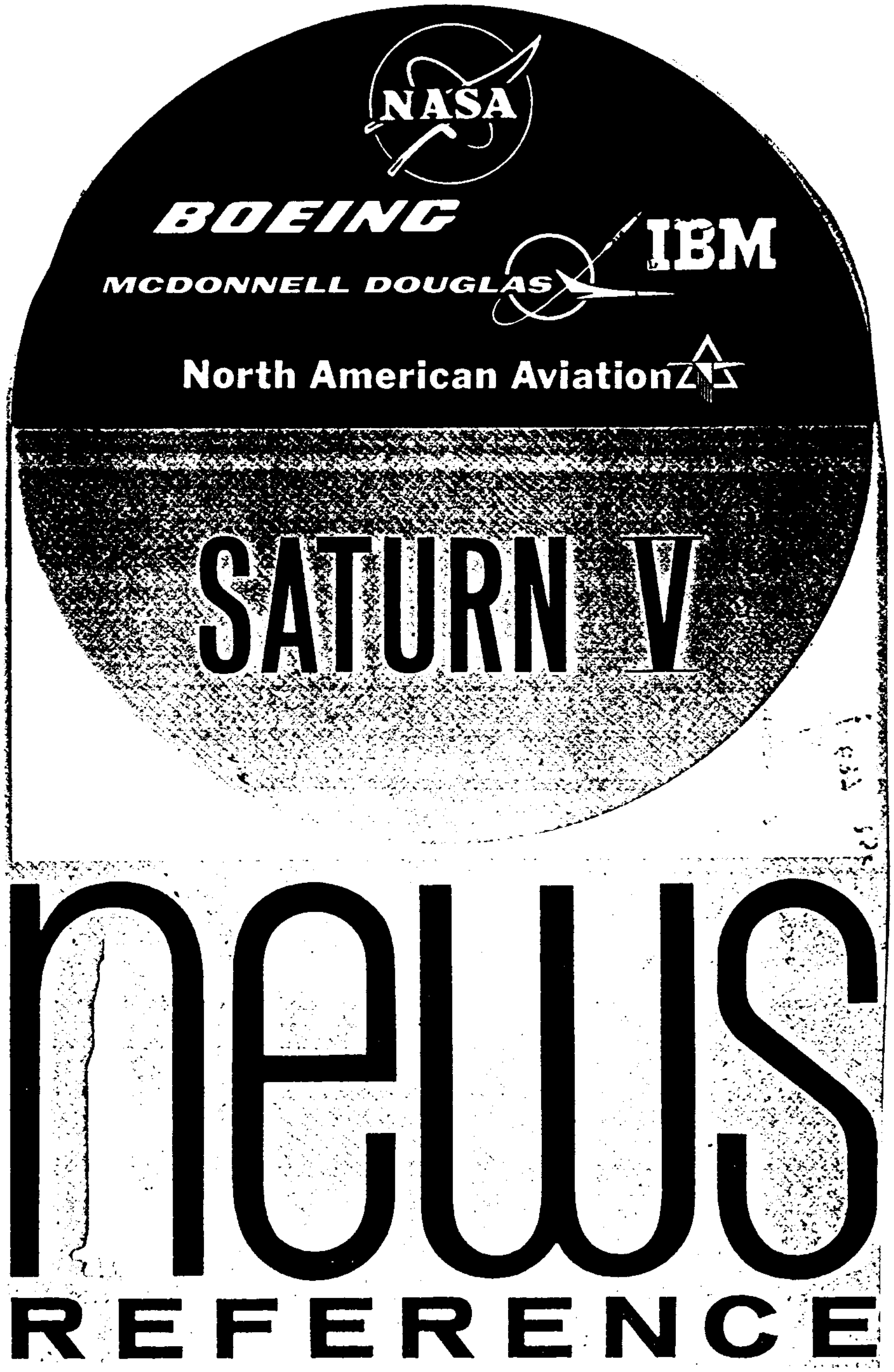


NASA

BOEING

IBM

MCDONNELL DOUGLAS

North American Aviation

SATURN V

news REFERENCE

ADDENDUM

Since the printing of this book, additional refinement of data has indicated that the overall height of the Saturn V should be 363 feet, not 364 as stated in the body of the publication.

CHANGED DECEMBER 1968

SATURN V NEWS REFERENCE

AUGUST 1967

National Aeronautics and Space Administration
  George C. Marshall Space Flight Center
  John F. Kennedy Space Center

The Boeing Company
  Launch Systems Branch

McDonnell Douglas Astronautics Company

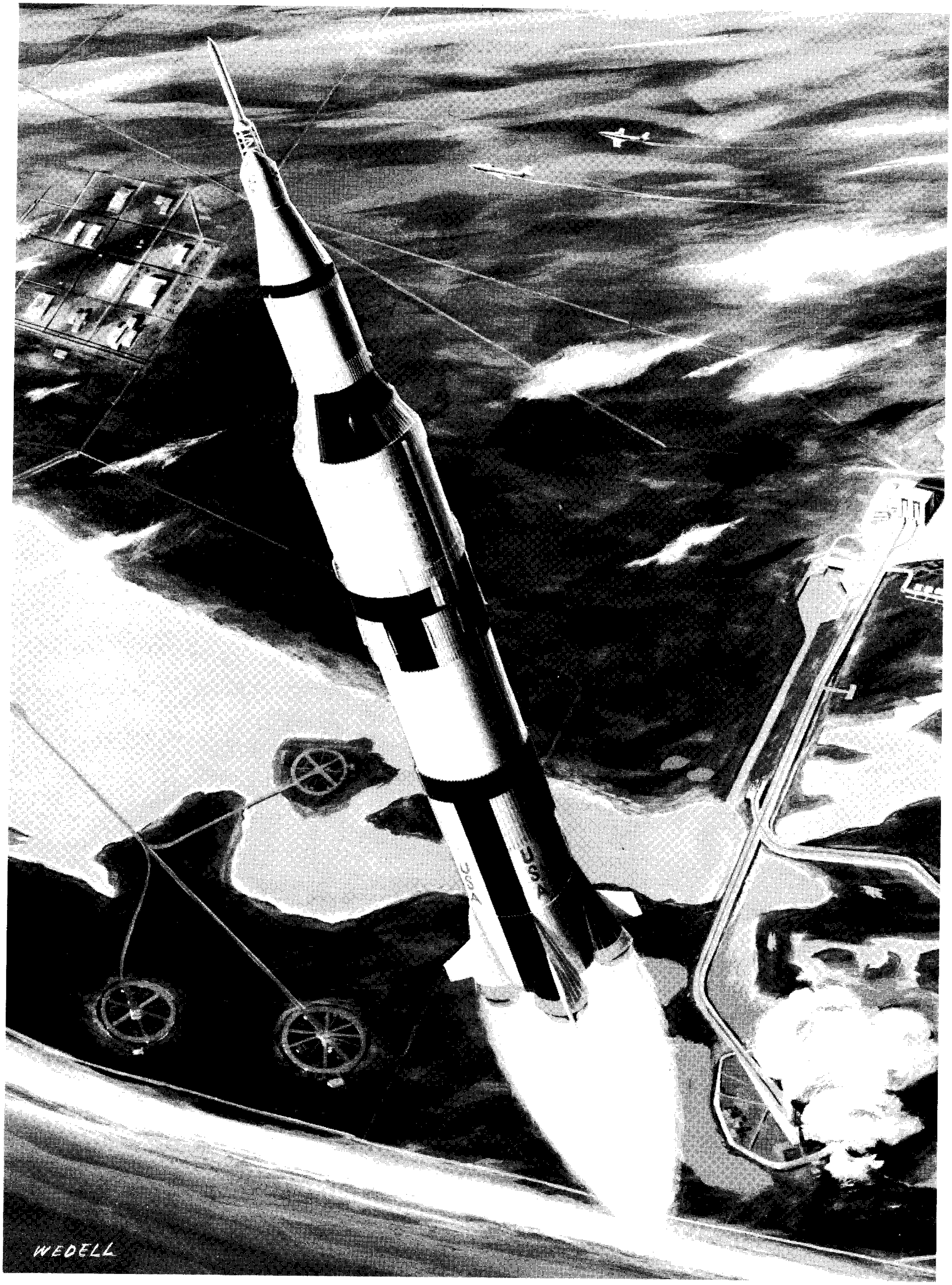
International Business Machines Corporation
  Federal Systems Division

Rocketdyne Division
  North American Rockwell Corporation

Space Division
  North American Rockwell Corporation

Printed in U. S. A.

"Educational Liaison Branch,
Public Affairs Office"


USA
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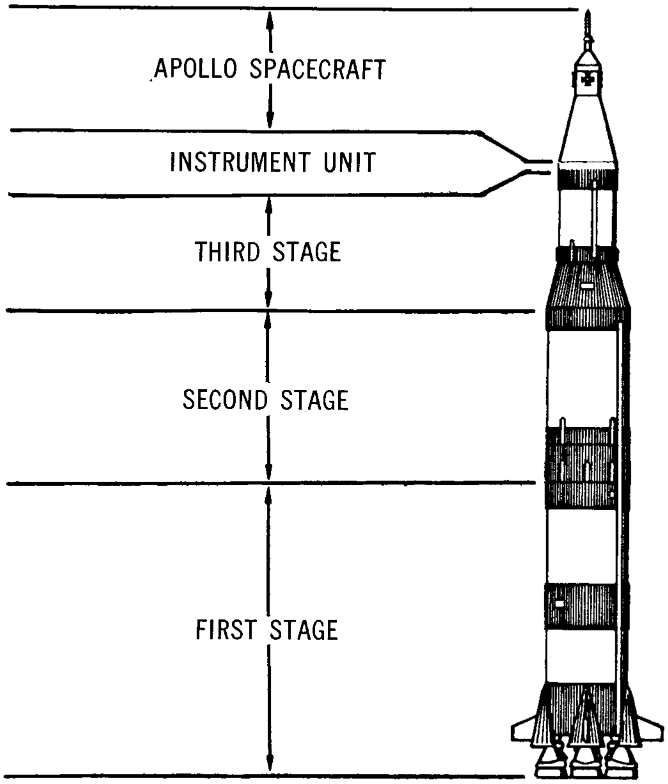
SATURN V FACT SHEET



PHYSICAL CHARACTERISTICS

OVERALL VEHICLE	DIAMETER	HEIGHT	WEIGHT
	33 ft.	363 ft.*	6,262,500 lb. (total liftoff)
FIRST STAGE	33 ft.	138 ft.	303,000 lb. (dry)
SECOND STAGE	33 ft.	81 ft. 7 in.	99,200 lb. (dry)**
THIRD STAGE	21 ft. 8 in.	58 ft. 7 in.	33,600 lb. (dry)**
INSTRUMENT UNIT	21 ft. 8 in.	3 ft.	4,500 lb.
APOLLO SPACECRAFT		80 ft.	98,000 lb.

*SINCE INDIVIDUAL STAGE DIMENSIONS OVERLAP IN SOME CASES, OVERALL VEHICLE LENGTH IS NOT THE SUM OF INDIVIDUAL STAGE LENGTHS

**INCLUDES AFT INTERSTAGE WEIGHT

PROPULSION SYSTEMS

FIRST STAGE —Five bipropellant F-1 engines developing 7,500,000 lb. thrust
RP-1 Fuel—209,000 gal. (1,400,000 lb.), LOX—334,500 gal. (3,178,000 lb.)

SECOND STAGE—Five bipropellant J-2 engines developing more than 1,000,000 lb. thrust
LH_2—275,000 gal. (159,500 lb.) ,LOX—84,750 gal. (805,500 lb.)

THIRD STAGE —One bipropellant J-2 engine developing up to 225,000 lb. thrust
LH_2—69,500 gal. (39,750 lb.), LOX—20,150 gal. (192,250 lb.)

CAPABILITY

FIRST STAGE —Operates about 2.5 minutes to reach an altitude of about 200,000 feet (38 miles) at burnout

SECOND STAGE—Operates about 6 minutes from an altitude of about 200,000 feet to an altitude of 606,000 feet (114.5 miles)

THIRD STAGE —Operates about 2.75 minutes to an altitude of about 608,000 feet (115 miles) before second firing and 5.2 minutes to translunar injection

PAYLOAD—280,000 lb. into a 115 statute-mile orbit, 100,000 lb. to the moon.

TABLE OF CONTENTS

ADDENDUM

FOREWORD

This volume has been prepared by the five Saturn V major contractors: The Boeing Company; McDonnell Douglas Astronautics Company; Space Division of North American Rockwell Corporation; Rocketdyne Division of North American Rockwell Corporation; and International Business Machines Corporation in cooperation with the National Aeronautics and Space Administration.

It is designed to serve as an aid to newsmen in present and future coverage of the Saturn V in its role in the Apollo program and as a general purpose large launch vehicle. Every effort has been made to present a comprehensive overall view of the vehicle and its capabilities, supported by detailed information on the individual stages and all major systems and subsystems.

Weights and measurements cited throughout the book are average figures. They may vary from mission to mission to meet differing requirements.

All photographs and illustrations in the book are available for general publication. The first letter in each photo number is a code identifying the organization holding that negative: B for Boeing; R for Rocketdyne Division of North American; D for Douglas; IBM for IBM; S for Space Division of North American; H for NASA, Huntsville, Ala.; and K for NASA, Kennedy Space Center, Fla.

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THE SATURN V

INTRODUCTION

The United States decided in 1961 to undertake a manned lunar landing effort as the focal point of a broad new space exploration program. There was no rocket in the country at that time even approaching the needed capability, but there was a sort of "test bed" in the making, the eight-engine Saturn I. It had never flown and was much too small to offer any real hope of sending a trio to the moon, except possibly through as many as a half dozen separate launchings from earth and the perfection of rendezvous and docking techniques, which had never been tried.

This situation brought about the announcement on Jan. 10, 1962, that the National Aeronautics and Space Administration would develop a new rocket, much larger than any previously attempted. It would be based on the F-1 rocket engine, the development of which had been underway since 1958, and the hydrogen-fueled J-2 engine, upon which work had begun in 1960.

The Saturn V, then, is the first large vehicle in the U. S. space program to be conceived and developed for a specific purpose. The lunar landing task dictated the make-up of the vehicle, but it was not developed solely for that mission. As President Kennedy pointed out when he issued his space challenge to the Congress on May 25, 1961, the overall objective is for "this nation to take a clearly leading role in space achievement which in many ways may hold the key to our future on earth." He said of the lunar landing project: "No single space project in this period will be more exciting, or more impressive to mankind, or more important for the long-range exploration of space; and none will be so difficult or expensive to accomplish..."

The Saturn V program is the biggest rocket effort undertaken in this country. Its total cost, including the production of 15 vehicles will be more than $7 billion.

NASA formally assigned the task of developing the Saturn V to the Marshall Space Flight Center on Jan. 25, 1962. Launch responsibility was committed to the Kennedy Space Center. (The Manned Spacecraft Center, the third center involved in manned space flight, is responsible for spacecraft development, crew training, and inflight control.)

DESCRIPTION

Marshall Center rocket designers conceived the Saturn V in 1961 and early 1962. They decided that a three-stage vehicle would best serve the immediate needs for a lunar landing mission and would serve well as a general purpose space exploration vehicle.

One of the more important decisions made early in the program called for the fullest possible use of components and techniques proven in the Saturn I program. As a result, the Saturn V third stage (S-IVB) was patterned after the Saturn I second stage (S-IV), and the Saturn V instrument unit is an outgrowth of the one used on Saturn I. In these areas, maximum use was made of designs and facilities already available to save time and costs.

Many other components were necessary, including entirely new first and second stages (S-IC and S-II). The F-1 and J-2 engines were already under development, although much work remained to be done. The guidance system was to be an improvement on that of the Saturn I.

Saturn V, including the Apollo spacecraft, is 363 feet tall. Fully loaded, the vehicle will weigh some 6.2 million pounds.

The 303,000-pound first stage is 33 feet in diameter and 138 feet long. It is powered by five F-1 engines generating 7.5 million pounds thrust. The booster will burn 209,000 gallons of RP-1 (refined kerosene) and 334,500 gallons of liquid oxygen (LOX) in 2.5 minutes. Propellant consumption varies with cutoff times tailored for different missions.

Saturn V's second stage is powered by five J-2 engines that generate a total thrust of a million pounds. The 33-foot diameter stage weighs 99,200 pounds empty and more than a million pounds loaded. It burns some 275,000 gallons of liquid hydrogen and 84,750 gallons of liquid oxygen during a typical flight of about 6 minutes.

Third stage of the vehicle is 21 feet and 8 inches in diameter and 58 feet and 7 inches long. An interstage adapter connects the larger diameter second stage to the smaller upper stage. Empty weight of the stage is 33,600 pounds and the fueled weight is about 265,600 pounds. A single J-2 engine developing up to 230,000 pounds of thrust powers the stage. Typical burn time is 2.75 minutes for the first burn and 5.2 minutes to a translunar injection.

The vehicle instrument unit sits atop the third stage. The unit, which weighs an average 4,500 pounds, contains the electronic gear that controls engine ignition and cutoff, steering, and all other commands necessary for the Saturn V mission. Diameter of the instrument unit is 21 feet and 8 inches, and height is 3 feet.

Directly above the instrument unit in the Apollo configuration is the Apollo spacecraft. It consists of the lunar module, the service module, the command module, and the launch escape system. Total height of the package is about 80 feet.

TYPICAL LUNAR LANDING MISSION

The jumping-off place for a trip to the moon is NASA's Launch Complex 39 at the Kennedy Space Center. After the propellants are loaded, the three astronauts will enter the spacecraft and check out their equipment.

While the astronauts tick off the last minutes of the countdown in the command module, a large crew in the launch control center handles the complicated launch operations. For the last two minutes, the countdown is fully automatic.

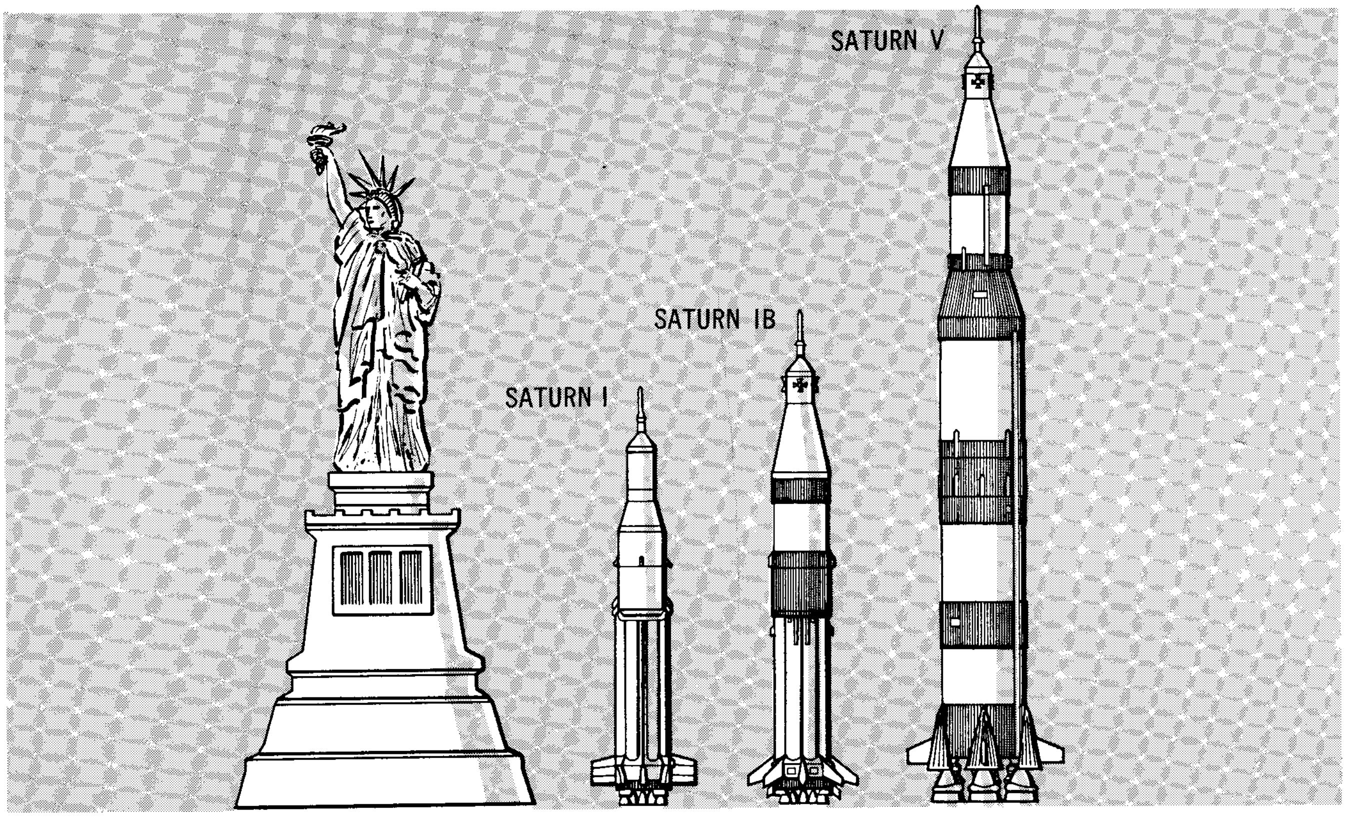
At the end of countdown, the five F-1 engines in the first stage ignite, producing 7.5 million pounds of thrust. The holddown arms release the vehicle, and three astronauts begin their ride to the moon.

Turbopumps, working together with the strength of 30 diesel locomotives, force almost 15 tons of fuel per second into the five engines. Steadily increasing acceleration pushes the astronauts back into their couches as the rocket generates 4-1/2 times the force of earth gravity.

After 2.5 minutes, the first stage has burned its 4,578,000 pounds of propellants and is discarded at about 38 miles altitude. The second stage's five J-2 engines are ignited. Speed at this moment is about 6,000 miles per hour.

The second stage's five J-2 engines burn for about 6 minutes, pushing the Apollo spacecraft to an altitude of nearly 115 miles and a velocity of about 15,300 miles per hour. After burnout the second stage drops away and retrorockets slow it for its fall into the Atlantic Ocean west of Africa.

The single J-2 engine in the third stage now ignites and burns for 2.75 minutes. This brief burn boosts the spacecraft to orbital velocity, about 17,500 miles an hour. The spacecraft, with the third stage still attached, goes into orbit about 12 minutes after liftoff. Propellants in the third stage are not depleted when the engine is shut down. This stage stays with the spacecraft in earth orbit, for its engine will be needed again.



Saturn Comparisons to Statue of Liberty

S-33040

Throughout the launch phase of the mission, telemetry systems are transmitting continously, tracking systems are locked on, and voice communications are used to keep in touch with the astronauts. All stage separations and engine thrust terminations are reported to the Mission Control Center at Houston.

The astronauts are now in a weightless condition as they circle the earth in a "parking orbit" until the timing is right for the next step to the moon.

The first attempt at a lunar landing is planned as an "open-ended" mission with detailed plans at every stage for mission termination if necessary. A comprehensive set of alternate flight plans will be laid out and fully rehearsed for use if such a decision should prove necessary. For example, a decision might be made in the earth parking orbit not to continue with the mission. At every stage of the mission, right up to touchdown on the moon, this termination decision can be made and an earth flight plan initiated.

During the one to three times the spacecraft circles the earth, the astronauts make a complete check of the third stage and the spacecraft. When the precise moment comes for injection into a translunar trajectory, the third stage J-2 engine is reignited. Burning slightly over 5 minutes, it accelerates the spacecraft from its earth orbital speed of 17,500 miles an hour to about 24,500 miles an hour in a trajectory which would carry the astronauts around the moon. Without further thrust, the spacecraft would return to earth for re-entry.

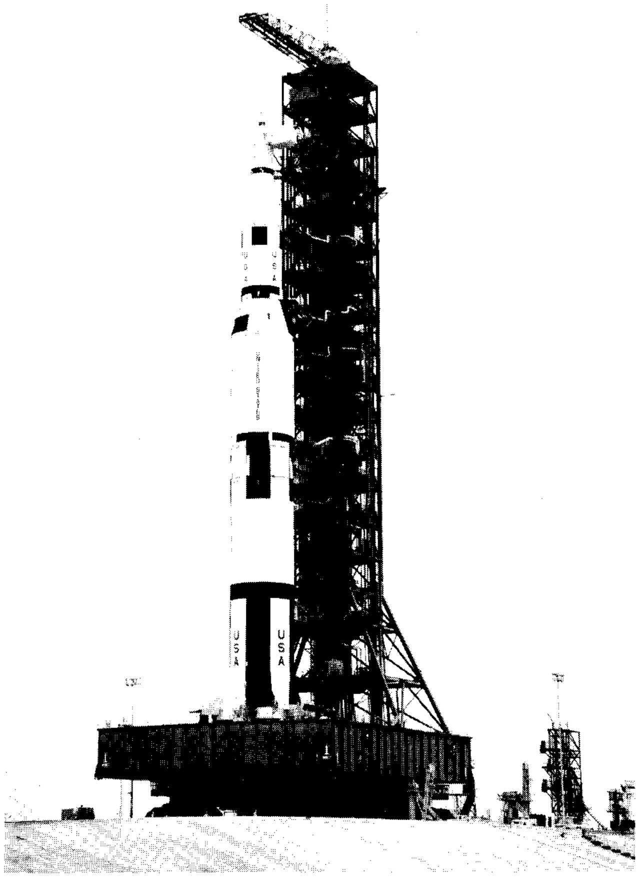
If everything is operating on schedule, the astronauts will turn their spacecraft around and dock with the lunar landing module. After the docking maneuver has been completed, the lunar module will be pulled out of the forward end of the third stage, which will be abandoned. This completes the Saturn V's work on the lunar mission.

EARLIER SATURNS

Saturn I

Studies which led to the Saturn family of rockets were started by the Wernher von Braun organization in April of 1957. The aim of the program was to create a 1.5 million-pound-thrust booster by clustering previously developed and tested engines.

On Aug. 15, 1958, the Advanced Research Projects Agency (ARPA) formally initiated what was to become the Saturn project. The agency, a separately organized research and development arm of the Department of Defense, authorized the Army Ballistic Missile Agency to conduct a research and development program at Redstone Arsenal for a 1.5 million-pound-thrust vehicle booster. A number of available rocket engines were to be clustered and tested by a full-scale static firing by the end of 1959.



K-107P-66P-258

Test Vehicle—The first assembled Apollo/Saturn V vehicle approaches the launch pad at Kennedy Space Center. It was used to verify launch facilities, train launch crews, and develop test and checkout procedures at KSC. It was rolled out on May 25, 1966.

The program objectives were expanded by ARPA in October of 1958 to include a multi-stage carrier vehicle capable of performing advanced space missions. Concurrent with the development of a multistage vehicle, static test facilities at Redstone Arsenal and launch complex facilities at Cape Canaveral—now Cape Kennedy—were being constructed.

The proposed large vehicle project was officially renamed Saturn on Feb. 3, 1959, by ARPA memorandum. The space agency assumed technical direction of the Saturn project in late 1959. The project was transferred officially on Mar. 16, 1960, and the Army development group at Huntsville was transferred to NASA and became the nucleus of the new Marshall Space Flight Center. The first static firing of a Saturn I booster was conducted April 29, 1960.

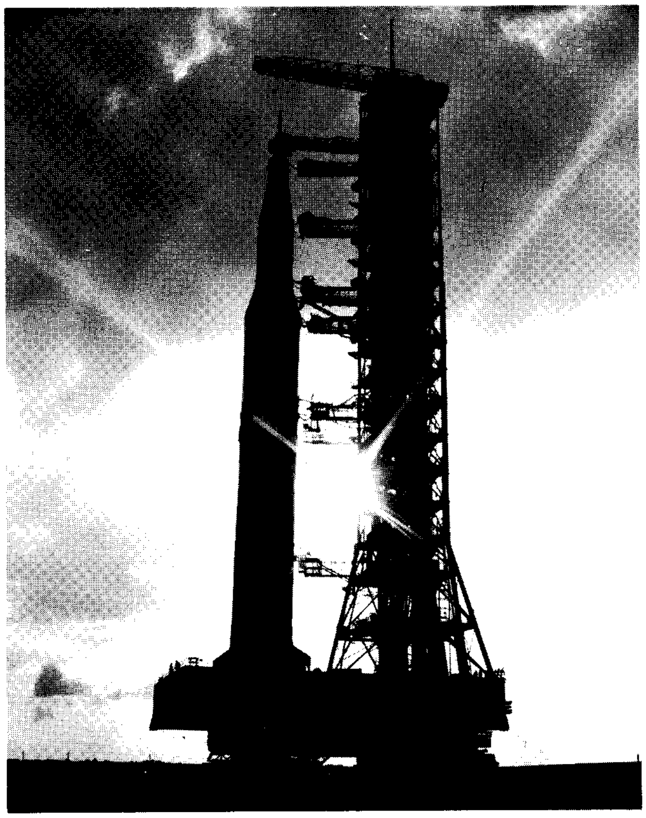
The NASA Saturn Vehicle Evaluation Committee (Silverstein Committee) on Dec. 15, 1959, recommended a long-range development program for a Saturn vehicle with upper stage engines burning liquid hydrogen and liquid oxygen. The initial vehicle, identified as Saturn C-1 and now as Saturn I, was to be a stepping stone to a larger vehicle. A building-block concept was proposed that would yield a variety of Saturn configurations, each using previously proven developments as far as possible.

Early in 1960 the Saturn program was given the highest national priority, and a 10-vehicle research and development program was approved.

The two-stage Saturn I vehicle with the Apollo spacecraft was about 188 feet tall and weighed some 1,125,000 pounds at liftoff.

While plans for the lunar mission were progressing, the Saturn I project made history. On Oct. 27, 1961, the first Saturn I booster was flight tested successfully from Cape Kennedy. The first flight booster with dummy upper stages was called SA-1. This vehicle was followed by successful flights of SA-2 on April 25, 1962, SA-3 on Nov. 16, 1962, and SA-4 on Mar. 28, 1963.

The SA-5 vehicle, combining the first stage (S-1) with the second stage (S-IV), was successfully launched on Jan. 29, 1964, with both stages functioning perfectly to place a 37,700-pound payload into earth orbit. SA-6, launched on May 28, 1964, and SA-7, launched on Sept. 18, 1964, each placed "unmanned" boilerplate configurations of Apollo spacecraft into earth orbit.



K-P39928

Saturn V Launch Vehicle at Sunset

SA-9, launched on Feb. 19, 1965, was the first Saturn I vehicle to launch a Pegasus meteoroid technology satellite into earth orbit.

The SA-8 and SA-10 Saturn I vehicles were successfully launched on May 25, 1965, and July 30, 1965, respectively, also placing a Pegasus satellite into earth orbit to complete the test and launch program with an unprecedented 100 per cent record of success.

Saturn IB

The space agency, using the building-block approach, conceived the Saturn IB as the quickest, most reliable, and most economical means of providing a vehicle with greater payload than the Saturn I. This vehicle was planned for orbital missions with the Apollo spacecraft.

The Saturn IB is based on a blending of existing elements of Saturn I and Saturn V. A redesigned Saturn I booster (designated the S-IB stage), and an S-IVB upper stage and instrument unit from the Saturn V are used on this launch vehicle.

Maximum use of designs and facilities available from the earlier approved Saturn programs saved both time and costs.

The Saturn I first stage was redesigned in several areas by NASA and the Chrysler Corporation, the stage contractor, for the expanded role as the Saturn IB booster. Basically, it retained the same shape and size, but required some modification for mating with the upper stage, which has a greater diameter and weight than the Saturn I upper stage.

Stage weight was cut by more than 20,000 pounds to increase payload capacity. The Rocketdyne H-1 engine was uprated to 200,000 pounds of thrust, compared with 188,000 pounds of thrust for each engine in the final Saturn I configuration. The engines will be improved again to 205,000 pounds beginning with the SA-206.

For the Saturn IB a guidance computer used in the early Saturn I was replaced by another IBM computer of completely new design which incorporates the added flexibility and extreme reliability necessary to carry out the intended Saturn IB missions.

The Saturn IB, topped by the Apollo spacecraft, stands approximately 224 feet tall and is about 21.7 feet in diameter. Total empty weight is about 78 tons, and liftoff weight fully fueled is approximately 650 tons.

The first Saturn IB vehicle (AS-201) was launched Feb. 26, 1966. The next four were launched July 5 and Aug. 25, 1966, and Jan. 22 and Oct. 11, 1968.

HOW SATURN V DESIGN WAS REACHED

While a major effort of this country's space commitment was to explore the moon, the broader target was to build a capability—people, launch vehicles, propulsion, spacecraft, production, testing, and launching sites—to explore a vast new frontier and develop a long-range spacefaring capability that would establish continuing national preeminence.

The questions facing national space planners in 1961 and 1962 were complex. Although the use of a Saturn I for a manned lunar landing was theoretically possible, it would have been extremely difficult. About six Saturn I launches would have been required, their payloads being assembled in earth orbit to form a moon ship. No space rendezvous and docking had taken place at that time.

During the first half of 1962, two paramount decisions were announced: to develop a new general purpose launch vehicle in the middle range of several under consideration, and to conduct the manned lunar landing by use of a lunar orbit rendezvous (LOR) technique.

The Saturn V, as the chosen vehicle was named, was given the go-ahead in January, 1962.

It was to be composed of three propulsive stages and a small instrument unit to contain guidance and control. It could perform earth orbital missions through the use of the first two stages, while all three would be required for lunar and planetary expeditions. The ground stage was to be powered by five F-1 engines, each developing 1.5 million pounds of thrust, and the stage would have five times the power of the Saturn I booster then under development. The upper stages would use the J-2 hydrogen/oxygen engine, five in the second stage and one in the third. Each would develop up to 225,000 pounds of thrust. Such a rocket would be capable of placing 120 tons into earth orbit or dispatching 45 tons to the moon. (The numbers have been uprated now to about 140 and 50.)

During its assembly, checkout, and launch, the Saturn V would use a new mobile launch concept. It would be assembled in a huge Vehicle Assembly Building, and then transported in an upright position to a launch pad several miles away.

Propulsion development decisions preceded those for the vehicles.

The need for a building-block rocket engine in the million-pound-thrust class was apparent even as ARPA was ordering work to begin on the first stage cluster of engines for the Saturn I. In January, 1959, NASA contracted with North American Aviation's Rocketdyne Division for development of the F-1.

Late in 1959, the Silverstein Committee recommended the development of a new high-thrust hydrogen engine to meet upper stage requirements. In June, 1960, Rocketdyne was selected to develop the J-2 engine after NASA evaluation of competitive proposals.

Three proposed Apollo modes which were considered in detail were: the direct flight mode, using a very large launch vehicle called "Nova"; the earth orbital rendezvous (EOR) mode, requiring separate Saturn launches of a tanker and a manned spacecraft; and the lunar orbital rendezvous mode, requiring a single launch of the manned spacecraft and the lunar module.

Selected was the LOR mode, in which the injected spacecraft weight would be reduced by eliminating the requirement for the propulsion needed to soft-land the entire spacecraft on the lunar surface.

A small lunar excursion module, or LEM, now referred to as the lunar module, would be detached after deboost into lunar orbit. The lunar module would carry two of the three-man Apollo crew to a soft landing on the moon and would subsequently be launched from the moon to rendezvous with the third crew member in the "mother ship." The entire crew would then return to earth aboard the command module.

NASA concluded that LOR offered the greatest assurance of successful accomplishment of the Apollo objectives at the earliest practical date.

Members of NASA's Manned Space Flight Management Council recommended LOR unanimously in 1962 because it:

1. Provided a higher probability of mission success with essentially equal mission safety;
2. Promised mission success some months earlier than did other modes;
3. Would cost 10 to 15 per cent less than the other modes; and
4. Required the least amount of technical development beyond existing commitments while advancing significantly the national technology.

As a part of the Saturn V decision, it was determined that elements of the existing Saturn I vehicle and the planned Saturn V would be combined to form a new mid-range vehicle, the Saturn IB. The Saturn IB would have a payload capability 50 per cent greater than the Saturn I and would make possible the testing of the Apollo spacecraft in earth orbit about one year earlier than would be possible with the Saturn V.

By the end of 1962, all elements of the new program were under way, with the Marshall Space Flight Center directing the work for NASA. The Boeing Company; Space Division of North American Aviation, Inc.; and Douglas Aircraft Company were acting as prime contractors for the Saturn V first, second, and third stages, respectively. Engines were being developed by the Rocketdyne Division of North American. MSFC designed the instrument unit and awarded a production contract to International Business Machines Corp. (Chrysler Corp. had been selected to produce the first stage of the Saturn IB.)

A large network of production, assembly, testing, and launch facilities was also being prepared by the end of 1962. Aside from the provision of various facilities at contractor plants and the augmentation of the Marshall Space Flight Center resources, three new government operations were established: the launch complex in Florida operated by the NASA-Kennedy Space Center and two new elements of MSFC—Michoud Assembly Facility in New Orleans, La., for the production of boosters, and Mississippi Test Facility, Bay St. Louis, Miss., for captive firing of stages.

Four years after its establishment, the Saturn V program was progressing on schedule, pointing toward the launch of the first vehicle in 1967 and fulfillment of the manned lunar landing before the end of the decade.

PROGRAM HIGHLIGHTS

Following are highlights of the Saturn V development program:

1961

Aug. 24	NASA announced the selection of the 88,000-acre site at Merritt Island, Fla., adjacent to Kennedy Space Center, then Cape Canaveral, for the assembly, checkout, and launch of the Saturn V.
Sept. 7	NASA selected the government-owned Michoud plant, New Orleans, as production site for Saturn boosters. It became a part of the Marshall Space Flight Center.
Sept. 11	NASA selected North American Aviation, Inc., to develop and build the second stage for an advanced Saturn launch vehicle (as yet undefined) for manned and unmanned missions. One month later the Marshall Center directed NAA to design the second stage using five J-2 engines. A preliminary contract was signed in February, 1962.
Oct. 6	NASA selected the Picayune-Bay St. Louis, Miss., area for its Mississippi Test Facility—an arm of the Marshall Center—for use in static testing of rocket stages and engines.
Dec. 15	The Boeing Company was selected as prime contractor for the first stage of the advanced Saturn vehicle—not yet fully defined. A preliminary contract was signed in February, 1962, with the work to be conducted at the Michoud Assembly Facility.
Dec. 21	NASA selected the Douglas Aircraft Company to negotiate a contract to develop the third stage (S-IVB) of the advanced Saturn, based on the Saturn I's S-IV stage. A supplemental contract for production of 11 third stages was signed in August, 1962.

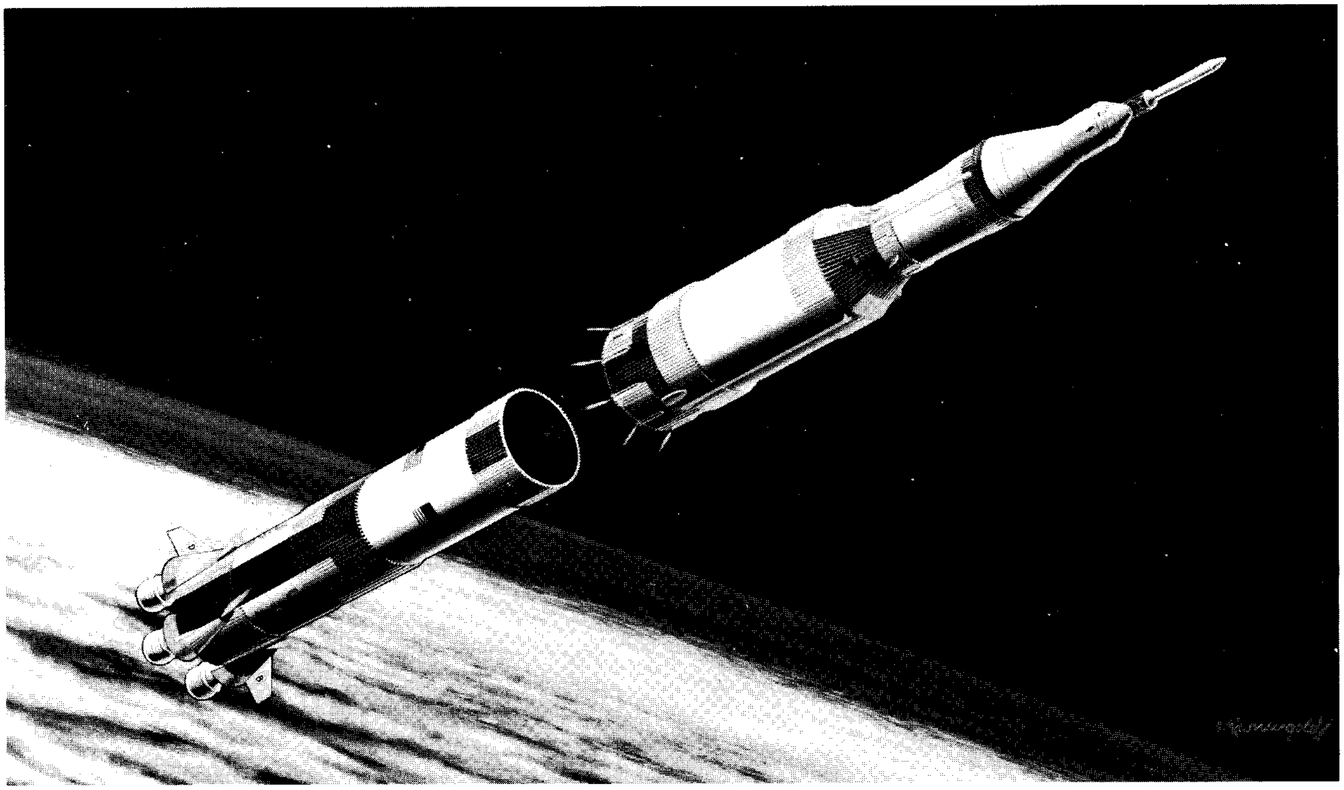
1962

Jan. 10	Announcement was made that the advanced Saturn vehicle would have a first stage powered by five F-1 engines, a second stage powered by five J-2 engines, and for lunar missions a third stage with one J-2 engine.
Jan. 25	NASA formally assigned development of the three-stage Saturn C-5 (Saturn V became the name in February, 1963) to MSFC.
April 11	NASA Headquarters gave the Apollo/Saturn I/Saturn V highest national priority.
May 26	Rocketdyne Division of NAA conducted the first full-thrust, long-duration F-1 engine test.
July 11	It was announced that the Saturn IB would be developed and that the lunar orbit rendezvous method of accomplishing a lunar landing had been selected.
December	The U. S. Army Corps of Engineers awarded a contract for the design of the Vehicle Assembly Building (VAB) at the Florida launch complex.

1963

Feb. 27	The first contract for the Mississippi Test Facility (MTF) Saturn V test facilities was awarded.
May	The J-2 engine was successfully fired for the first time in a simulated space altitude of 60,000 feet.
Oct. 31	The Marshall Center received the first production model of the F-1 engine.

Nov. 12	NASA contracted for the first Saturn V launch pad at the Kennedy Space Center.
1964	
March	IBM was awarded an instrument unit contract for the digital computer and data adapter by the Marshall Center. IBM became the prime IU contractor in May.
Oct. 9	The Edwards AFB test facility was accepted as the F-1 test complex, amounting to a cost of $34 million.
Dec. 1	The first mainstage shakedown firing of the third stage battleship was accomplished, lasting 10 seconds.
Dec. 23	First full-duration firing of the third stage battleship occurred.
1965	
April 16	All five engines of the S-IC-T, first stage test vehicle, were fired at the Marshall Center for 6.5 seconds.
April 24	The first cluster ignition test of the second stage battleship was successfully completed.
Aug. 5	The first full-duration firing of the first stage was conducted successfully at the Marshall Center.
Aug. 8	Third stage flight readiness test of 452 seconds, fully automated, was accomplished at Sacramento.
Aug. 13	The IU was qualified structurally and man-rated for Saturn V use by withstanding a 140 per cent load limit.
Aug. 17	The third stage battleship was tested in Saturn V configuration for full duration (start-stop-restart).
Dec. 16	The S-IC-T static firings were completed at the Marshall Center with a total of 15 firings—three of full duration.
1966	
Feb. 17 & 25	The S-IC-1 underwent static firing at the Marshall Center and required no more static firings.
Mar. 30	The S-IU-500F was mated to the three stages of the Saturn V facilities vehicle at the Kennedy Space Center's VAB.
May 20	First full-duration firing of the second stage flight stage was conducted at MTF.
May 25	The Apollo/Saturn V facilities vehicle, AS-500-F, was transported to Pad A at Launch Complex 39, KSC, on the crawler.
May 26	Full-duration acceptance firing of the S-IVB-501, the first flight version of the third stage for Saturn V, was accomplished.
September	The F-1 and J-2 engines were qualified for manned flights.
Dec. 1	Initial static firing of the first flight version of the second stage occurred at MTF.
Nov. 15	The first flight version of the first stage was static fired at MSFC.



First Stage Separation During an Apollo/Saturn V Shot

B-14066-1

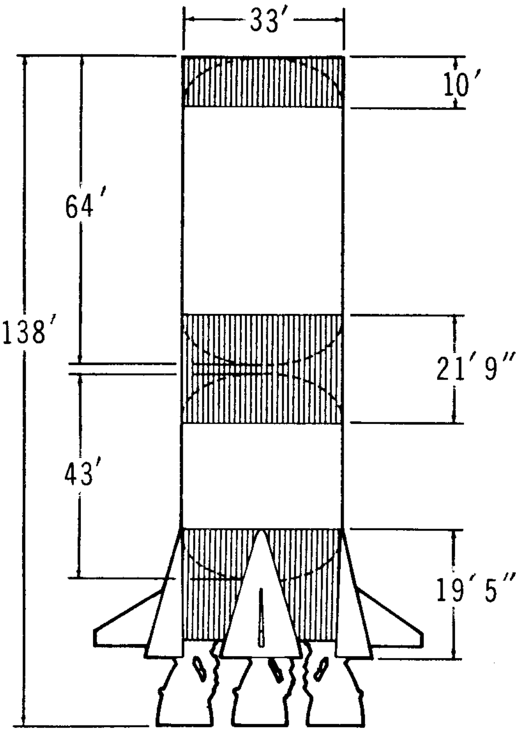
1967

March 3	S-IC static test stand at MTF declared operational following firing of S-IC-T.
March 16	Start and restart tests of J-2 engine in vacuum chamber at Arnold Engineering and Development Center, Tullahoma, Tenn. completed successfully.
April 12	Ten-week phase of dynamic testing of complete Saturn V launch vehicle and Apollo spacecraft completed at MSFC.
Aug. 26	First Saturn V flight vehicle, AS-501, rolled out of VAB at KSC and transported to LC 39.
Nov. 9	Launch of first Saturn V, the AS-501, from LC 39 at KSC.

1968

April 4	Launch of second Saturn V, the AS-502, from LC 39 at KSC.
May 1	NASA announced decision to man the third Saturn V, the AS-503, after studying results of AS-501 and AS-502 flights.

FIRST STAGE FACT SHEET



NOTE:
Figures given for weights and contents are average. These may vary to meet requirements for the differing missions. In those cases where the numbers in the fact sheet differ with the text, the fact sheet contains more current information.

WEIGHT: 303,000 lb. (dry)
4,881,000 lb. (loaded)
DIAMETER: 33 ft.
HEIGHT: 138 ft.
BURN TIME: About 2.5 min.
VELOCITY: 6,000 miles per hour at burnout (approx.)
ALTITUDE AT BURNOUT: About 38 miles

MAJOR STRUCTURAL COMPONENTS

THRUST STRUCTURE
FUEL TANK
INTERTANK
LOX TANK
FORWARD SKIRT

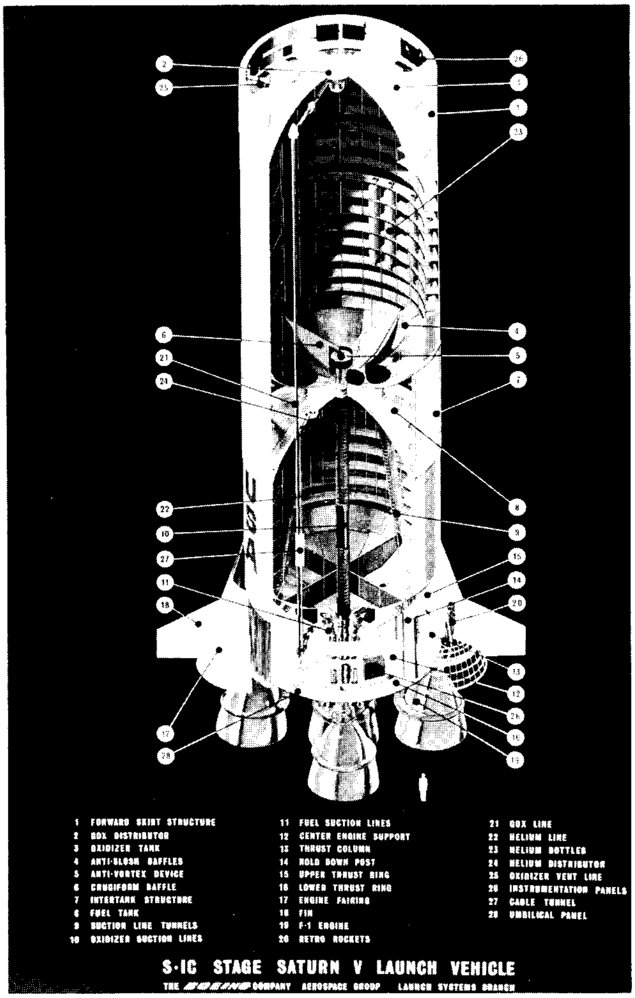
MAJOR SYSTEMS

PROPULSION: Five bipropellant F-1 engines
Total thrust: 7.5 million lb.
Propellant: RP-1—209,000 gal. or 1,400,000 lb.
LOX—334,500 gal. or 3,178,000 lb.
Pressure: Control—1.27 cubic feet of gaseous nitrogen at 3,250 psig
Fuel pressurization—124 cubic feet or 636 lb. of gaseous helium at 3,100 psig
LOX pressurization—gaseous oxygen converted from 6,340 pounds of LOX by the engines
HYDRAULIC: Power primarily for engine start and for gimbaling four outboard engines
ELECTRICAL: Two 28 VDC batteries, basic power for all electrical functions
INSTRUMENTATION: Handles approx. 900 measurements
TRACKING: ODOP Transponder

FIRST STAGE

FIRST STAGE DESCRIPTION

The Saturn V first stage (S-IC) is a vertical grouping of five cylindrical major components and a cluster of five F-1 rocket engines. Upward from the engines are the thrust structure, fuel tank, intertank structure, LOX tank, and forward skirt. The total stage measures 138 feet in height and 33 feet in diameter without its fins. It weighs 4,881,000 pounds at liftoff and delivers 7.5 million pounds of thrust.

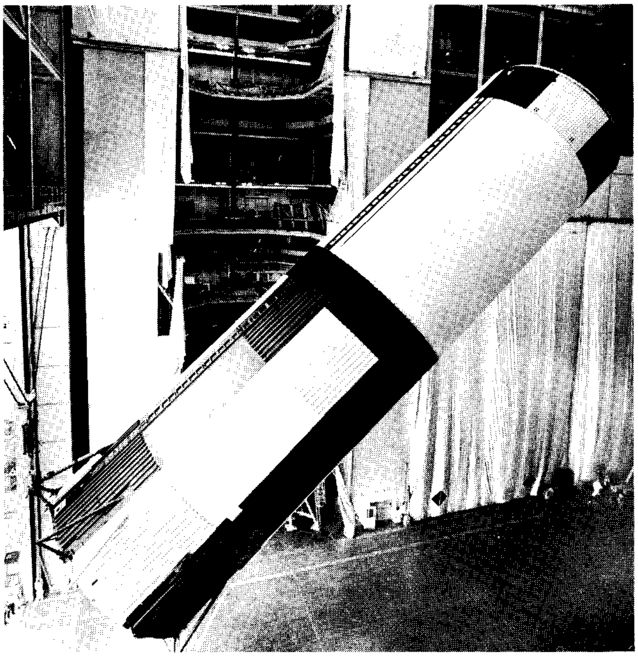


B-9454-1

First Stage Cutaway

FIRST STAGE FABRICATION AND ASSEMBLY

Design, assembly, and test of the first stage booster are the prime tasks being performed by The Boeing Company at the Marshall Space Flight Center, Huntsville, Ala., the Michoud Assembly Facility, New Orleans, La., and the Mississippi Test Facility in southwestern Mississippi. Launch operations support is provided by the Boeing Atlantic Test Center, Kennedy Space Center, Fla. Contractor suppliers lend support for much of the first stage fabrication. Several ground test stages were completed before manufacture of a series of flight stages was begun. Huntsville and Michoud installations shared responsibility for assembly of four ground test stages and the first two flight stages. All other flight stages are being assembled at Michoud.



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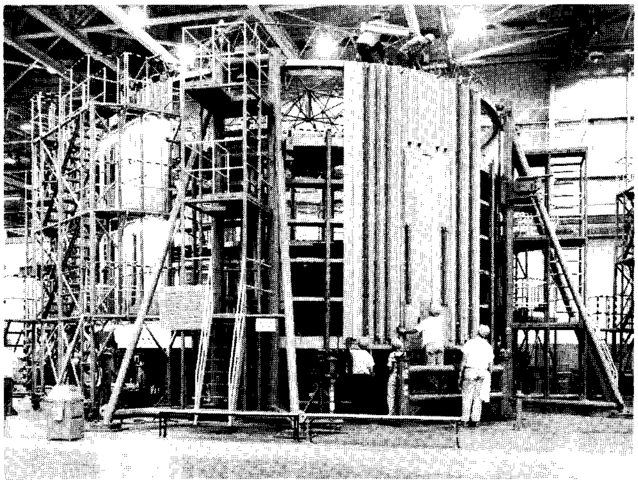
Assembled First Stage

Thrust Structure

The thrust structure is the heaviest of first stage components, weighing 24 tons. It is 33 feet in diam-



B-10648-5

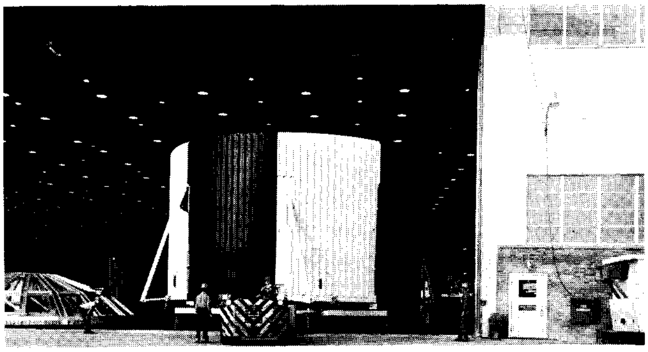
Base Assembly—Workmen cover the thrust structure shell with aluminum skin.

eter and about 20 feet tall with these major components: the lower thrust ring assembly, the center engine support assembly, four holddown posts, engine thrust posts, an upper thrust ring assembly, intermediate rings, and skin panel assemblies.

The upper ring provides stability for the corrugated skins around the structure. Four F-1 engines are mounted circumferentially upon the thrust posts and the fifth upon the center engine support assembly. The center engine remains rigid while the others gimbal or swivel, allowing the stage to be guided.

A base heat shield protects internal parts from engine heat, and four holddown posts restrain the vehicle while the engines build up power for liftoff.

The thrust structure supports the entire vehicle weight and distributes the forces of the engines.



B-6018-7

Thrust Structure—The 24-ton base of the booster is being taken to the Vertical Assembly Building for mating with other first stage components.

Fuel Tank

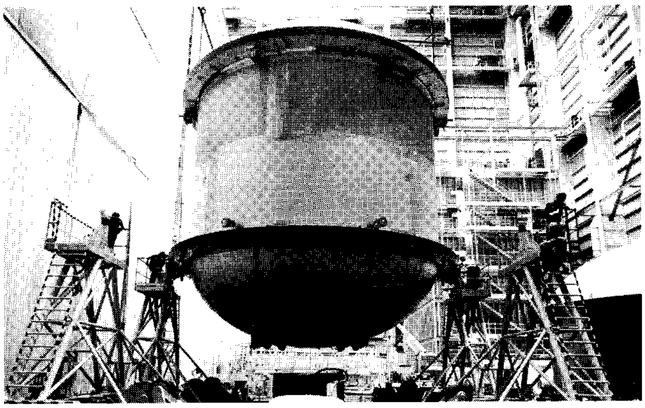
The fuel tank holds 203,000 gallons of kerosene and encloses a system of five LOX tunnels.

The tank, weighing more than 12 tons dry, is capable of releasing 1,350 gallons of kerosene per second to the engines through 10 fuel-suction lines. The LOX tunnels carry liquid oxygen from the LOX tank, through the fuel tank, and to the engines.

Bound by eight aluminum skin panels, the fusion-welded fuel tank assembly is 33 feet in diameter and 44 feet tall. Ends are enclosed by ellipsoidal bulkheads.

The bulkheads consist of eight pie-shaped gores mated with a polar cap to form a dome shape.

Connecting links between the skin rings and bulkheads are circular bands known as the Y-rings. The Y-rings are used on both propellant tanks and link them to other segments of the booster at final assembly.



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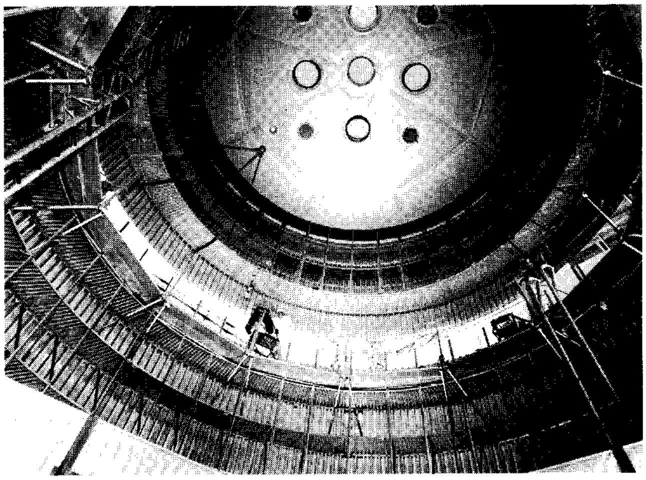
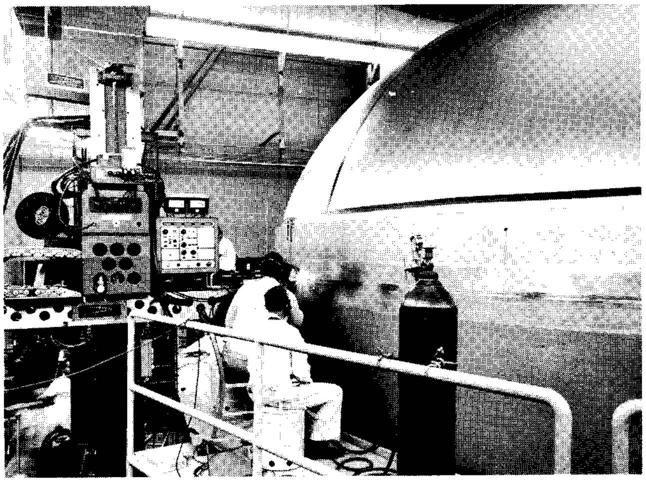
Fuel Tank—Kerosene is fed to the engines at 1,300 gallons per second from this 203,000 gallon tank. Here the finished tank is being lowered onto its transporter.



B-5622-4

Inside View—The fuel tank contains horizontal baffles, which are designed to prevent sloshing of fuel.

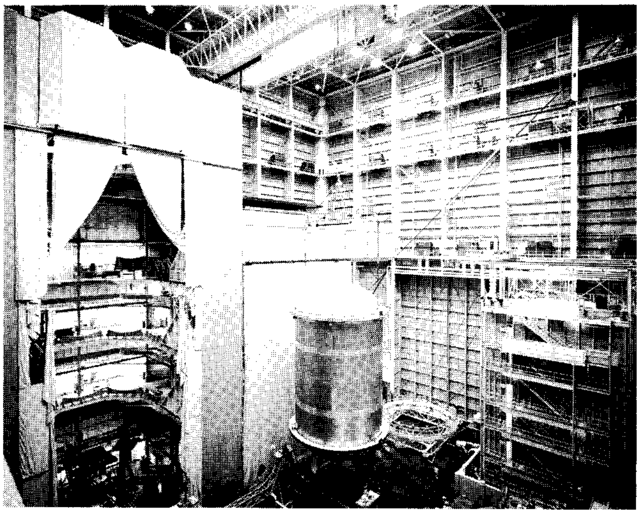


B-4780-6

Fuel Tank Assembly—Workmen weld the base of the 27-inch-high Y-ring to the cylindrical segment of the fuel tank. This ring joins the tank sides to the dome and to the intertank structure.

LOX Tank

The 331,000-gallon liquid oxygen tank is the largest component of the first stage booster, standing more than 64 feet in height. Its content is 297 degrees below zero Fahrenheit and provides the oxidizer to support combustion of the kerosene. Mixing of the two propellants is in a proportion to ensure complete combustion. Each second during flight, the engines consume more than 2,000 gallons of liquid oxygen.



B-8219-3

LOX Tank—The completed 331,000-gallon LOX tank is being carried to the hydrostatic testing facility where it will be tested for leaks.

The LOX tank's construction is similar to that of the fuel tank with the LOX tunnels beginning at the tank base, running through the intertank and fuel tank and to the engines. Dry weight of the LOX tank exceeds 19 tons.

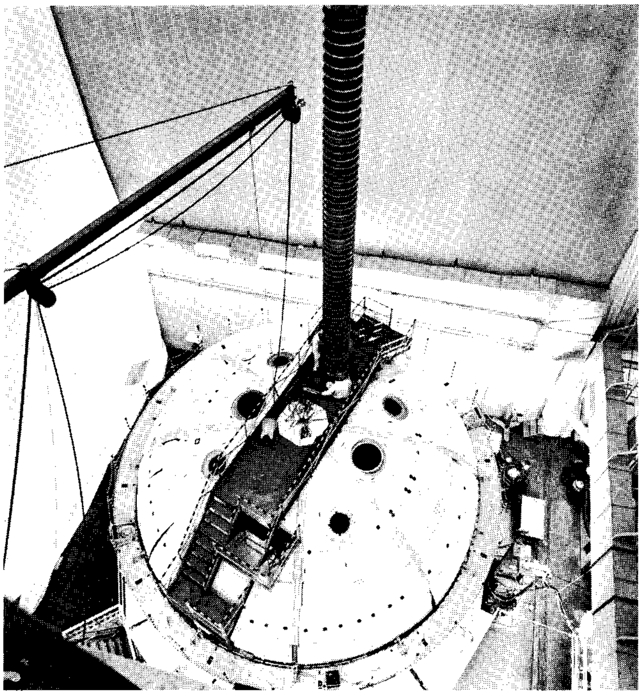
Intertank

The intertank is not a tank in itself but serves as a 6-1/2-ton link between fuel and LOX tanks. Its composition is 18 corrugated skin panels supported by five frame ring assemblies.

The lower bulkhead of the LOX tank dips into the intertank while the upper bulkhead of the fuel tank extends upward into the intertank. Around the edges of the intertank are attached 216 fittings, which fasten the tank together with the Y-rings of the fuel and LOX tanks. The intertank structure also contains a personnel access door.

Umbilical Openings

An umbilical opening in the intertank provides for electrical and instrumentation requirements, emergency LOX drain, line pressurization, electrical conduit, and provisions for venting internal pressure. The thrust structure contains three of four other umbilical openings on the booster. The fourth is located in the forward skirt. The thrust structure umbilicals carry the fuel line, liquid oxygen drain, ground supply fluid lines, and all control functions essential in case of a vehicle abort.



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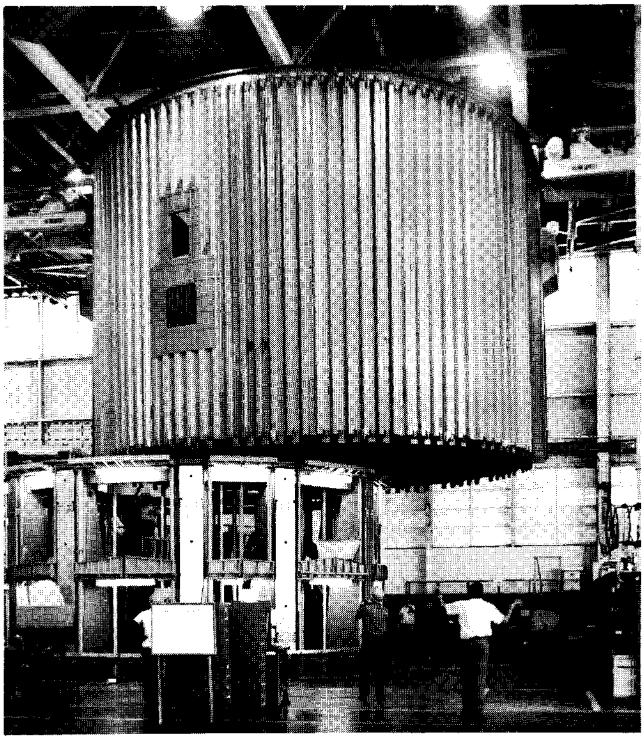
LOX Tunnel—Five 42-foot tunnels bring liquid oxygen from the LOX tank through the fuel tank and to the engines. Here a tunnel is being fitted into the fuel tank.



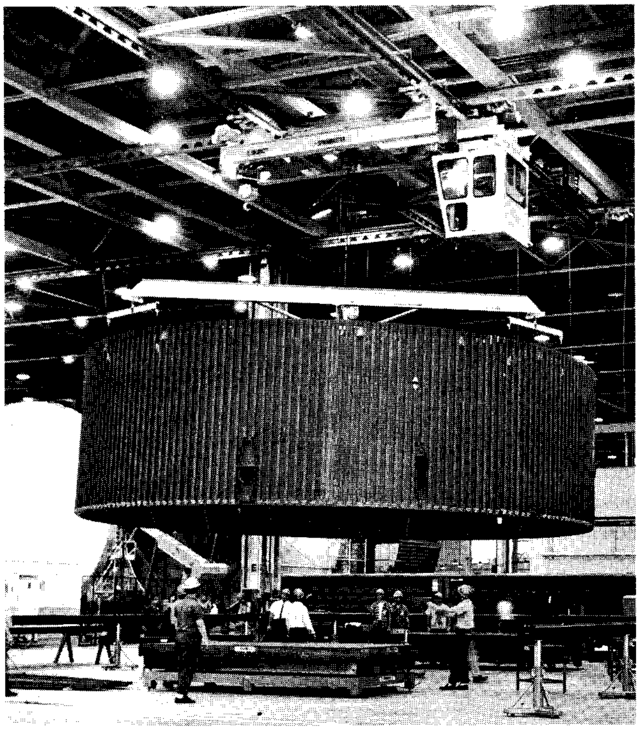
B-3291-6

A Completed Intertank

Forward Skirt

The forward skirt tops the first stage and provides a connecting link for the first and second stages of the Saturn V.

Weighing 2-1/2 tons, the structure consists of 12 skin panels attached to three circumferential support rings. It contains a small personnel access door; an umbilical opening for telemetry cables, an environmental air duct, and minor pneumatic lines; and an umbilical disconnect door.



B-2835-34

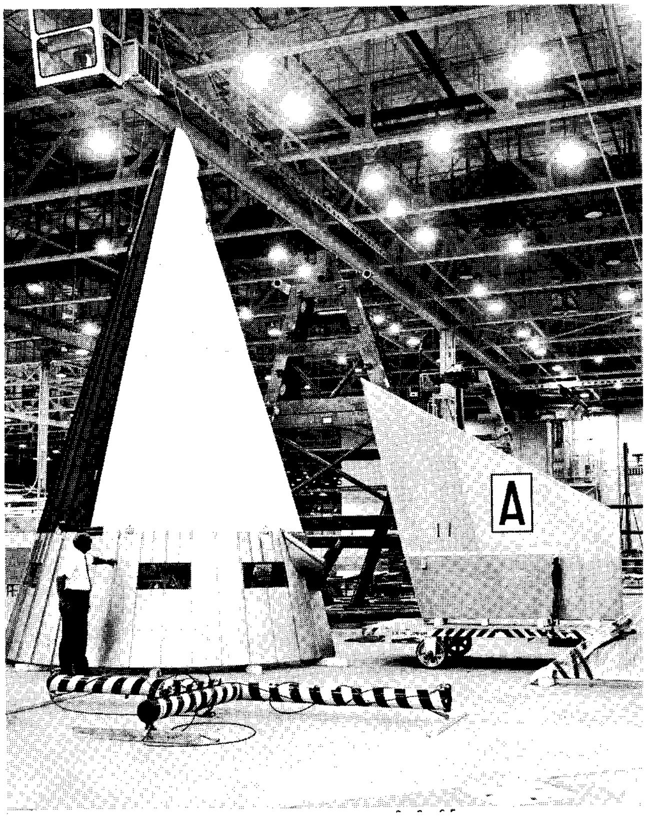
Forward Skirt—The structural link between the LOX tank and the engine shroud of the second stage is shown being lowered for dimensional inspection.

Fins and Fairings

Four fairings attach to the thrust structure and partially surround the outboard engines at the foot of the booster. They house the eight retrorockets and the actuator support structures. Fairings are shaped like cone halves and are constructed of aluminum. Their purpose is to smooth the air flow over the engines.

The fins are airfoil attachments to the fairings. Fins are rigid and add to the vehicle's flight stability. A titanium skin covers the fin for greatest protection against temperatures as high as 2,000 degrees Fahrenheit.

Each of the eight retrorockets generates about 86,600 pounds of thrust for two-thirds of a second and, upon firing, blows off the tips of the fairings. (Retrorocket thrust varies with propellant temperature.)



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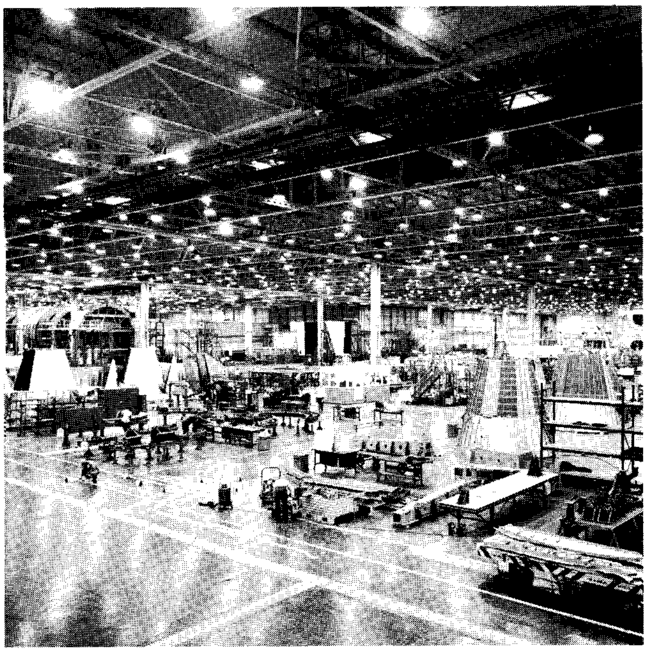
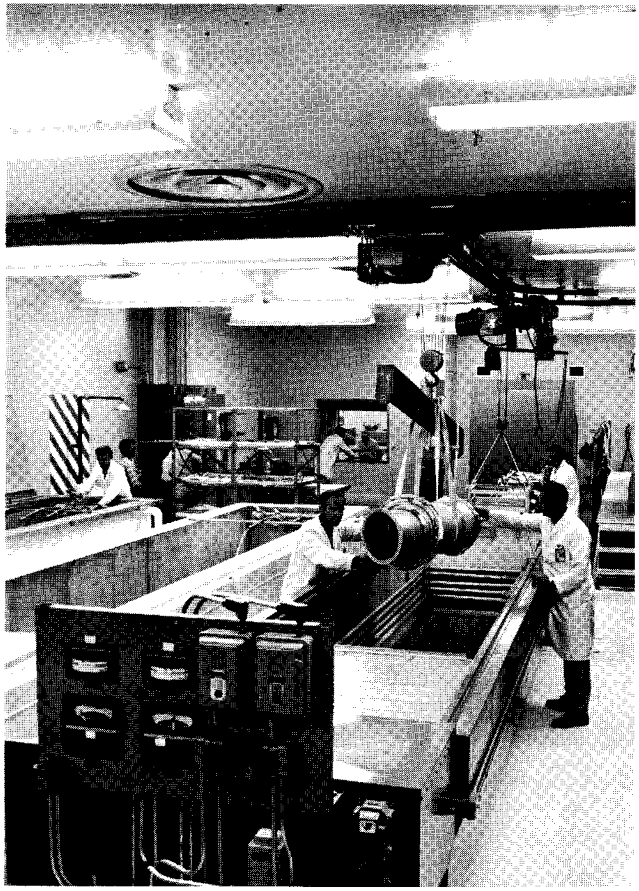
Fin and Fairing Assembly—Fairings are fitted over each of the outboard engines to smooth the air flow. Fins are attached to the fairings.



B-9580-8

Michoud Manufacturing Area—In the foreground of this Michoud plant view, fairings are being assembled.



B-9940-7

Tube and Valve Cleaning Vat—Each stage component is treated in a cleaning solution before final assembly.

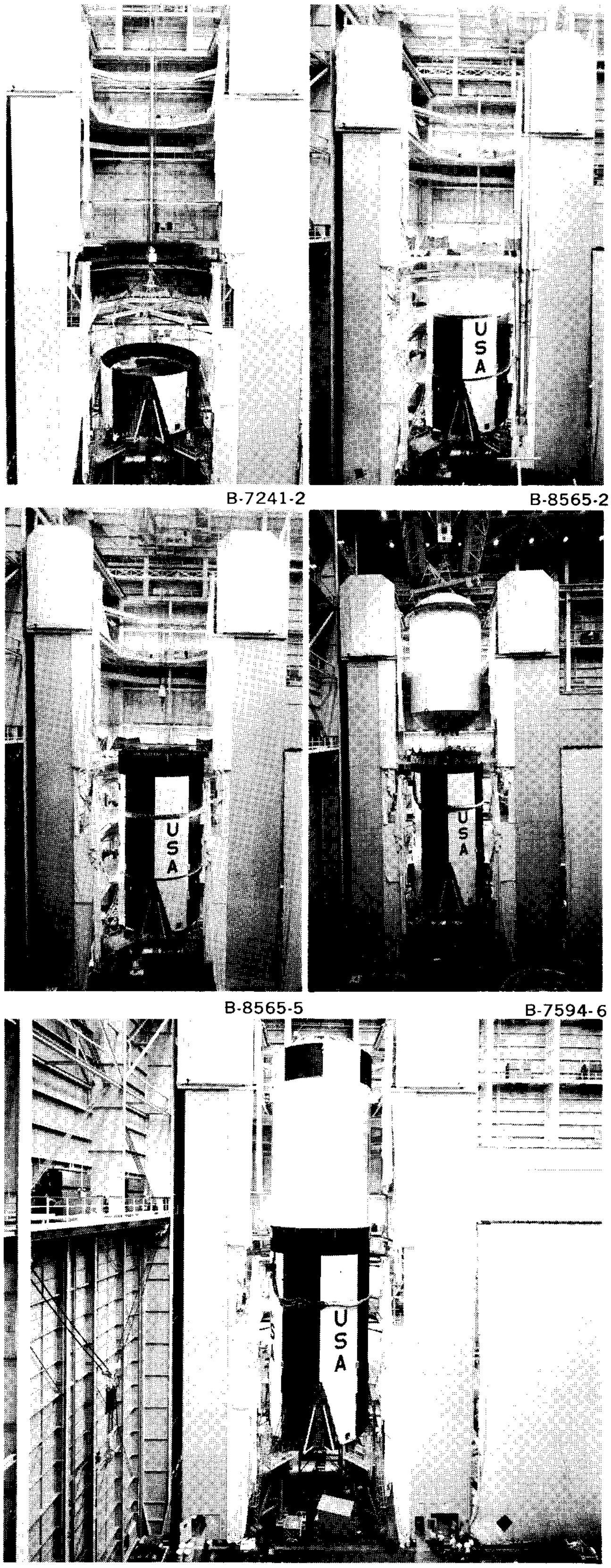
Vertical Assembly

When all major components of the first stage are assembled in NASA's Michoud Assembly Facility, they are routed to the Vertical Assembly Building to be assembled.

Manipulated by an overhead crane, the components are placed in final assembly position in the single-story building rising the equivalent of 18 stories.

First the thrust structure is placed on four heavy pylons 20 feet above floor level. Meanwhile, two of the segments—the fuel and LOX tanks which are brought to the Vertical Assembly Building in segments—are being completed on two tank assembly bays. Then, in building-block fashion, the thrust structure is joined by the fuel tank, intertank, LOX tank, and forward skirt. When the forward skirt is secured, the first stage stands 138 feet high.

Vertical assembly completed, the 180-ton-capacity overhead crane lifts the booster by a forward handling ring attached to the forward skirt and returns it to horizontal position on its 435,000-pound transporter.



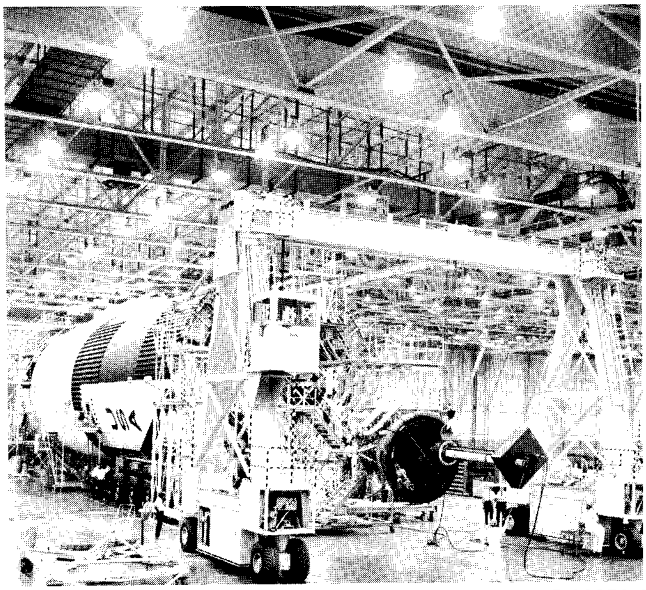
B-7241-2 B-8565-2

B-8565-5 B-7594-6

B-7594-1

Vertical Assembly—Booster sections are mated in the Vertical Assembly Building. At top left the thrust structure is shown. Fuel tank, intertank assembly, LOX tank, and forward skirt are added in successive pictures.

As assembly jobs approach completion, installation of internal systems and engines is made in preparation for systems test and checkout.



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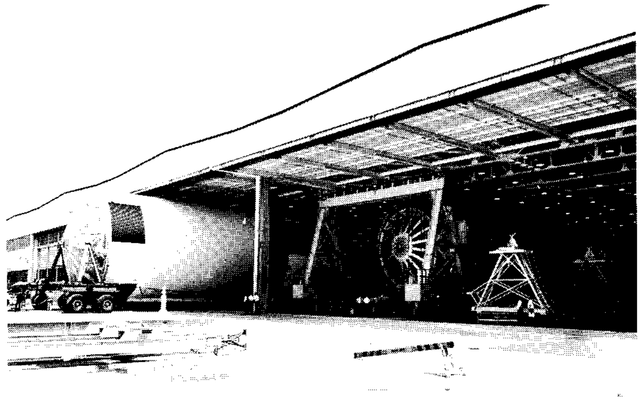
Engines—One of the first stage's F-1 engines is mounted. Together the five will consume 4,492,000 pounds of propellants in 2.5 minutes.

POST MANUFACTURING CHECKOUT

Before a booster leaves Michoud for test firing, its electrical and mechanical systems are tested extensively by Boeing technicians and engineers. The Stage Test Building with four giant test cells provides the facility. Inside the building are four control rooms, four computer rooms, and two telemetry rooms. These rooms house equipment that demonstrates the acceptability of the integrated systems of the booster. This includes telemeter calibration, continuity checks, and discrete-function monitoring. RF (radio frequency) also is evaluated.



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Moving—A completed first stage is readied for post-manufacturing checkout.

Mechanical, hydraulic, and pneumatic systems tests are conducted to leak-check and functionally check the propellant systems and the engine complex. Checks then are performed to demonstrate the proper operation of the electrical and instrumentation systems. All systems are operated and checked individually and then checked as an integrated system in the automatic all-systems checkout.

B-9964-2

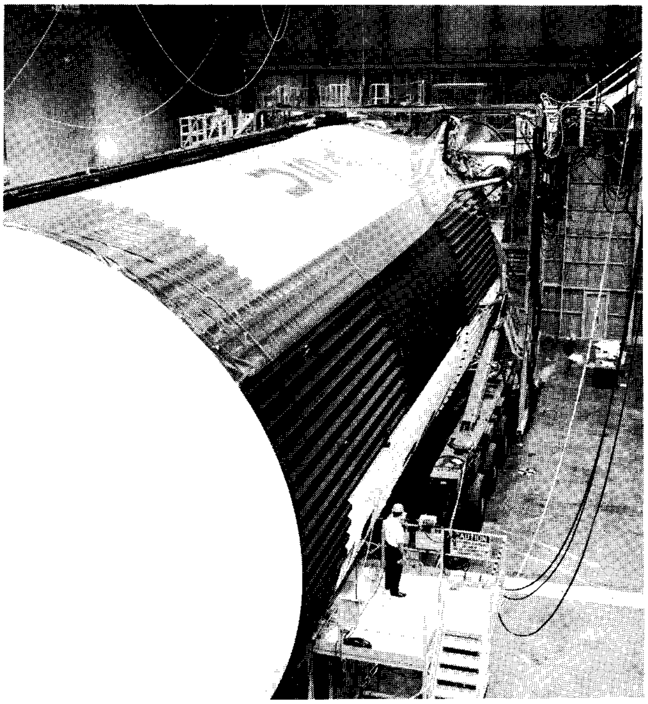
Monitoring—Technicians check booster performance during a simulated flight from a stage test control room.

After the operation of the test and checkout equipment is verified, all electrical, pneumatic, and hydraulic connections are made to the stage, resistance checks are run, and the stage undergoes physical examination.

The environmental control system is connected and checked for proper operation, and the stage's electrical circuits are physically checked for resistance. Stage electrical power is applied in sequential steps and the distribution monitored. The stage instrumentation transmission system is checked out on both coaxial hardwire and RF links. The electrical systems checkout includes checks of the power distribution circuits, heater power subsystems, destruct system, sequencing subsystem, separation subsystem, and emergency detection system.

The range safety systems undergo a complete end-to-end checkout including transmittal of RF commands to the range safety command receiver and monitoring the arm, cutoff, and destruct signals generated by the system.

Instrumentation system testing during stage checkout includes: identification of data channels, gain adjustment of signal conditioners, and checks of measurement systems, telemetry systems, and operational RF systems.



B-9908-1

First Stage in Test Cell

Pressure and leak checks are conducted on fuel and LOX tanks and associated lines, engines, fuel and LOX delivery systems, fuel and LOX pressurization systems, and the control pressure system. Checks are made of the calibration pressure switch simulation, fill and drain operation, and prevalve operation on both fuel and LOX systems.

Propulsion system checks include checks of firing command preparation and execution, engine shutdown prior to "launch commit," malfunction cutoff, and normal propulsion sequences.

Most of the above-mentioned tests are run for a second time prior to static testing and again during post-static checkout.

FIRST STAGE SYSTEMS

Fuel System

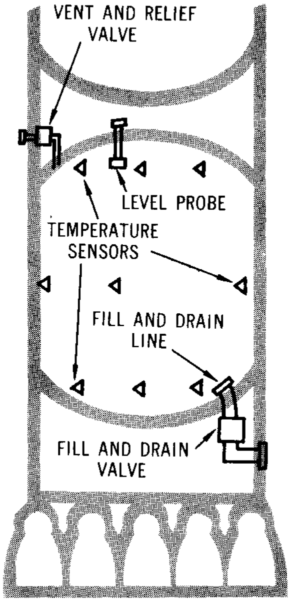
The first stage fuel system supplies RP-1 fuel to the F-1 engines. The system consists of a fuel tank, fuel feed lines, pressurization system, fill and drain components, fuel conditioning system, and associated hardware to meet the propulsion system requirements.

FUEL TANK

The fuel tank, previously described, holds 203,000 gallons of kerosene and is capable of providing 1,350 gallons of fuel per second to the engines through 10 fuel-suction lines.

FUEL FILL AND DRAIN SYSTEM

The fuel tank is filled through a 6-inch duct at the bottom of the tank. Fill rate is 200 gallons per minute until the tank is 10 per cent full. After reaching the 10 per cent mark, filling is increased to 2,000 gallons per minute until the tank is full. Normal nonemergency drain takes place through the same duct. A ball-type valve in the fill and drain line provides fuel shutoff.



Fuel Fill and Drain

The fuel fill and drain system consists of a fill and drain line, a fill and drain valve, a fuel loading level probe, and nine temperature sensors. During fuel fill, the temperature sensors provide continuous fuel temperature information used to compute fuel density. When the fuel level in the fuel tank rises to about 102 per cent of flight requirements, the fuel loading probe indicates an overload.

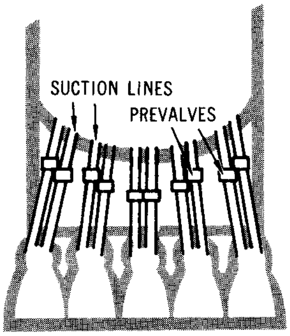
After adjusting fuel to meet requirements, the fill and drain valve is closed.

The fuel tank can be drained under pressure by closing the fuel tank vent and relief valve, supplying a pressurizing gas to the tank through the fuel tank prepressurization system, and opening the fuel fill and drain valve.

FUEL FEED SYSTEM

Ten fuel suction lines (two per engine) supply fuel from the fuel tank to the five F-1 engines. The suction line outlets attach directly to the F-1 engine fuel pump inlets.

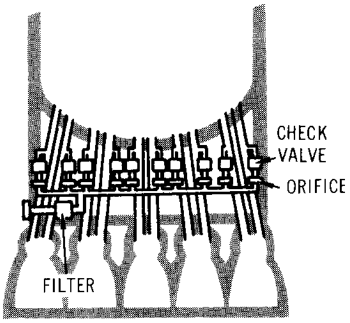
Each suction line has a pneumatically controlled fuel prevalve which normally remains open. This prevalve serves as an emergency backup to the main engine fuel shutoff valves to terminate fuel flow to the engines.



Fuel Feed

FUEL-CONDITIONING (BUBBLING) SYSTEM

The fuel-conditioning system bubbles gaseous nitrogen through the fuel feed lines and fuel tank to prevent fuel temperature stratification prior to launch. A wire mesh filter in the nitrogen supply line prevents discharge of contaminants into the conditioning system.



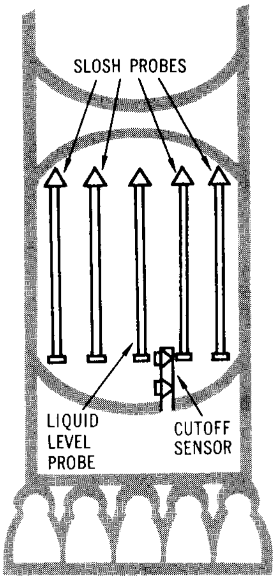
Fuel Conditioning

A check valve in the outlet of each fuel-conditioning line prevents fuel from entering the nitrogen lines when the fuel-conditioning system is not operating.

An orifice located near each fuel-conditioning check valve provides the proper nitrogen flow into each fuel duct.

FUEL LEVEL SENSING AND ENGINE CUTOFF SYSTEMS

A cutoff sensor mounted on the bottom of the fuel tank provides signal voltages to shut off fuel after a predetermined level of depletion is reached. The fuel is measured during flight by four fuel slosh probes and a single liquid level measuring probe. Fuel levels are detected electronically and reported through the stage telemetry system. Telemetry signals are transmitted to ground support either by radio frequency or, before launch, by coaxial cable. The cutoff sensor, mounted in the lower fuel tank bulkhead, initiates engine cutoff as fuel level falls below two sensing points on the probe. Engine cutoff will normally be initiated by sensors in the LOX system. The cutoff capability is provided as a backup system should fuel be depleted before LOX.



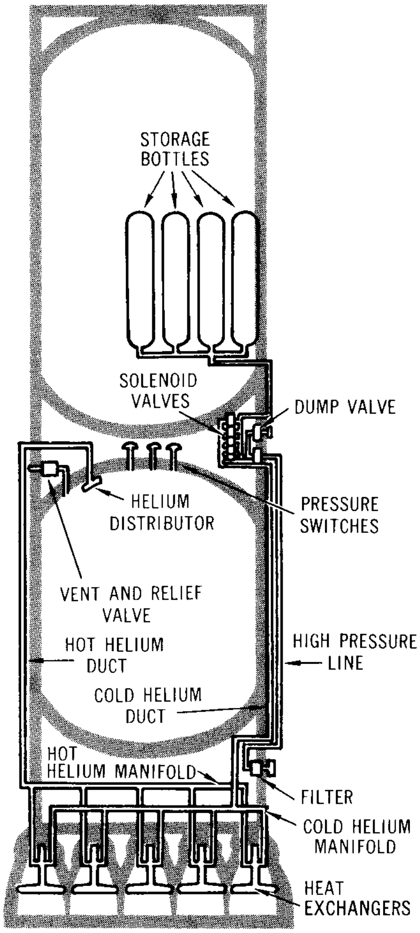
Fuel Level Sensing and Engine Cutoff

FUEL PRESSURIZATION SYSTEM

The fuel pressurization system maintains enough pressure in the fuel tank to provide proper suction at the fuel turbopumps to start and operate engines. The system consists of a helium supply, a helium flow controller, helium fill and drain components, a prepressurization subsystem, a fuel tank vent and relief valve, and associated ducts.

Four 31-cubic-foot, high pressure storage bottles in the LOX tank store the helium required for inflight pressurization of the fuel tank ullage. A high pressure line is used for filling the bottles and routing the helium to the flow controller. A solenoid dump valve is installed for emergencies. The helium flow controller uses five solenoid valves mounted parallel in a manifold to control helium flow to the fuel tank ullage. The cold helium duct routes helium from the flow controller to the cold helium manifold. From there, it is distributed to the heat exchangers on the five F-1 engines. The hot helium manifold receives the heated, expanded helium from the engine heat exchangers and routes it to the hot helium duct which then carries it through the helium distributor and on to the fuel tank ullage.

Three absolute pressure switches, mounted atop the fuel tank, monitor and control fuel tank prepressurization before engine ignition, fuel tank pressurization during flight, and overpressure.



Fuel Pressurization

Design strength of the four helium bottles at atmospheric temperatures and prior to LOX loading is about 1,660 pounds per square inch gage (psig). After LOX loading, when the bottles are cold, pressure is increased to about 3,100 psig.

A filter in the helium fill line prevents contaminants from entering the flight pressurization system.

LOX System

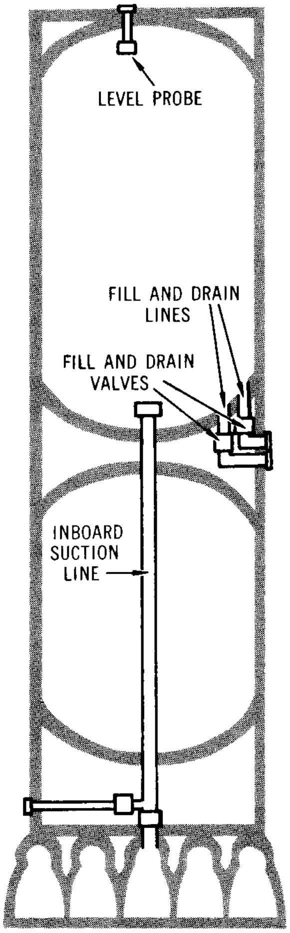
The liquid oxygen (LOX) system supplies LOX to the five F-1 engines. The system consists of a LOX tank, fill and drain components, LOX suction lines, pressurization subsystem, and associated hardware.

LOX TANK

In addition to the components of the LOX tank previously described, the tank contains internal ring baffles which line the tank walls to provide wall support and prevent excessive sloshing of LOX. A cruciform baffle in the lower tank head limits LOX swirling. Four LOX liquid level probes continuously monitor LOX level in the tank. The probes are made up of a series of continuous capacitive level sensors separated by discrete level sensors.

LOX FILL AND DRAIN SYSTEM

LOX is forced under pressure through two 6-inch LOX fill and drain lines into the tank at a slow fill rate of 1,500 gallons per minute until the tank is 6.5 per cent full. The slow fill rate avoids splash damage to LOX tank components. After a visual leak check, a fill rate of 10,000 gallons per minute takes place until the tank is 95 per cent full. After this, the rate is reduced to 1,500 gallons per minute until the LOX loading level probe senses a full tank and terminates LOX fill. In addition to the two 6-inch lines used for LOX fill and drain, a third line is available for filling the tank through the inboard suction line.



LOX Fill and Drain

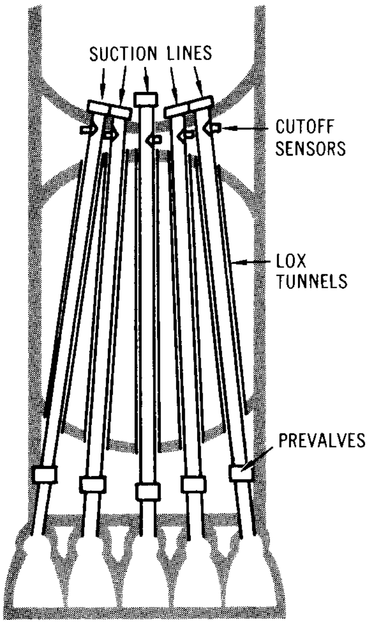
LOX boils continuously to maintain the temperature of -297 degrees Fahrenheit at sea level pressure.

It is replenished between the periods of loading and prepressurization through the fill and drain line.

Before LOX drain can be performed, the helium cylinders in the LOX tank must have their pressure decreased from about 3,100 psig to about 1,660 psig. Fill and drain valves are opened to complete drainage of the LOX tank although total evacuation of LOX from the tank requires draining the engines or waiting for boil-off of residual LOX. LOX drain can be speeded with the aid of a pressurizing gas, usually nitrogen.

LOX DELIVERY SYSTEM

LOX is delivered to the engines by five 17-inch suction lines which pass through the fuel tank in five LOX tunnels. LOX suction ducts make up the lines from the LOX tank to the prevalves in the thrust structure. The ducts are equipped with gimbals and sliding joints to counteract vibration and swelling or contraction caused by temperature. Inside the tunnels, air acts as the insulation between the LOX-wetted lines and the fuel-wetted tunnels.



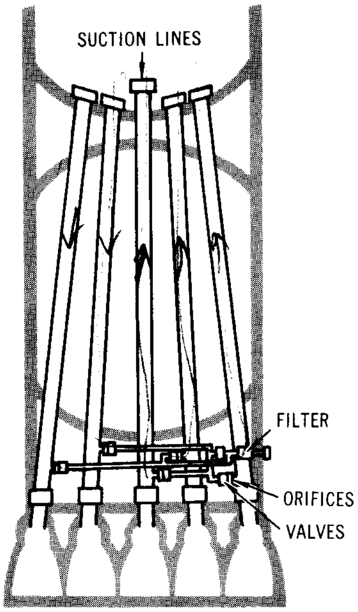
LOX Delivery

LOX level engine cutoff sensors in the suction lines assure safe engine shutdown and leave a minimum amount of unused LOX in the system.

In case of emergency, LOX prevalves in each suction line can stop the flow of LOX to the engines.

LOX CONDITIONING SYSTEM

LOX cannot exceed -297 degrees Fahrenheit or it will result in gaseous oxygen (GOX). If heat is increased, the result is boiling and not temperature increase since evaporation is a cooling process. Depth in a body of LOX can increase due to the increase in hydrostatic pressure.



LOX Conditioning

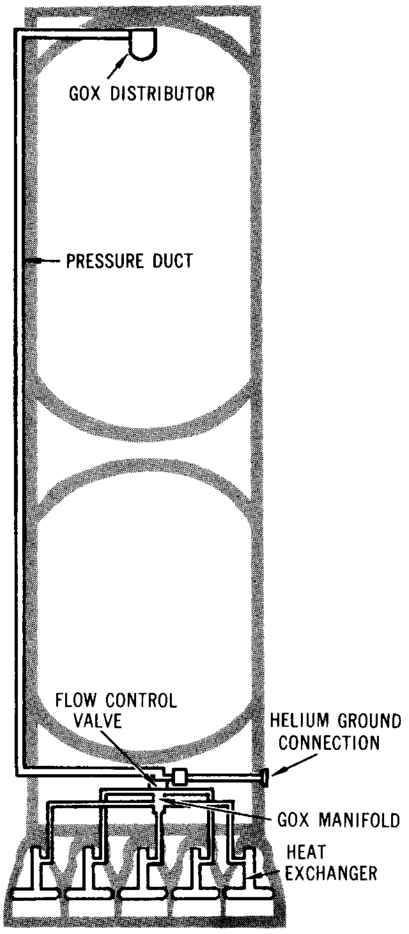
The greatest chance for overheating in the LOX system is in the transmission surface of the suction lines. Also, the suction lines are too slender for maintenance of self-contained convection currents. This situation is unacceptable since intense boiling can lead to LOX geysering, which in turn can damage the LOX tank structurally. In addition, too high a LOX temperature near the engine inlets can cause a cavity in the LOX pumps and interfere with normal engine starting. Emergency bubbling or thermal pumping is used to correct this situation.

The bubbling technique sends helium into all five suction lines to cool the LOX rapidly. Ground support supplies helium through an umbilical coupling, and filter valves and orifices control the flow of helium into the suction lines. Thermal pumping is a term used to define pumping relatively cold LOX from the LOX tank into the suction lines.

LOX PRESSURIZATION SYSTEM

Pressurizing gases used in the LOX tank are helium, gaseous oxygen, and nitrogen. These gases are used in prepressurization, flight pressurization. and storage pressurization.

Prepressurization is necessary 45 seconds prior to engine ignition to give sufficient tank ullage pressure for engine start and thrust buildup. Helium, used as the pressurizing gas to reduce flight weight, is supplied by ground support through the helium ground connection. It proceeds up the gaseous oxygen line into the LOX tank through the GOX distributor. The flow of helium is monitored by the pressure duct and stopped at 26 pounds per square inch absolute (psia) maximum and is resumed when the pressure drops to 24.2 psia during engine start. Ground-supplied helium is available until liftoff. GOX is added to the LOX tank for pressurization during flight. Each engine contributes to GOX pressurization. A portion of LOX—6,340 pounds—passing through the engine is diverted from the LOX dome into the engine heat exchanger where hot gases exhausted from each engine turbine transform LOX into GOX. The GOX flows from each heat exchanger into the GOX line manifold through the flow control valve, up the GOX line, and into the LOX tank through the GOX distributor. The GOX flow is approximately 40 pounds per second to maintain a LOX tank ullage pressure of 18 to 23 psia.

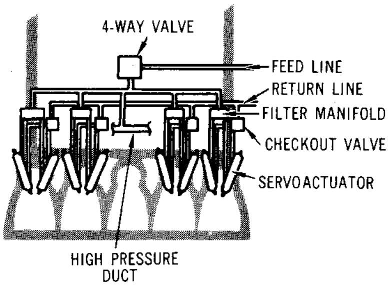


LOX Pressurization

While the booster is being stored or transferred from one location to another, a slight positive nitrogen pressure is maintained for cleanliness and low humidity conditions. The external nitrogen pressure source is removed during flight operations.

Fluid Power System

An unusual but convenient type of fluid power or hydraulic system is in use on the Saturn V first stage. It incorporates the same types of fuels—RP-1 and RJ-1 (kerosene)—that are used in the stage fuel system. Ordinarily a different and weaker type of fluid is used for hydraulics. This system eliminates the use of a separate pumping system.



Fluid Power System

The fluid power system provides ground and flight fluid power for valve actuation and thrust vectoring. It gives power primarily to the engine start system and the engine gimbaling system. Its source is the fuel system. RJ-1 is provided from the ground before liftoff, and RP-1 is supplied from the fuel tank during flight.

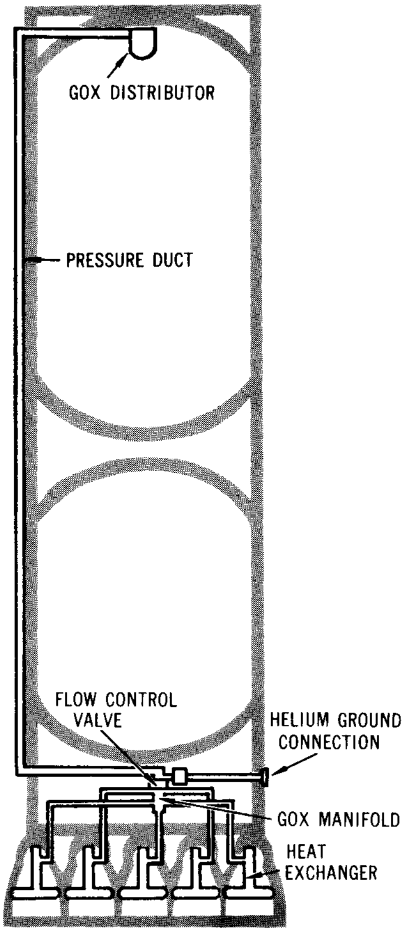
The ground supply of RJ-1 is routed to all five engines at 1,500 psig and eventually back to the ground supply. After ignition, RP-1 is routed from the high pressure fuel duct to the servoactuators for hydraulic power to position the engines.

The center engine, which has no thrust vectoring system, directs its hydraulic fluid through the feed line and 4-way hydraulic control valve to supply pressure to the closing ports of the gas generator, main fuel valves, and main LOX valves. The fuel passes through orifices and then is ducted through the ground checkout valve and back to ground supply through the return line.

The four outboard engines direct RJ-1 through the servoactuators to the ground checkout valve where it is returned through a coupling to ground supply.

Electrical System

The electrical power and distribution system of the engine ignition to give sufficient tank ullage pressure for engine start and thrust buildup. Helium, used as the pressurizing gas to reduce flight weight, is supplied by ground support through the helium ground connection. It proceeds up the gaseous oxygen line into the LOX tank through the GOX distributor. The flow of helium is monitored by the pressure duct and stopped at 26 pounds per square inch absolute (psia) maximum and is resumed when the pressure drops to 24.2 psia during engine start. Ground-supplied helium is available until liftoff. GOX is added to the LOX tank for pressurization during flight. Each engine contributes to GOX pressurization. A portion of LOX—6,340 pounds—passing through the engine is diverted from the LOX dome into the engine heat exchanger where hot gases exhausted from each engine turbine transform LOX into GOX. The GOX flows from each heat exchanger into the GOX line manifold through the flow control valve, up the GOX line, and into the LOX tank through the GOX distributor. The GOX flow is approximately 40 pounds per second to maintain a LOX tank ullage pressure of 18 to 23 psia.



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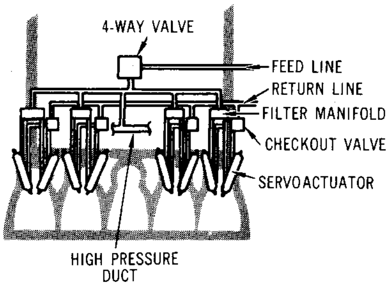

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Electrical System

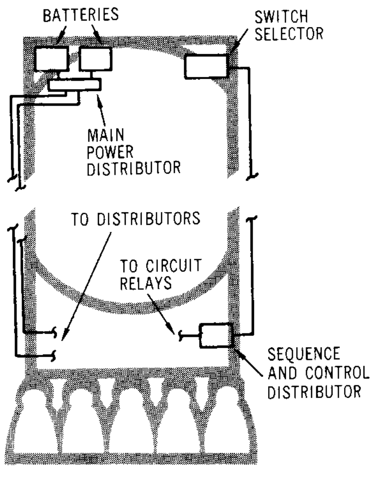
The electrical power and distribution system of the first stage provides power for controlling and measuring functions of the vehicle. The system operates during static firing, launch preparation and checkout, launch, and flight.

The electrical system consists of two batteries, a main power distributor, a sequence and control distributor, propulsion distributor, timer distributor, measuring distributors, thrust OK distributor, and measuring power distributor.

Two independent 28-volt DC power systems are installed on the stage. System No. 1, the main power battery, energizes the stage controls. The battery has a 640-ampere-minute rating, weighs about 22 pounds, and is used to control various solenoids. Battery No. 2, the instrumentation battery, energizes the flight measurement system and gives power to redundant systems for greater mission reliability. It has a 1,250-ampere-minute rating and weighs approximately 55 pounds. The range safety system can be operated by either battery.

Preflight power is supplied from ground equipment through umbilical connections. The supply for each system is 28 volts. Ground sources supply power for heaters, ignitors, and valve operators that are not operated during flight.

The distributors subdivide the electrical circuits and serve as junction boxes. Both electrical systems share the same distributors. The main power distributor houses relays, the power transfer switch, and electrical distribution buses. The relays control circuits that must be time-programmed. The motor-operated, multi-contact, power transfer switch transfers the stage load from the ground supply to the stage batteries. The transfer is tried several times during countdown to verify operation. Power is distributed by the main buses.



Electrical System

The switch selector, actuated by the instrument unit (IU), commands the sequence and control distributor, which in turn amplifies the signals received. The sequence and control distributor then energizes the various circuit relays required to implement the flight program. The switch selector is an assembly of redundant low power relays and transistor switches, which control the sequence and control distributor. It is activated by a coded signal from the instrument unit computer.

The propulsion distributor contains the monitor and control circuits for the propulsion system.

The thrust OK distributor contains the circuits that shut down the engines when developed thrust is inadequate. Two of the three thrust OK switches must operate or the engine will be shut down.

The timer distributor houses the circuits to delay the operation of relay valves and other electromechanical devices. The programmed delays are essential for optimum performance and safety.

The measuring power distributor contains electrical buses, and the measuring distributors route data from measuring racks, serve as measurement signal junction boxes, and switch data between the hardwire and telemetry.

Instrumentation System

The first stage instrumentation system measures and reports information on stage systems and components and provides data on internal and external environments. It keeps abreast of approximately 900 measurements on the stage, such as measurements of valve positions, propellant levels, temperatures, voltages, and pressures. The measurements are telemetered by coaxial cable to ground support equipment and by radio frequency transmission to ground stations.

The instrumentation system consists of a measurement system, a telemetry system, and the Offset Doppler tracking system. A remote automatic calibration system provides remote rapid checkout of the measurements and telemetry systems.

MEASUREMENT

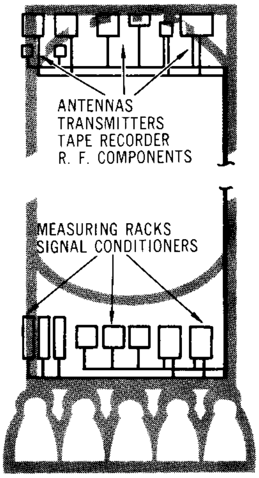
The measurement system reports environmental situations and how the first stage reacts to them. Making use of transducers, signal conditioners, measuring rack assemblies, measuring distributors, and the onboard portion of the remote automatic calibration system, this system involves many phases of stage operation. Included are measurements of acceleration, acoustics, current, flow, flight angles, valve position, pressure, RPM's, stress, temperature, vibration, and separation.

TELEMETRY

Telemetry is a method of remote monitoring of flight information accomplished by means of a radio link. The first stage telemetry system is composed of six radio frequency links.

Most of the components of the telemetry systems are located in the thrust structure; RF assemblies and a tape recorder are located in the forward skirt. The telemeter transmits data through two common antenna systems.

Links F1, F2, and F3 are identical systems which transmit narrow-band, frequency-type data such as that generated by strain gages, temperature gages, and pressure gages. The system can handle 234 measurements on a time-sharing basis and 14 measurements transmitted continously. Data may be sampled either 120 times per second or 12 times per second.



Telemetry

Links S1 and S2 transmit wide-band, frequency-type data generated by vibration sensors. Each link provides 15 continuous channels or a maximum of 75 multiplexed channels depending on the specific measuring program.

Telemeter P1 transmits either pulse code modulated or digital type data. Five multiplexers, four analogs, and one digital supply data to the PCM assembly. This provides the most accurate data and is used for ground checkout as well.

A telemetering calibrator is used to improve the accuracy of the telemetry systems. The calibrator supplies known voltages to the telemeters periodically during the stage operation. Their reception at tracking stations provides a valid reference for data reduction.

The effects of ullage and retrorocket firing attenuation can seriously degrade the telemetry transmission during stage separation; therefore, a tape recorder installed in the forward skirt records data for delayed transmission. The commands for tape recorder operation originate in the digital computer located in the instrument unit.

ODOP SYSTEM (Offset Doppler Tracking System)

The ODOP system is an elliptical tracking system that measures the rate of motion at which the vehicle is moving away from or toward a tracking station. The total Doppler shift in the frequency of a continuous wave, ultra-high frequency signal transmitted from the ground to the first stage is measured. The signal is received by the transponder at the stage, modified, and then retransmitted back to the ground. Retransmitted signals are received simultaneously by three tracking stations. Separate antennas on the stage are used for receiving and retransmitting the signals.

SEPARATION SYSTEM

A redundant initiation system actuates the separation of the first stage from the second stage. A command signal for arming and another for firing the initiation systems are programmed by the instrument unit computer.

After LOX depletion, the computer signals operate relays in the switch selector and sequence and control distributor to control the exploding bridgewire firing units. When armed, the firing units store a high voltage electrical charge. When fired, the electrical charge actuates the ordnance.

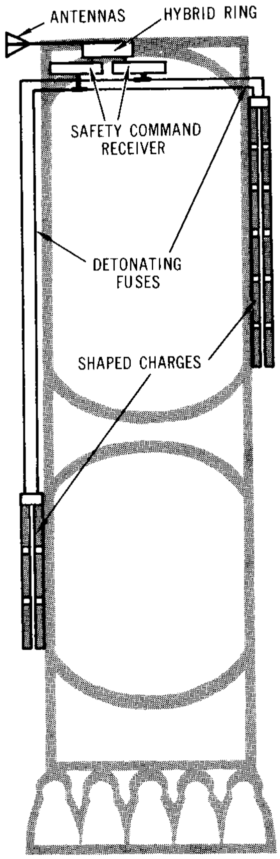
Two firing units are installed on the first stage for the eight retrorockets, and two are installed on the second stage for the separation ordnance.

Range Safety System

The function of the range safety system is to provide ground command with the capability of flight termination by shutting off the engines, blowing open the stage propellant tanks, and dispersing the fuel in event of a flight malfunction.

The system is redundant, consisting of two identical, independent systems, each made up of electronic and ordnance subsystems.

Flight termination by way of the range safety system goes into effect upon receipt of the proper radio frequency commands from the ground. A frequency-modulated RF signal transmitted from the ground range safety transmitter is received by the antennas and transmitted by way of a hybrid ring to the range safety command receiver. There, the signal is conditioned, demodulated, and decoded.



Range Safety System

The resulting signal simultaneously causes arming of the exploding bridgewire firing unit and shutdown of the stage engines. A second command signal transmitted by the ground range safety transmitter ignites the explosive train (detonating fuses and shaped charges) to blow open the stage propellant tanks.

Control Pressure System

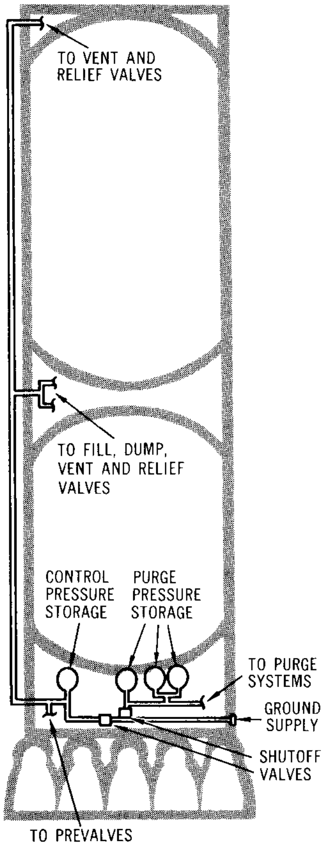
The control pressure system supplies pressurized gaseous nitrogen for the pneumatic actuation of propellant system valves and purging of various F-1 engine systems.

The complete integrated system is made up of an onboard control pressure system, a ground control pressure system, and an onboard purge pressure system. The object in each system is to deliver an actuating or purge medium to an interfacing stage system.

ONBOARD CONTROL PRESSURE SYSTEM

The onboard control pressure system consists of a high-pressure nitrogen storage bottle, an umbilical coupling and tubing assembly for filling the storage bottle, a manifold assembly, and control valves at the terminal ends of various nitrogen distribution lines. In some cases, two valves are paired with other associated equipment and block-mounted to form a control assembly.

The nitrogen onboard storage bottle has 2,200-cubic inch capacity and is made of titanium alloy. It is designed for a maximum proof pressure of 5,000 psig. It is filled and discharged through a port in the single boss. During flight launch preparation, the bottle is filled from a ground supply first to a pressurization of 1,600 psig well in advance of final countdown. This weight pressure is adequate for any prelaunch operational use. The second step occurs in the last hour of the launch countdown and



Control Pressure System

brings the storage bottle pressure up to its normal capacity of 3,250 plus or minus 50 psig.

The manifold assembly serves as a gaseous nitrogen central receiving and distributing center as well as a mounting block for filters, shutoff solenoid valves, a pressure regulator, a relief valve, and pressure transducers. Ported manifolds provide tubing assembly connections to the storage bottle, umbilical coupling, and various tubing assembly distribution lines to control valves throughout the stage.

GROUND CONTROL PRESSURE SYSTEM

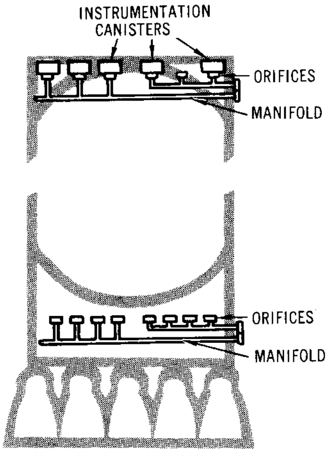
The ground control pressure system provides a direct ground pressure supply for some of the first stage pneumatically actuated valves. The valves are involved with propellant fill and drain and emergency engine shutdown system operations. Direct ground control assures a backup system in case of emergency and conserves the onboard nitrogen supply.

ONBOARD PURGE PRESSURE SYSTEM

The onboard purge pressure system consists of three high-pressure nitrogen storage bottles identical to the onboard control pressure storage bottle, an umbilical coupling and tubing for filling the bottles, and a manifold assembly and tubing for receiving and delivering the gas to the engine and calorimeter purge systems. These purge systems expel propellant leakage and are necessary from the time of loading throughout flight.

Environmental Control System

The environmental control system protects stage equipment from temperature extremes in both the forward skirt and thrust structure areas and provides a nitrogen purge during prefiring and firing operations.



Environmental Control System

Temperature-controlled air is provided by a ground air conditioning unit from approximately 14 hours before launch to approximately 6 hours before launch. At this time, gaseous nitrogen from an auxiliary nitrogen supply unit is introduced into the system and used to purge and condition the forward skirt and thrust structure areas until umbilical disconnect at launch.

A distribution manifold vents air and gaseous nitrogen through orifices into the thrust structure to maintain proper temperature. Air and nitrogen are supplied from the ground.

The system also distributes air and gaseous nitrogen to instrumentation canisters mounted in the forward skirt. Temperatures in the canisters are held to meet requirements of electrical equipment. From the canisters, the conditioning gas is vented into the forward skirt compartment.

Visual Instrumentation

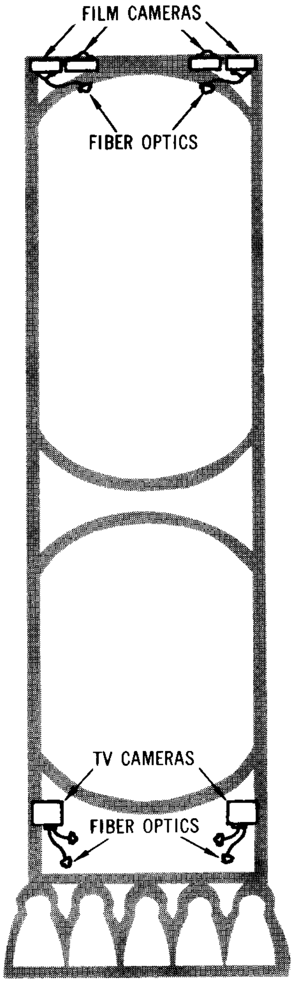
Visual instrumentation, presently planned to be installed on two flight stages, is designed to monitor critical stage functions prior to and during static test and flight conditions.

FILM CAMERAS

The first stage film cameras provide photographic coverage of the LOX tank interior during launch, flight, and separation. The stage carries four film cameras. The two LOX-viewing cameras will provide color motion pictures to show the following: behavior of the liquid oxygen, possible wave or slosh motions, and cascading or waterfall effects of the liquid from the internal tank structure. The capsules, which contain the cameras, are ejected automatically about 25 seconds after separation and are recovered after descent into the water. First stage flight versions of the camera consist of the LOX tank-viewing configuration plus two direct-viewing stage separation capsules. The installation is in the forward skirt area. The tank-viewing optical lenses and the two strobe flash light assemblies are mounted in the LOX tank manhole covers. Connecting the remotely located camera capsules and the flash head are the optical assemblies. consisting of coupling lens attached to the ejection tube, a 9-foot length of fiber optics, and the objective lens mounted in the flash-head assembly. The equipment required to complete the system, such as batteries, power supplies, timer, and synchronizing circuitry, is contained in the environmentally controlled equipment racks or boxes mounted around the interior of the forward skirt structure. The combined timer and synchronizing unit serves two functions. The digital pulse timer supplies real time correlation pulses which are printed on one edge of the film. The timer also supplies event marker pulses to the opposite edge of the film to record selected significant events such as liftoff, engine shutdown, and stage separation. The synchronizing unit times the intermittent illumination provided by the strobe lamps to coincide with the open portion of the rotating shutter as it passes the motion picture film gate. The capsule assembly consists of the heavy nose section and quartz window, which protect the capsule during re-entry heating and impact on the water. The body of the capsule, including the camera, is sealed and watertight. A paraloon and drag skirt aid its descent and flotation. A radio beacon and flashing light are mounted on the capsule to aid in recovery.

TELEVISION SYSTEM

The television system on the first stage will transmit four views of engine operation and other engine area functions in the interval from fueling to first stage separation. The system utilizes two split fiber optics viewing systems and two cameras. Extremes in radiant heat, acoustics, and vibration prohibit the installation of the cameras in the engine area; therefore, fiber optics bundles are used to transmit the images to the cameras located in the thrust structure. Quartz windows are used to protect the lens. Both nitrogen purging and a wiping action are used to prevent soot buildup on the protective window.



Visual Instrumentation

Image enhancement improves the fiber-optical systems by reducing the effects of voids between fibers and broken fibers. An optically flat disc with parallel surfaces rotates behind each objective lens.

The drive motor rotates in synchronism with the master drive motor. A DC to AC inverter energizes the synchronous drive motors. A camera control unit houses amplifiers, fly back, sweep, and other circuits required for the video system. Each vidicon output (30 frames/second) is amplified and sampled every other frame (15 frames/second) by the video register. A 2.5 watt FM transmitter feeds the 7-element yagi antenna array covered by a radome.

FIRST STAGE FLIGHT

The first stage is loaded with RP-1 fuel and LOX at approximately 12 and 4 hours respectively, before launch. With all systems in a ready condition, the stage is ignited by sending a start signal to the five F-1 rocket engines. The engine main LOX valves open first allowing LOX to begin to enter the main thrust chamber. Next the engines' gas generators and turbopumps are started. Each engine's turbopump assembly will develop approximately 60,000 horsepower. Combustion is initiated by injecting a hypergolic solution into the engine's main thrust chamber to react with the LOX already present. The main fuel valves then open, and fuel enters the combustion chamber to sustain the reaction previously initiated by the LOX and hypergolic solution. Engine thrust then rapidly builds up to full level. The five engines are started in a 1-2-2 sequence, the center engine first and opposing outboard pairs at 300-millisecond stagger times. The stage is held down while the engines build up full thrust. After full thrust is reached and all engines and stage systems are functioning properly, the stage is released. This is accomplished by a "soft release" mechanism. First, the restraining hold-down arms are released. Immediately thereafter the vehicle begins to ascend but with a restraining force caused by tapered metal pins being pulled through holes. This "soft release" lasts for about 500 milliseconds.

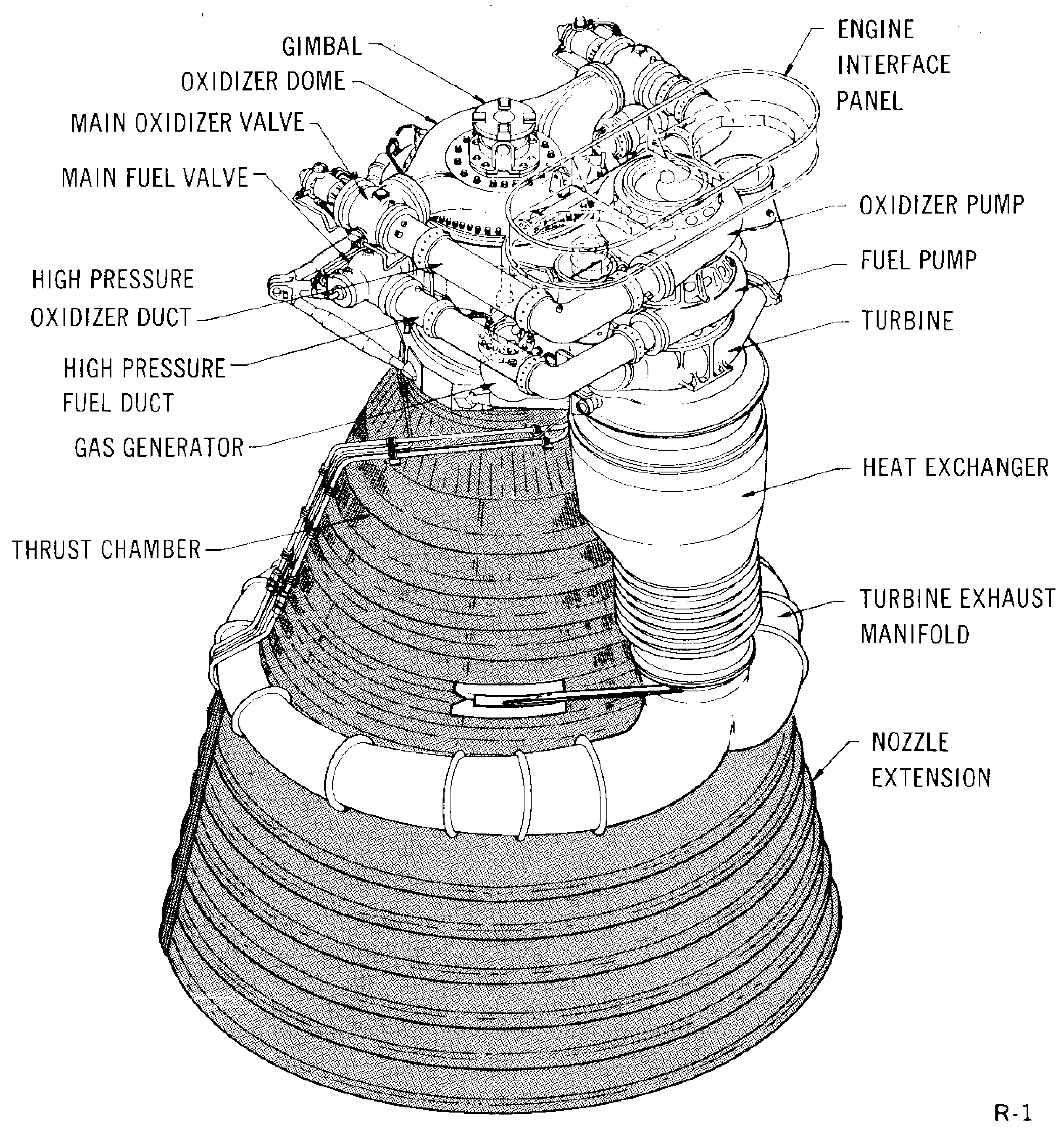
The vehicle rises vertically to an altitude of approximately 430 feet to clear the launch umbilical tower and then begins a pitch and roll maneuver to attain the correct flight azimuth. As the vehicle continues its flight, its path is controlled by gimbaling the outboard F-1 engines consistent with a preprogrammed flight path and commanded by the instrument unit.

At approximately 69 seconds into the flight, the vehicle experiences a condition of maximum dynamic pressure. At this time, the restraining drag force is approximately equal to 460,000 pounds.

At 135.5 seconds into the flight most of the LOX and fuel will be consumed, and a signal is sent from the instrument unit to shut down the center engine. The outboard engines continue to burn until either LOX or fuel depletion is sensed. LOX depletion is signaled when a "dry" indication is received from at least two of the four LOX cutoff sensors; one sensor is located near the top of each outboard LOX suction duct. Fuel depletion is signaled by a "dry" indication from a redundant fuel cutoff sensor bolted directly to the fuel tank lower bulkhead. The LOX depletion cutoff is the main cutoff system with fuel cutoff as the backup.

Six hundred milliseconds after the outboard engines receive a cutoff signal, a signal is given to fire the first stage retrorockets. Eight retrorockets are provided and each produces an average effective thrust of 88,600 pounds for 0.666 seconds. The first stage separates from the second stage at an altitude of about 205,000 feet. It then ascends to a peak altitude near 366,000 feet before beginning its descent. While falling, the stage assumes a semistable engines down position and impacts into the Atlantic Ocean at approximately 350 miles down range of Cape Kennedy.

F-1 ENGINE FACT SHEET



LENGTH	19 ft.
WIDTH	12 ft. 4 in.
THRUST (sea level)	1,500,000 lb.
SPECIFIC IMPULSE (minimum)	260 sec.
RATED RUN DURATION	150 sec.
FLOWRATE: Oxidizer	3,945 lb./sec. (24,811 gpm)
Fuel	1,738 lb./sec. (15,471 gpm)
MIXTURE RATIO	2.27:1 oxidizer to fuel
CHAMBER PRESSURE	965 psia
WEIGHT FLIGHT CONFIGURATION	18,500 lb. maximum
EXPANSION AREA RATIO	16:1 with nozzle extension 10:1 without nozzle extension
COMBUSTION TEMPERATURE: Thrust Chamber	5,970°F
Gas Generator	1,465°F
MAXIMUM NOZZLE EXIT DIAMETER	11 ft. 7 in.

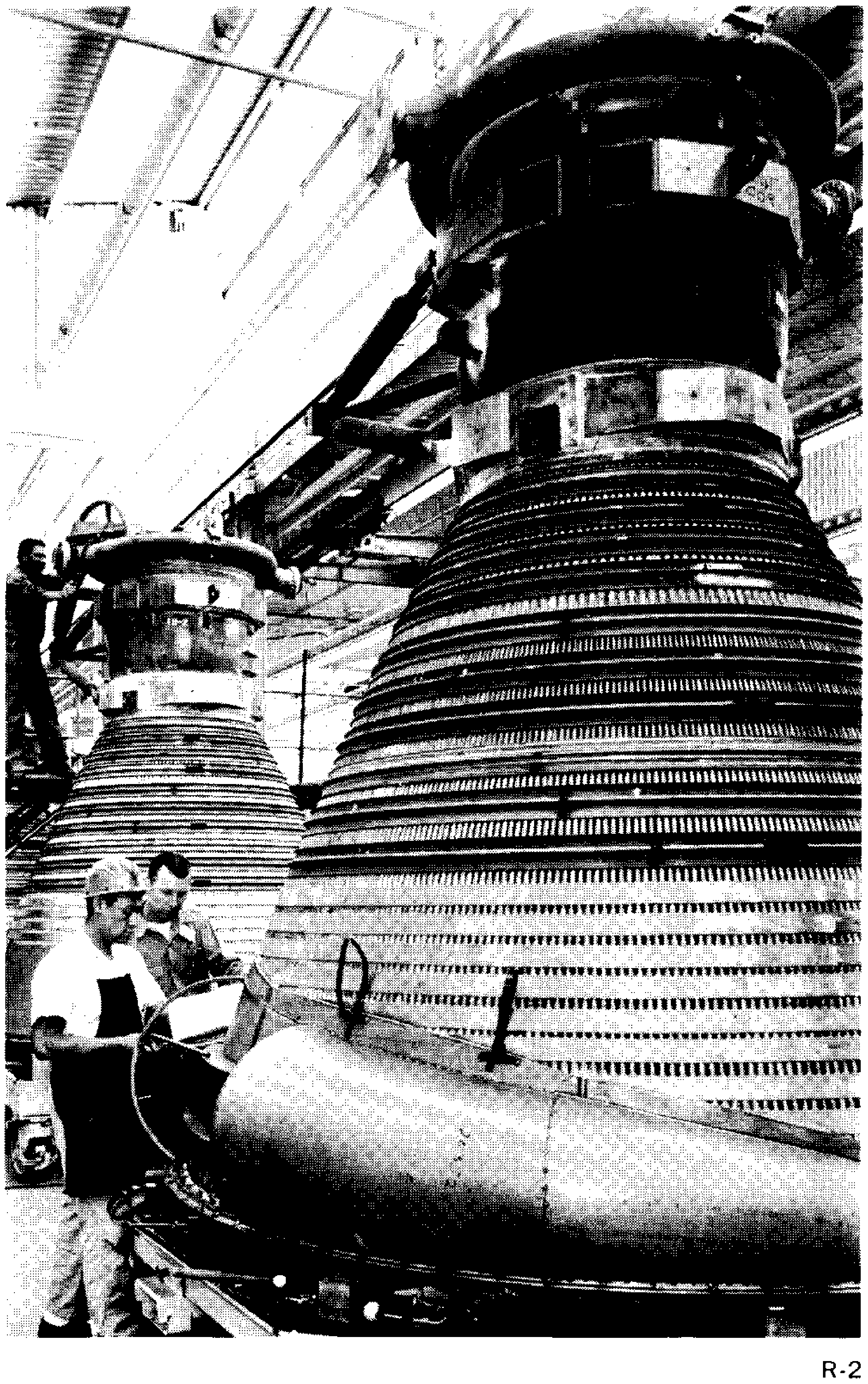
NOTE: F-1 engine will be uprated to 1,522,000 lb. thrust for Vehicle 504 and all subsequent operational vehicles.

F-1 ENGINE

ENGINE DESCRIPTION

The F-1 engine is a single-start, 1,500,000-pound fixed-thrust, bipropellant rocket system. The engine uses liquid oxygen as the oxidizer and RP-1 (kerosene) as fuel. The engine is bell-shaped, with an area expansion ratio—the ratio of the area of the throat to the base—of 16:1. RP-1 and LOX are combined and burned in the engine's thrust chamber assembly. The burning gases are expelled through an expansion nozzle to produce thrust. The five-engine cluster used on the first stage of the Saturn V produces 7,500,000 pounds of thrust. All of the engines are identical with one exception. The four outboard engines gimbal; the center engine does not.

The major engine systems are the thrust chamber assembly, the propellant feed system, the turbopump, the gas generator system, the propellant tank pressurization system, the electrical system, the hydraulic control system, and the flight instrumentation system.



R-2

Assembly—Thrust chambers of the F-1 rocket engine—the most powerful engine under development by the United States—are assembled in this manufacturing line.

THRUST CHAMBER ASSEMBLY

The thrust chamber assembly consists of a gimbal bearing, an oxidizer dome, an injector, a thrust chamber body, a thrust chamber nozzle extension, and thermal insulation. The thrust chamber assembly receives propellants under pressure supplied by the turbopump, mixes and burns them, and imparts a high velocity to the expelled combustion gases to produce thrust. The thrust chamber assembly also serves as a mount or support for all engine hardware.

Gimbal Bearing

The gimbal bearing secures the thrust chamber assembly to the vehicle thrust frame and is mounted on the oxidizer dome. The gimbal is a spherical, universal joint consisting of a socket-type bearing with a bonded Teflon-fiberglass insert which provides a low-friction bearing surface. It permits a maximum pivotal movement of 6 degrees in each direction of both the X and Z axes (roughly analogous to pitch and yaw) to facilitate thrust vector control. The gimbal transmits engine thrust to the vehicle and provides capability for positioning and thrust alignment.

Oxidizer Dome

The oxidizer dome serves as a manifold for distributing oxidizer to the thrust chamber injector, provides a mounting surface for the gimbal bearing, and transmits engine thrust forces to the vehicle structure. Oxidizer at a volume flowrate of 24,811 gpm enters the dome through two inlets positioned 180 degrees apart (to maintain even distribution of the propellant).

Thrust Chamber Injector

The thrust chamber injector directs fuel and oxidizer into the thrust chamber in a pattern which ensures efficient and satisfactory combustion. The injector is multi-orificed with copper fuel rings and copper oxidizer rings forming the face (combustion side) of the injector and containing the injection orifice pattern. Assembled to the face are radial and circumferential copper baffles which extend downward and compartmentalize the injector face. The baffles and rings, together with a segregated igniter fuel system, are installed in a stainless steel body.

Oxidizer enters the injector from the oxidizer dome. Fuel enters the injector from the thrust chamber fuel inlet manifold, and in order to facilitate the engine start phase and to reduce pressure losses, part of the flow is introduced directly into the thrust chamber. The remaining fuel (controlled by orifices) flows through alternate tubes which run the length of the thrust chamber body to the nozzle exit. There, it enters a return manifold and flows back to the injector through the remaining tubes.

Thrust Chamber Body

The thrust chamber body provides a combustion chamber for burning propellants under pressure and an expansion nozzle for expelling gases produced by the burned propellants at the high velocity required to produce the desired thrust. The thrust chamber is tubular-walled and regeneratively fuel-cooled, and the nozzle is bell-shaped. There are four sets of outrigger struts attached to the exterior of the thrust chamber; two sets of the struts are turbopump mounts and the other two are attach points for the vehicle contractor's gimbal actuators. The thrust chamber incorporates a turbine exhaust manifold at the nozzle exit and a fuel inlet manifold at the injector end which directs fuel to the fuel down tubes. Brackets and studs welded to the reinforcing "hatbands" surrounding the thrust chamber provide attach points for thermal insulation blankets.

Fuel enters the fuel inlet manifold through two diametrically opposed inlets. From the manifold, 70 per cent of the fuel is diverted through 89 alternate CRES "down" tubes the length of the chamber. A manifold at the nozzle exit returns the fuel to the injector through the remaining 89 return tubes. The fuel flowing through the chamber tubes provides regenerative cooling of the chamber walls during engine operation. The thrust chamber tubes are bifurcated; that is, they are comprised of a primary tube from the fuel manifold to the 3:1 expansion ratio area. At that point, two secondary tubes are spliced into each primary tube. This is necessary to maintain a desired cross-sectional area in each of the tubes through the large-diameter belled nozzle section.

The turbine exhaust manifold, which is fabricated from preformed sheet metal shells and which forms a torus around the aft end of the thrust chamber body, receives turbine exhaust gases from the heat exchanger. Upon entering the manifold, the gases are distributed uniformly. As the gases are expelled from the manifold, flow vanes in the exit slots provide uniform static pressure distribution in the nozzle extension. Radial expansion joints compensate for thermal growth of the manifold.

Thrust Chamber Nozzle Extension

The thrust chamber nozzle extension increases the expansion ratio of the thrust chamber from 10:1 to 16:1. It is a detachable unit that is bolted to the exit end ring of the thrust chamber. The interior of the nozzle extension is protected from the engine exhaust gas environment (5800 Fahrenheit) by film cooling, using the turbine exhaust gases (1200 Fahrenheit) as the coolant. The gases enter the extension between a continuous outer wall and a shingled inner wall, pass out through injection slots between the shingles, and flow over the surfaces of the shingles forming a boundary layer between the inner wall of the nozzle extension and the hotter exhaust gases exiting from the main engine combustion chamber. The nozzle extension is made of high strength stainless steel.

Hypergol Cartridge

The hypergol cartridge supplies the fluid to produce initial combustion in the thrust chamber. The cartridge, which is cylindrical and has a burst diaphragm welded to either end, contains a hypergolic fluid consisting of 85 per cent triethylborane and 15 per cent triethylaluminum. As long as the fluid is in the hermetically sealed cartridge, it is stable, but it will ignite spontaneously upon contact with oxygen in any form. During the start phase of operation, increasing fuel pressure in the igniter fuel system ruptures the burst diaphragms. The hypergolic fluid and the fuel enter the thrust chamber through a segregated igniter fuel system in the injector and contact the oxidizer. Spontaneous combustion occurs and thrust chamber ignition is established.

Pyrotechnic Igniter

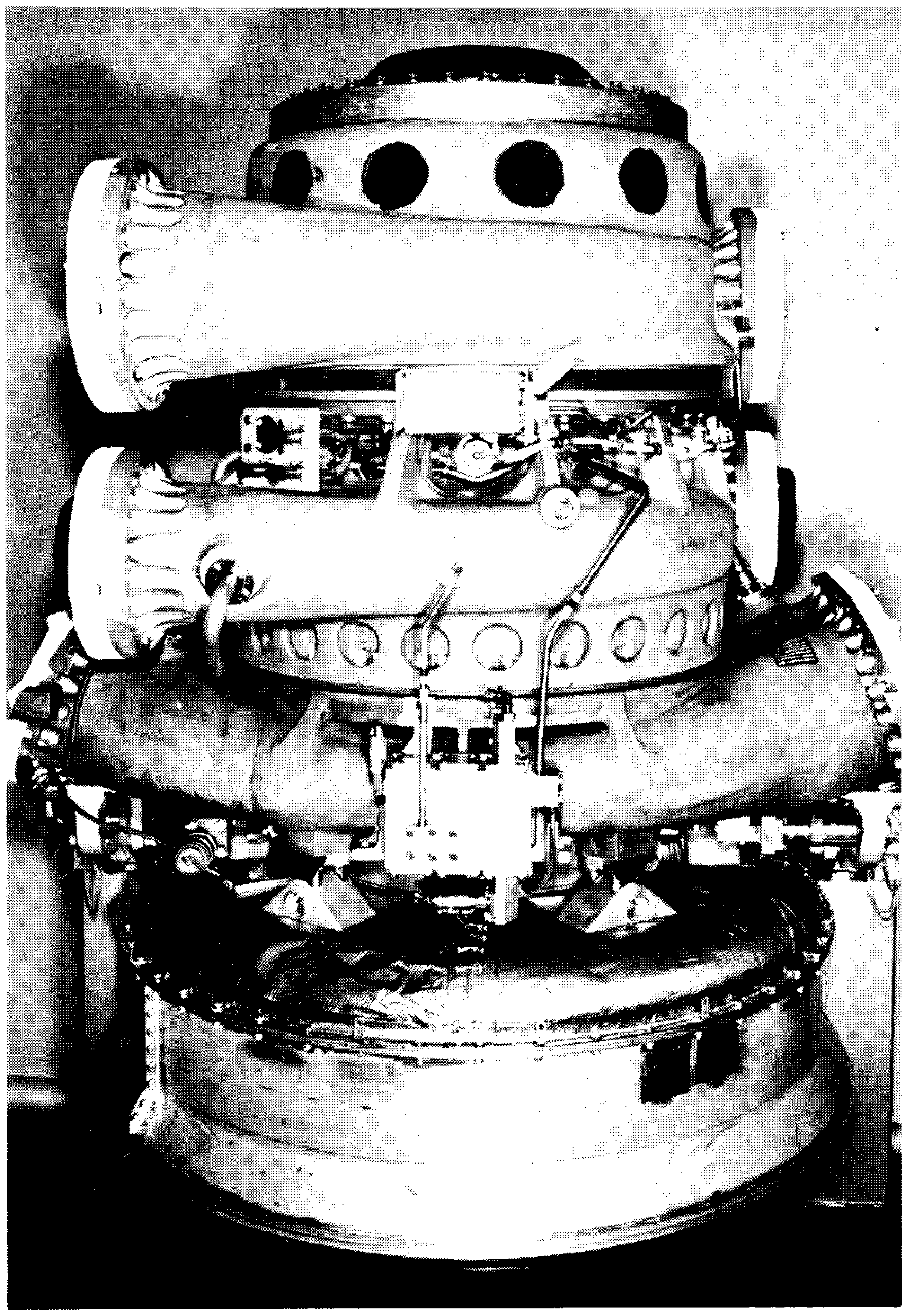
Pyrotechnic igniters, actuated by an electric spark, provide the ignition source for the propellants in the gas generator and re-ignite the fuel-rich turbine exhaust gases as they exit from the nozzle extension.

Thermal Insulation

The thermal insulation protects the F-1 engine from the extreme temperature environment (2550 Fahrenheit maximum) created by radiation from the exhaust plume and backflow during clustered-engine flight operation. Two types of thermal insulators are used on the engine—foil-batt on complex surfaces and asbestos blankets on large, simple surfaces. They are made of lightweight material and are equipped with various mounting provisions, such as grommeted holes, clamps, threaded studs, and safetywire lacing studs.

TURBOPUMP

The turbopump is a direct-drive unit consisting of an oxidizer pump, a fuel pump, and a turbine mounted on a common shaft. The turbopump delivers fuel and oxidizer to the gas generator and the thrust chamber. LOX enters the turbopump axially through a single inlet in line with the shaft and is discharged tangentially through dual outlets. Fuel enters the turbopump radially through dual inlets and is discharged tangentially through dual outlets. The dual inlet and outlet design provides a balance of radial loads in the pump.


R-4A

F-1 Turbopump

Three bearing sets support the shaft. Matched tandem ball bearings, designated No. 1 and No. 2, provide shaft support between the oxidizer and fuel pumps. A roller bearing, No. 3, provides shaft support between the turbine wheel and the fuel pump. The bearings are cooled with fuel during pump operation. A heater block provides the outer support for No. 1 and No. 2 bearings, and is used during LOX chilldown of the oxidizer pump to prevent freezing of the bearings.

A gear ring installed on the shaft is used in conjunction with the torque gear housing for rotating the pump shaft by hand, and also is used in conjunction with a magnetic transducer for monitoring shaft speed.

There are nine carbon seals in the turbopump: primary oxidizer seal, oxidizer intermediate seal, lube seal No. 1 bearing, lube seal No. 2 bearing, primary fuel seal, fuel inlet seal, fuel inlet oil seal, hot-gas secondary, and hot-gas primary seal.

The main shaft and the parts attaching directly to it are dynamically balanced prior to final assembly on the turbopump.

Oxidizer Pump

The oxidizer pump supplies oxidizer to the thrust chamber and gas generator at a flowrate of 24,811 gpm. The pump consists of an inlet, an inducer, an impeller, a volute, bearings, seals, and spacers. Oxidizer is introduced into the pump through the inlet which is connected by duct to the oxidizer tank. The inducer in the inlet increases the pressure of the oxidizer as it passes into the impeller to prevent cavitation. The impeller accelerates the oxidizer to the desired pressure and discharges it through diametrically opposed outlets into the high-pressure oxidizer lines leading to the thrust chamber and gas generator.

The oxidizer inlet, which attaches to a duct leading to the vehicle oxidizer tank, is bolted to the oxidizer volute. Two piston rings seated between the inlet and the volute expand and contract with temperature changes to maintain an effective seal between the high and low pressure sides of the inlet. Holes in the low-pressure side of the inlet allow leakage past the ring seals to flow into the suction side of the inducer, thus maintaining a low pressure.

The oxidizer volute is secured to the fuel volute with pins and bolts which prevent rotational and axial movement. The primary oxidizer seal and spacer located in the oxidizer volute prevent fuel from leaking into the primary oxidizer seal drain cavity. The oxidizer intermediate seal directs a purge flow into the primary seal and No. 3 drain cavities where the purge acts as a barrier to permit positive separation of the oxidizer and bearing lubricants.

Fuel Pump

The fuel pump supplies fuel to the thrust chamber and gas generator at a flowrate of 15,471 gpm. The pump consists of an inlet, an inducer, an impeller, a volute, bearings, seals, and spacers. Fuel is introduced into the pump from the vehicle fuel tank through the inlet. The inducer in the inlet increases the pressure of the fuel as it passes into the impeller to prevent cavitation. The impeller accelerates the fuel to the desired pressure and discharges it through two diametrically opposed outlets into the high-pressure fuel lines leading to the thrust chamber and gas generator.

The fuel volute is bolted to the inlet and to a ring, which is pinned to the oxidizer volute. A wear-ring installed on the volute mates against the impeller. The cavity formed between the volute and the impeller is called the balance cavity. Pressure in the balance cavity exerts a downward force against the fuel impeller and counterbalances the upward force of the oxidizer impeller to control the amount of shaft axial force applied to the No. 1 and No. 2 bearings. Leakage between the impeller inlet and the discharge is controlled by a wear-ring, which mates with the impeller and acts as an orifice. The fuel volute provides support for the bearing retainer, which supports the No. 1 and No. 2 bearings and houses the bearing heater. The No. 3 seal, which is installed between the oxidizer intermediate seal and the No. 1 bearing, prevents lubricating fuel for the bearings from contacting the oxidizer. If fuel should pass the seal, purge flow from the oxidizer intermediate seal will expel the fuel overboard. On the fuel side of the No. 2 bearing, the No. 4 lube seal contains the lubricant within the bearing cavity. The remaining seal in the fuel volute is the primary seal and contains fuel under pressure in the balance cavity, maintains the desired balance cavity pressure, and keeps high-pressure fuel out of the low-pressure side.

Turbine

The turbine, producing 55,000 brake horsepower, drives the fuel and oxidizer pumps. It is a two-stage, velocity-compounded turbine consisting of two rotating impulse wheels separated by a set of stators. The turbine mounts on the fuel pump end of the turbopump so that the two elements of the turbopump having the greatest operating temperature extremes (1500 Fahrenheit for the turbine and -300 Fahrenheit for the oxidizer pump) are separated.

Hot gas from the gas generator enters the turbine at a flowrate of 170 pounds per second through the inlet manifold and is directed through the first-stage nozzle onto the 119-blade first-stage wheel. The hot gas then passes through the second-stage stators onto the 107-blade second-stage wheel, and then into the heat exchanger. This flow of hot gas rotates the turbine, which in turn rotates the propellant pumps. Turbine speed during mainstage operation is 5,550 rpm.

Bearing Coolant Control Valve

This valve, which incorporates three 40-micron filters, three spring-loaded poppets, and a restrictor, performs two functions. Its primary function is to control the supply of coolant fuel to the turbopump bearings. Its secondary function is to provide a means of preserving the turbopump bearings between static firings or during engine storage. During engine firing, the coolant poppet opens and delivers filtered fuel to the turbopump bearing coolant jets, and the restrictor provides the proper turbopump bearing jet pressure.

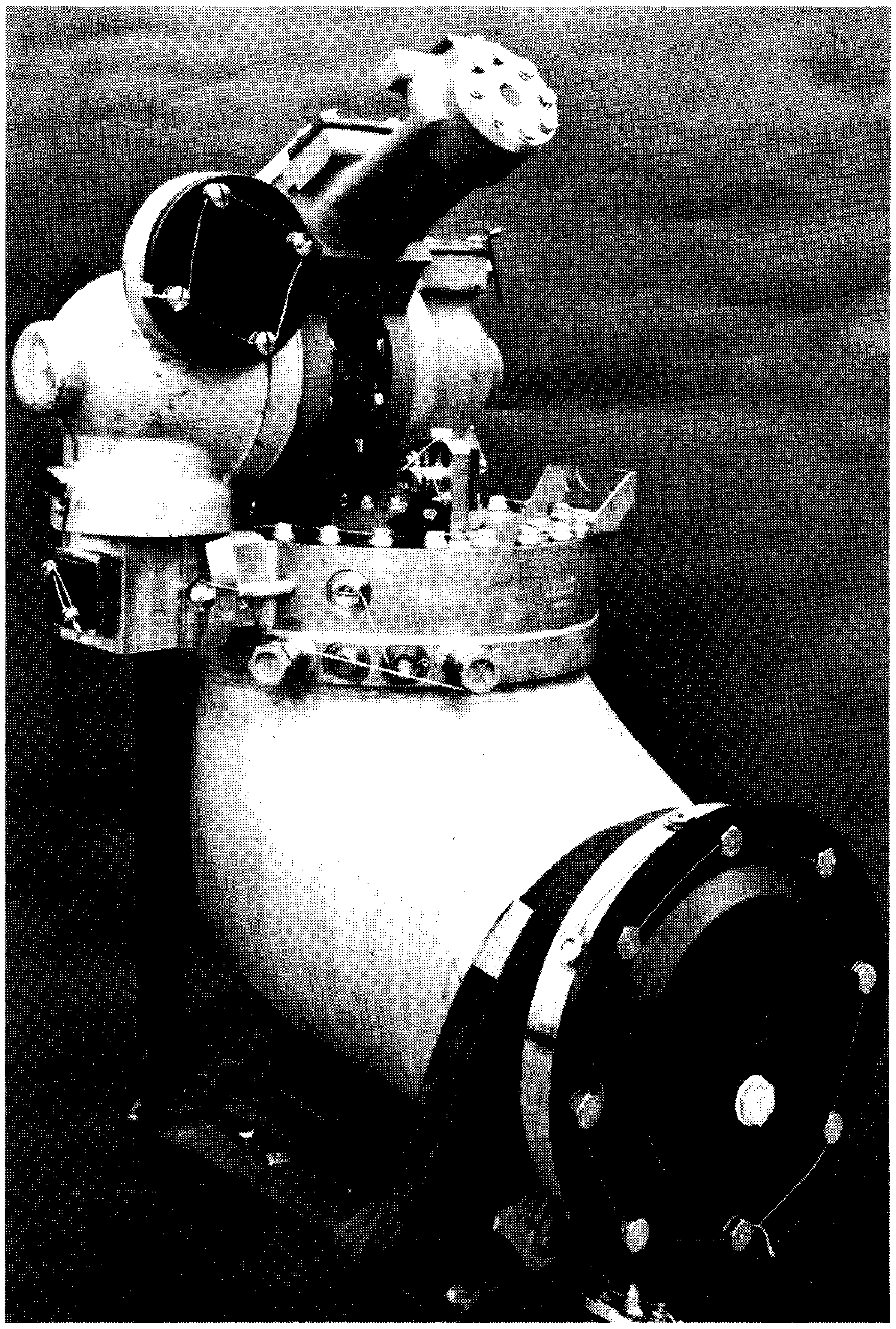
GAS GENERATOR SYSTEM

The gas generator system provides the hot gases for driving the velocity-compounded turbine, which drives the fuel and oxidizer pumps. The system consists of a gas generator valve, an injector, a combustion chamber, and propellant feed lines connecting the No. 2 turbopump fuel and oxidizer outlet lines to the gas generator. The propellants are supplied to the gas generator from the No. 2 turbopump fuel and oxidizer outlet lines. The gas generator mixture ratio, relative to the engine mixture ratio, is fuel-rich. This provides a lower combustion temperature in the uncooled gas generator and in the turbine.

Propellants enter the gas generator through the valve and injector and are ignited in the combustion chamber by dual pyrotechnic igniters. The gas generator valve is hydraulically operated by fuel pressure from the hydraulic control system.

Gas Generator Valve

The gas generator valve is a hydraulically operated valve which controls and sequences entry of propellants into the gas generator. Hydraulic fuel is recirculated through a passage in the valve housing to maintain seal integrity and to prevent the fuel in the fuel ball housing from freezing. Fuel is also recirculated through a passage in the piston between the opening port and the closing port to prevent the piston O-ring from freezing.



R-5

Gas Generator Assembly Including Control Valves

Gas Generator Injector

The gas generator injector directs fuel and oxidizer into the gas generator combustion chamber. It is a flat-faced, multi-orificed injector incorporating a dome, a plate, a ring manifold, five oxidizer rings, five fuel rings, and a fuel disc. The gas generator valve and the gas generator injector fuel inlet housing tee are mounted on the injector.

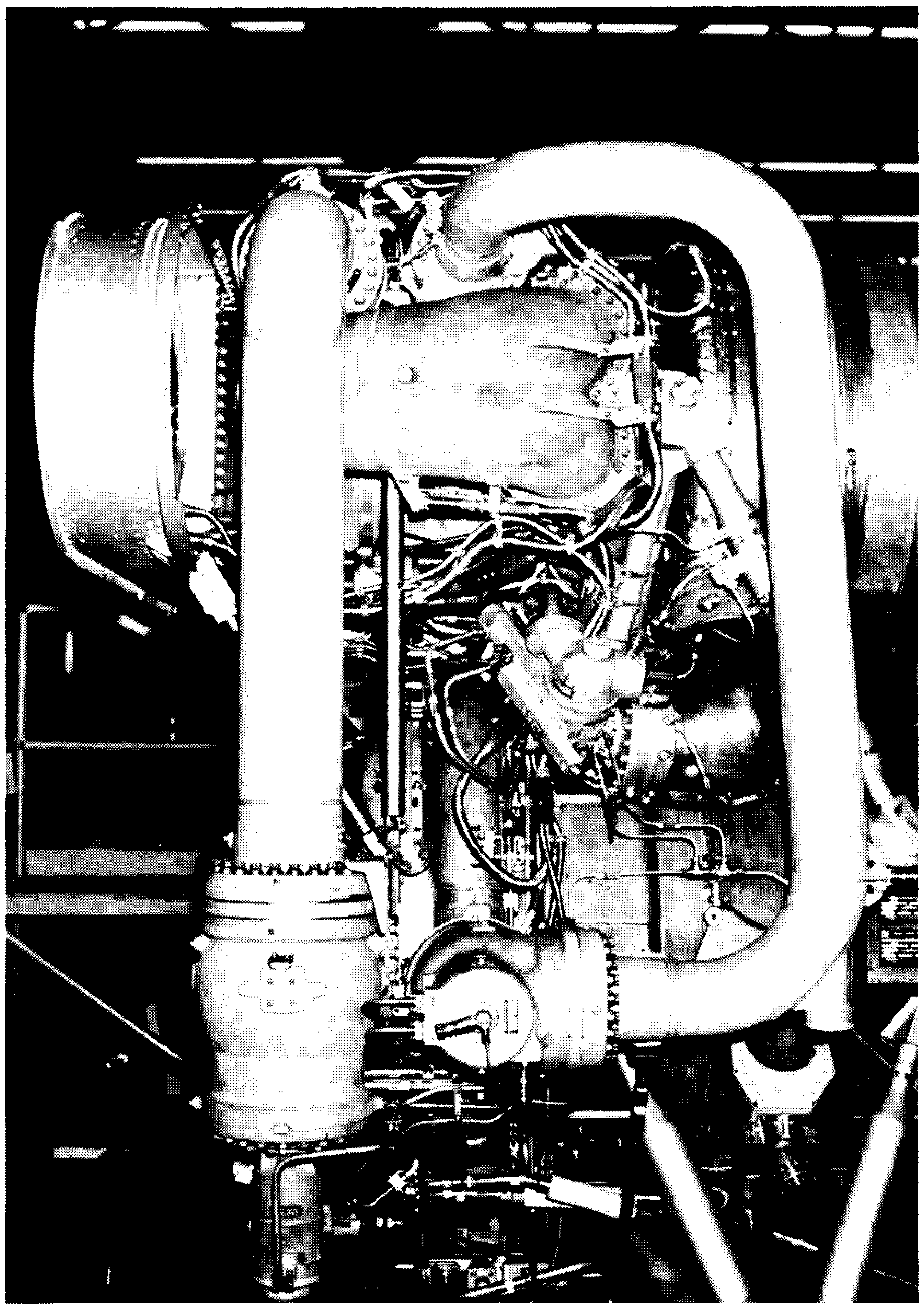
Fuel enters the injector through the gas generator fuel inlet housing tee from the gas generator valve. The fuel is directed through internal passages in the plate and injected into the combustion chamber through orifices in the fuel rings and the disc. Some of the orifices in the outer fuel ring also provide a cooling film of fuel for the combustion chamber wall. Oxidizer enters the injector through the oxidizer inlet manifold from the gas generator valve. The oxidizer is directed from the oxidizer manifold through internal passages in the plate and is injected into the combustion chamber through the orifices in the oxidizer rings.

Gas Generator Combustion Chamber

The gas generator combustion chamber provides a space for burning propellants and exhausts the gases from the burning propellants into the turbopump turbine manifold. It is a single-wall chamber located between the gas generator injector and the turbopump inlet.

PROPELLANT FEED CONTROL SYSTEM

The propellant feed system transfers LOX and fuel from the propellant tanks into the pumps which discharge into the high-pressure ducts leading to the gas generator and the thrust chamber. The system consists of two oxidizer valves, two fuel valves, a bearing coolant control valve, two oxidizer dome purge check valves, a gas generator and pump seal purge check valve, turbopump outlet lines, orifices, and lines connecting the components. High-pressure fuel is supplied from the propellant feed system of the engine to the vehicle-contractor-supplied thrust vector control system.

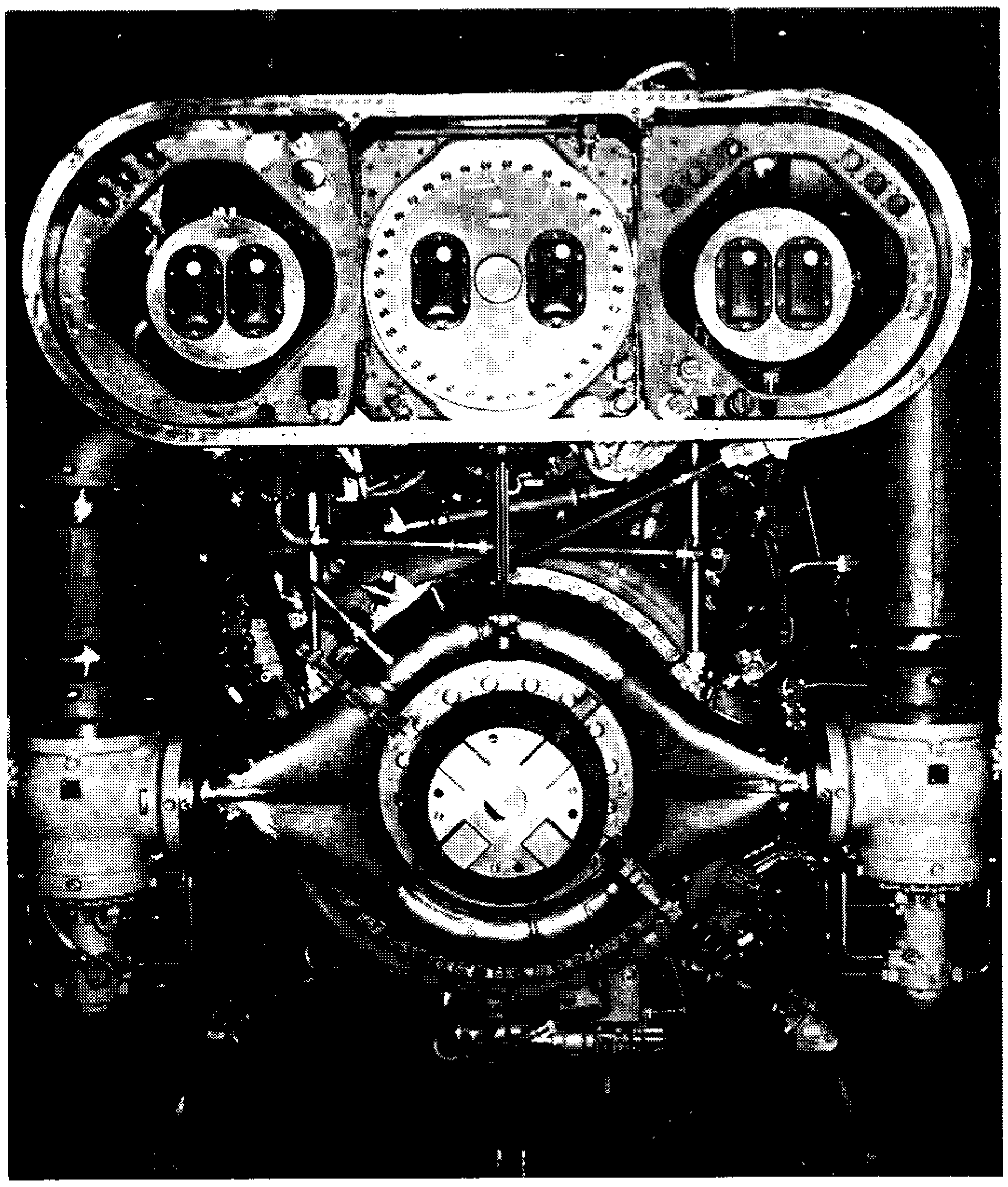


R-3A

Propellant Feed—The main LOX valve and high-pressure line are shown at left. At right are the main fuel valve and high-pressure line.

Oxidizer Valves

Two identical oxidizer valves, designated No. 1 and No. 2, control LOX flow from the turbopump to the thrust chamber oxidizer dome and sequence the hydraulic fuel to the opening port of the gas generator valve. Each of the oxidizer valves is a hydraulically actuated, pressure-balanced, poppet type, and contains a mechanically actuated sequence valve. A spring-loaded gate valve permits reverse flow for recirculation of the hydraulic fluid with the propellant valves in the closed position, but prevents fuel from passing through until the oxidizer valve is open 16.4 per cent. As the oxidizer valve reaches this position, the piston shaft opens the gate, allowing fuel to flow through the sequence valve, which in turn opens the gas generator valve.



R-6

LOX Distribution—Oxidizer is distributed by the LOX dome (lower center). Main LOX valves are shown at left and right with the engine interface panel above.

A position indicator provides relay logic in the engine electrical control circuit and provides instrumentation for recording movement of the oxidizer valve poppet.

The two oxidizer dome purge check valves, mounted on each of the oxidizer valves, allow purge gas to enter the oxidizer valves, but prevent oxidizer from entering the purge system.

Fuel Valves

Two identical fuel valves, designated No. 1 and No. 2, are mounted 180 degrees apart on the thrust chamber fuel inlet manifold and control the flow of fuel from the turbopump to the thrust chamber. When the valves are in the open position at rated engine pressures and flowrates, they will not close if hydraulic fuel pressure is lost.

Position indicators in the fuel valves provide relay logic in the engine electrical control circuit and instrumentation for recording movement of the valve poppets.

Thrust-OK Pressure Switches

Three pressure switches, mounted on a single manifold located on the thrust chamber fuel manifold, sense fuel injection pressure. These thrust-OK pressure switches are used redundantly in the vehicle to indicate that all five engines are operating satisfactorily. If pressure in the fuel injection cavity decreases, the switches deactuate, breaking the contact and interrupting the thrust-OK output signal.

PRESSURIZATION SYSTEM

The pressurization system heats GOX and helium for vehicle tank pressurization. The pressurization system consists of a heat exchanger, a heat exchanger check valve, a LOX flowmeter, and various heat exchanger lines. The LOX source for the heat exchanger is tapped from the thrust chamber oxidizer dome, and the helium is supplied from the vehicle. LOX flows from the thrust chamber oxidizer dome through the heat exchanger check valve, LOX flowmeter, and the LOX line to the heat exchanger.

Heat Exchanger

The heat exchanger heats GOX and helium with hot turbine exhaust gases, which pass through the heat exchanger over the coils. The heat exchanger consists of four oxidizer coils and two helium coils installed within the turbine exhaust duct. The heat exchanger is installed between the turbopump manifold outlet and the thrust chamber exhaust manifold inlet. The shell of the heat exchanger contains a bellows assembly to compensate for thermal expansion during engine operation.

Heat Exchanger Check Valve

The heat exchanger check valve prevents GOX or vehicle prepressurizing gases from flowing into the oxidizer dome. It consists of a line assembly and a swing check valve assembly. It is installed between the thrust chamber oxidizer dome and the heat exchanger LOX inlet line.

LOX Flowmeter

The LOX flowmeter is a turbine-type, volumetric, liquid-flow transducer incorporating two pickup coils. Rotation of the LOX flowmeter turbine generates an alternating voltage at the output terminals of the pickup coils.

Heat Exchanger Lines

LOX and helium are routed to and from the heat exchanger through flexible lines. The GOX and helium lines terminate at the vehicle connect interface. The LOX line connects the heat exchanger to the heat exchanger check valve.

ENGINE INTERFACE PANEL

The engine interface panel, mounted above the turbopump LOX and fuel inlets, provides the vehicle connect location for electrical connectors between the engine and the vehicle. It also provides the attachment point for the vehicle flexible heat-resistant curtain. The panel is fabricated from heat-resistant stainless-steel casting made in three sections and assembled by rivets and bolts.

ELECTRICAL SYSTEM

The electrical system consists of flexible armored wiring harnesses for actuation of engine controls and the flight instrumentation harnesses.

HYDRAULIC CONTROL SYSTEM

The hydraulic control system operates the engine propellant valves during the start and cutoff sequences. It consists of a hypergol manifold, a checkout valve, an engine control valve, and the related tubing and fittings.

Hypergol Manifold

The hypergol manifold directs hypergolic fluid to the separate igniter fuel system in the thrust chamber injector. It consists of a hypergol container, an ignition monitor valve, a position switch, and an igniter fuel valve. The hypergol container, position switch, and igniter fuel valve are internal parts of the hypergol manifold.

A spring-loaded, cam-lock mechanism incorporated in the hypergol manifold prevents actuation of the ignition monitor valve until after the upstream hypergol cartridge diaphragm bursts. The same mechanism actuates a position switch that indicates when the hypergol cartridge is installed. The igniter fuel valve is a spring-loaded, cracking check valve that opens and allows fuel to flow into the hypergol container. The hypergol cartridge diaphragms are ruptured by the resultant pressure surge when the igniter fuel valve opens.

Ignition Monitor Valve

The ignition monitor valve is a pressure-actuated, three-way valve mounted on the hypergol manifold. It controls the opening of the fuel valves and permits them to open only after satisfactory combustion has been achieved in the thrust chamber.

When the hypergol cartridge is installed in the hypergol manifold, a cam-lock mechanism prevents the ignition monitor valve poppet from moving from the closed position. The ignition monitor valve has six ports: a control port, an inlet port, two outlet ports, a return port, and an atmospheric reference port. The control port receives pressure from the thrust chamber fuel manifold. The inlet port receives hydraulic fuel pressure for opening the fuel valves. When the ignition monitor valve poppet is in the deactuated position, hydraulic fuel from the inlet port is stopped at the poppet seat. When the hypergol cartridge diaphragm bursts, the spring-loaded cam-lock retracts to permit the ignition monitor valve poppet unrestricted motion. When thrust chamber pressure (directed to the control port from the thrust chamber fuel manifold) increases, the ignition monitor valve poppet moves to the open (actuated) position and hydraulic fuel is directed through the outlet ports to the fuel valves.

Checkout Valve

The checkout valve consists of a ball, a poppet, and an actuator. The checkout valve provides for ground checkout of the ignition monitor valve and fuel valves and prevents the ground hydraulic return fuel, used during checkout, from entering the engine system and consequently the vehicle fuel tank.

When performing the engine checkout or servicing, the checkout valve ball is positioned so fuel entering the engine hydraulic return inlet port will be directed through the ball and out the GSE return port. For engine static firing or flight, the ball is positioned so fuel entering the engine hydraulic return inlet port will be directed through the ball and out the engine return outlet port.

Engine Control Valve (Hydraulic Filter and Four-Way Solenoid Valve Manifold)

The engine control valve incorporates a filter manifold, a four-way solenoid valve, and two swing check valves.

The filter manifold contains three filters. One filter is in the supply system and one each in the opening and closing pressure systems. The filters prevent entry of foreign matter into the four-way solenoid valve or the engine. Two swing check valves are "teed" into the supply system filter. The check valves permit hydraulic system operation from the ground supplied hydraulic fluid for checkout and servicing procedures or engine supplied hydraulic fluid for normal engine operation.

The four-way solenoid valve is comprised of a main spool and sleeves to achieve two-directional control of the fluid flow to the main fuel, main oxidizer, and gas generator valve actuators. The spool is pressure-positioned by two three-way slave pilots. Each slave pilot has a solenoid-controlled, normally open, three-way primary pilot.

The de-energized position of the engine control valve provides hydraulic closing pressure to all engine propellant valves. Momentary application of 28 VDC to the start solenoid will initiate control valve actuations that culminate in the positioning of the main spool so that hydraulic pressure is applied to the opening port, and the pressure previously applied to the closing port is vented to the return port.

An internal passage in the housing maintains common pressure applied between the opening port and start solenoid poppet. This pressure, after start solenoid de-energization, holds the main spool in its actuated position thereby maintaining the pressure directed to the opening port without further application of the start solenoid electrical signal. Momentary application of 28 VDC to the stop solenoid will initiate control valve actuations that culminate in positioning the main spool so that pressure is vented from the opening port and applied to the closing port. The override piston may be actuated at any time by a remote pressure supply, which, in the event of an electrical power loss, would reposition the main spool and apply hydraulic pressure to the closing port. If electrical power and hydraulic power are both removed, the valve will return to the de-energized position by spring force. If hydraulic pressure is then reapplied, pressure will be applied to the closing port. If an electrical signal is simultaneously sent to the start and stop solenoids, the stop solenoid will override the start and return the valve to a deactuated position.

Swing Check Valve

There are two identical swing check valves installed on the engine control valve. They allow the use of ground hydraulic fuel pressure during engine starting transient and engine hydraulic fuel pressure during engine mainstage and shutdown. One check valve is installed in the engine hydraulic fuel supply inlet port, the other in the ground hydraulic fuel supply inlet port.

FLIGHT INSTRUMENTATION SYSTEM

The flight instrumentation system consists of pressure transducers, temperature transducers, position indicators, a flow measuring device, power distribution junction boxes, and associated electrical harnesses, and permits monitoring of engine performance. The basic flight instrumentation system is composed of a primary and an auxiliary system. The primary instrumentation system is critical to all engine static firings and subsequent vehicle launches; the auxiliary system is used during research, development, and acceptance portions of the engine static test program and initial vehicle flights. The flight instrumentation system components, including both the primary and auxiliary systems, are listed below:

Primary Instrumentation

Fuel turbopump inlet No. 1 pressure
Fuel turbopump inlet No. 2 pressure
Common hydraulic return pressure
Oxidizer turbopump bearing jet pressure
Combustion chamber pressure
Gas generator chamber pressure
Oxidizer turbopump discharge No. 2 pressure
Fuel turbopump discharge No. 2 pressure
Oxidizer pump bearing No. 1 temperature
Oxidizer pump bearing No. 2 temperature
Turbopump bearing temperature
Turbopump inlet temperature
Turbopump speed

Auxiliary Instrumentation

Oxidizer turbopump seal cavity pressure
Turbine outlet pressure
Heat exchanger helium inlet pressure
Heat exchanger outlet pressure
Oxidizer turbopump discharge No. 1 pressure
Heat exchanger LOX inlet pressure
Heat exchanger GOX outlet pressure
Fuel turbopump discharge No. 1 pressure
Engine control opening pressure
Engine control closing pressure

Heat exchanger LOX inlet temperature
Heat exchanger GOX outlet temperature
Heat exchanger helium outlet temperature
Fuel pump inlet No. 2 temperature
Heat exchanger LOX inlet flowrate

Primary and Auxiliary Junction Box

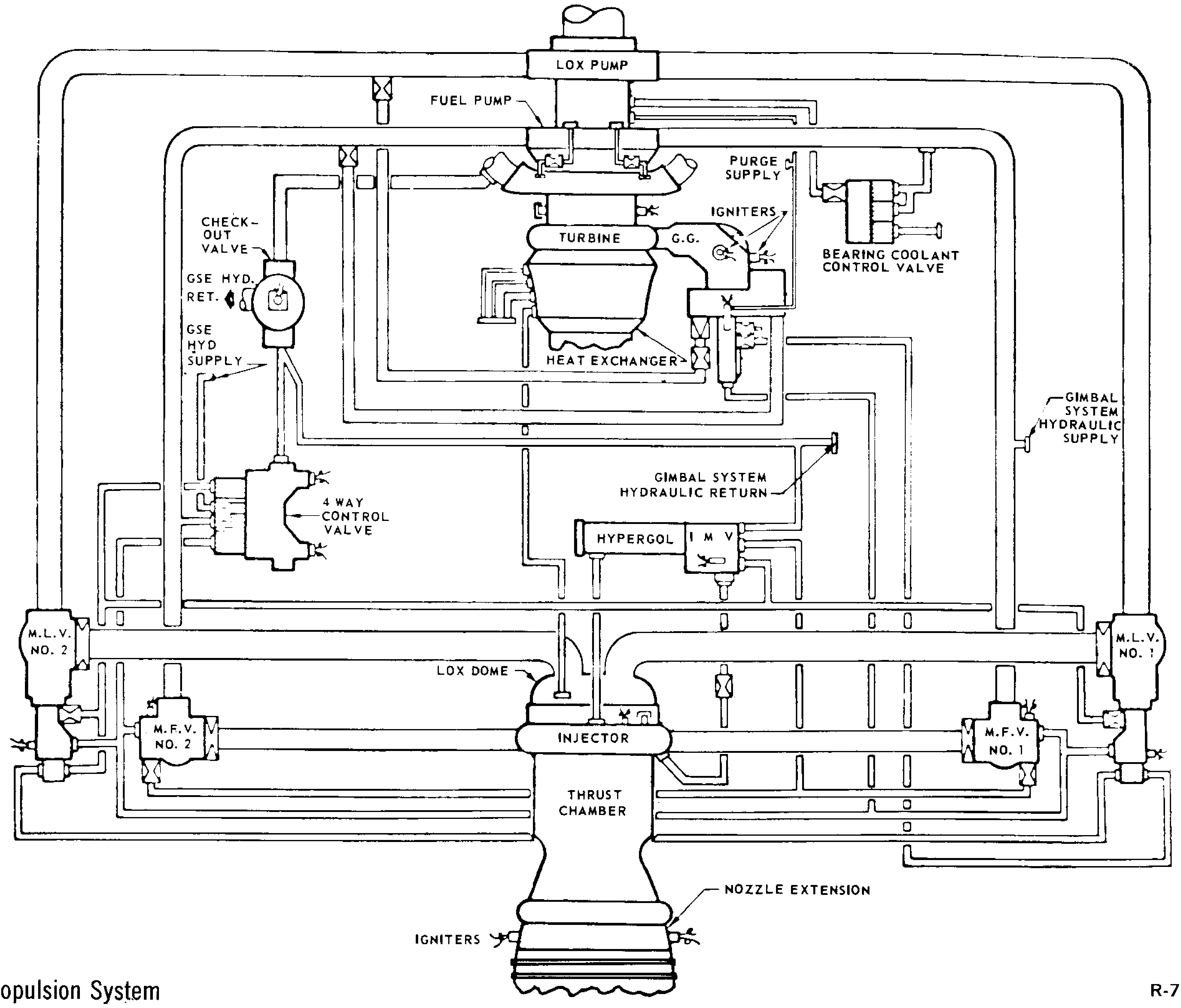
There are two electrical junction boxes in the flight instrumentation system. The primary junction box has provisions for eight electrical connectors, and the auxiliary junction box for five. Both junction boxes are welded closed and pressurized with an inert gas to prevent possible entry of contaminants and moisture.

ENGINE OPERATION

The engine requires a source of pneumatic pressure, electrical power, and propellants for sustained engine operation. A ground hydraulic pressure source, thrust chamber prefill, gas generator and turbine exhaust igniters, and hypergolic fluid are required to start the engine.

When the start button is actuated, the checkout valve moves to transfer the hydraulic fuel return from the ground line to the turbopump low-pressure fuel inlet. The high-level oxidizer purge is initiated to the gas generator and thrust chamber LOX dome.

The gas generator and turbine exhaust gas igniters fire, and the engine control valve start solenoid is energized. Hydraulic pressure is directed to the opening port of the oxidizer valves. The oxidizer valves are part way open, and the hydraulic pressure is directed to the gas generator valve opening port. The gas generator valve opens, propellants under tank pressure enter the gas generator combustion chamber, and the propellant mixture is ignited by the gas generator igniters. The exhaust gas is ducted through the turbopump turbine, the heat exchanger, and the thrust chamber exhaust manifold into the nozzle extension walls where the fuel-rich mixture is ignited by the turbine exhaust gas igniters. As the turbine accelerates the fuel and the oxidizer pumps, the pump discharge pressures increase and propellants at increasing flowrates are supplied to the gas generator. Turbopump acceler-



F-1 Propulsion System

R-7A

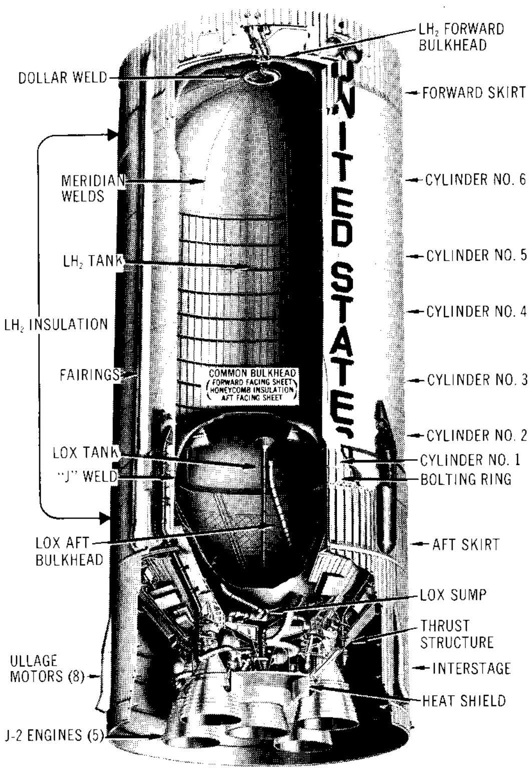
ation continues and, as the fuel pressure increases, the igniter fuel valve opens and allows fuel pressure to build up against the hypergol cartridge burst diaphragm. The hypergol diaphragms burst under the increasing fuel pressure. Hypergolic fluid, followed by the ignition fuel, enters the thrust chamber. When hypergolic fluid enters the thrust chamber and contacts the oxidizer, spontaneous combustion occurs, establishing thrust chamber ignition. Thrust chamber pressure is transmitted through the sense line to the diaphragm of the ignition monitor valve. When the thrust chamber pressure increases, the ignition monitor valve actuates and allows hydraulic fluid flow to the opening port of the fuel valves. The fuel valves open and fuel is admitted to the thrust chamber.

Fuel enters the thrust chamber fuel inlet manifold and passes through the thrust chamber tubes for cooling purposes and then through the injector into the thrust chamber combustion zone. As the thrust chamber pressure increases, the thrust-OK pressure switches are actuated indicating the engine is operating satisfactorily. The thrust chamber pressure continues to increase until the gas generator reaches rated power, controlled by orifices in the propellant lines feeding the gas generator. When engine fuel pressure increases above the ground-supplied hydraulic pressure, the hydraulic pressure supply source is transferred to the engine. Hydraulic fuel is circulated through the engine components and then returned through the engine control valve and checkout valve into the turbopump fuel inlet. The ground hydraulic source facility shutoff valve is actuated to the closed position when the fuel valves open. This allows the engine hydraulic system to supply the hydraulic pressure during the cutoff sequence.

ENGINE CUTOFF

When the cutoff signal is initiated, the LOX dome operational oxidizer purge comes on, and the engine control valve stop solenoid is energized. Hydraulic pressure holding open the gas generator valves, the oxidizer valves, and the fuel valves is routed to return. Simultaneously, hydraulic pressure is directed to the closing ports of the gas generator valve, the oxidizer valves, and the fuel valves. The checkout valve is actuated and, as propellant pressures decay, the high level oxidizer purge begins to flow; then the igniter fuel valve and the ignition monitor valve close. Thrust chamber pressure will reach the zero level at about the same time the oxidizer valves reach full-closed.

SECOND STAGE FACT SHEET



WEIGHT: 95,000 lb. (dry)
1,037,000 lb. (loaded)
DIAMETER: 33 ft.
HEIGHT: 81 ft. 7 in.
BURN TIME: 6 min. approx. (actually 395 sec.)
VELOCITY: 15,300 miles per hour at burnout (approx.)
ALTITUDE AT BURNOUT: 114.5 miles

MAJOR STRUCTURAL COMPONENTS

AFT INTERSTAGE
AFT SKIRT
THRUST STRUCTURE
AFT LOX BULKHEAD
COMMON BULKHEAD
LH$_2$ CYLINDER WALLS
LH$_2$ FORWARD BULKHEAD
FORWARD SKIRT

MAJOR SYSTEMS

PROPULSION: Five J-2 engines
Thrust: More than 1,000,000 lb. (225,000 maximum each engine)
Propellant: LH2—260,000 gal. (153,000 lb.)
LOX—83,000 gal. (789,000 lb.)

ELECTRICAL: 6 electrical bus systems, four 28-volt DC flight batteries, and motor-operated power transfer switches

ORDNANCE: Provides, in operational sequence, ignition of eight ullage motors before ignition of five main engines, explosive separation of second stage interstage skirt, explosive separation of second stage from third stage, and ignition of four retrorockets to decelerate second stage for complete separation

MEASUREMENT: Instrumentation, telemetry, and radio frequency subsystems

THERMAL CONTROL: A ground-operated system that provides proper temperature control for equipment containers in the forward and aft skirt

FLIGHT CONTROL: Gimbaling of the four outboard J-2 engines as required for thrust vector control, accomplished by hydraulic-powered actuators which are electrically controlled from signals initiated in the flight control computer of the instrument unit (atop the Saturn V third stage)

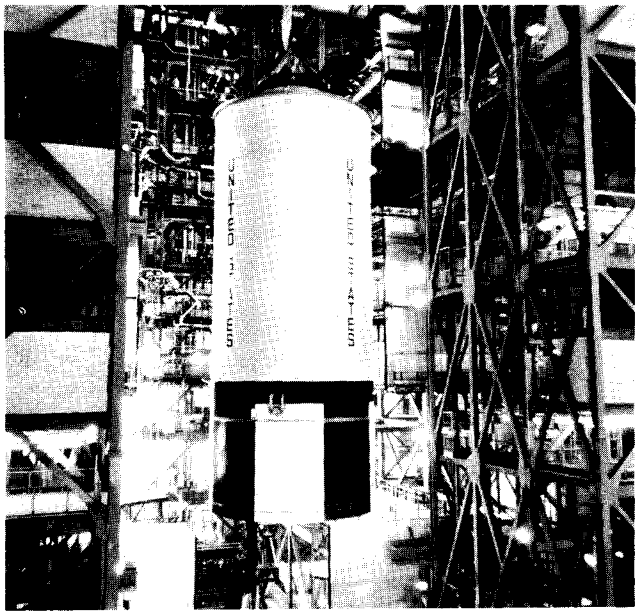
SECOND STAGE

SECOND STAGE DESCRIPTION

The second stage of the Saturn V is the most powerful hydrogen-fueled launch vehicle under production. Manufactured and assembled by North American Aviation's Space Division, it employs the cryogenic (ultra-low temperature) propellants of liquid hydrogen and liquid oxygen, which must be contained at temperatures of -423 and -297 degrees Fahrenheit, respectively.

For the lunar mission, the second stage takes over from the Saturn V's first stage at an altitude of approximately 200,000 feet (38 miles) and boosts its payload of the third stage and Apollo spacecraft to approximately 606,000 feet (114.5 miles). When its five J-2 engines ignite, the stage is pushing more than one million pounds, a load greater than that of any U.S. booster prior to the Saturn program. Speed of the stage ranges from 6,000 miles per hour to 15,300 miles per hour.

The beginning of second stage boost is a two-step process. When all the F-1 engines of the first stage have cut off, the first stage separates. Eight ullage rocket motors located around the bottom of the second stage then fire for approximately 4 seconds to give positive acceleration to the stage prior to ignition of the five J-2 engines. About 30 seconds after the first stage separation, the part of the second stage structure on which the ullage rockets are located (the aft interstage) is separated by firing explosive charges. This second separation is a precise maneuver: the 18-foot-high interstage must slip past the engines without touching them. With the stage traveling at great speed, the interstage must clear the engines by only a little more than 3 feet.



S-2

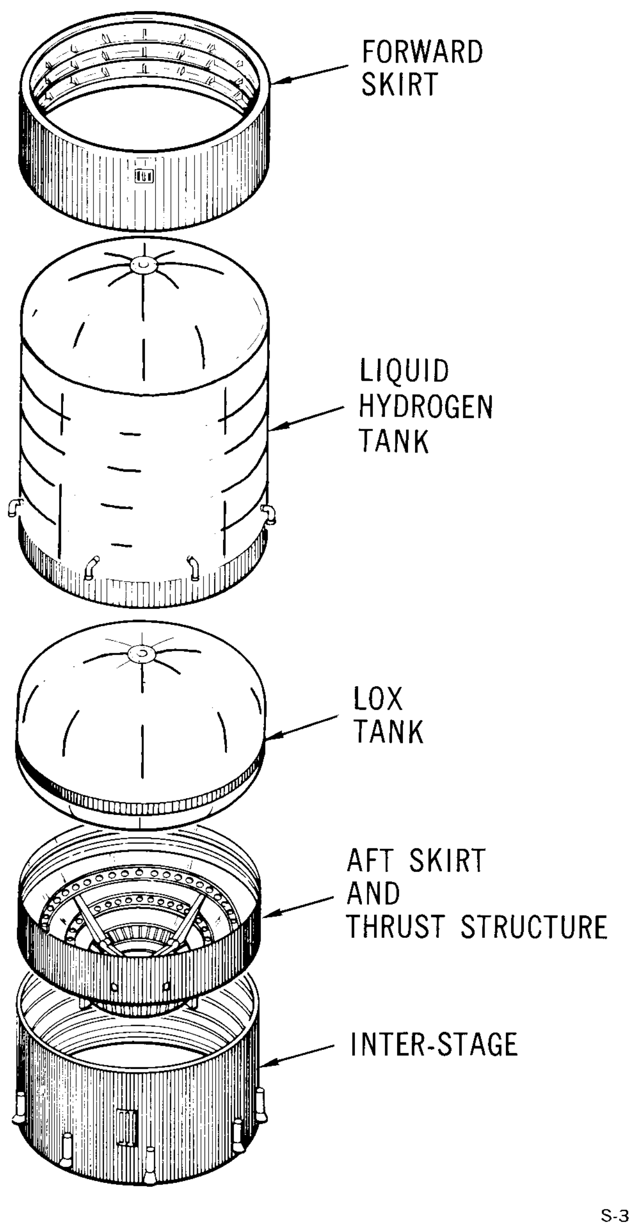
Mating—A completed second stage is mated to a first stage at Kennedy Space Center, Fla. This particular stage was used for facilities checkout.

The second stage burns for about 6 minutes, pushing its payload into space. At the end of boost, all J-2 engines cut off at once, the stages separate, and the J-2 engine on the third stage begins firing to take it and the Apollo spacecraft into a parking earth orbit. The 81-foot 7-inch second stage is basically a container for its 942,000 pounds of propellant with engines attached at the bottom. Propellants represent more than 90 per cent of the stage's total weight. Despite this great weight of propellant and the stresses the stage must take during launch and boost, the stage is primarily without an internal framework. It is constructed mostly of lightweight aluminum alloys ribbed in such a fashion that it is rigid enough to withstand the pressures to which it is subjected. Special lightweight insulation had to be developed to keep its cryogenic propellants from warming and thus turning to gas and becoming totally useless as propellant. The insulation that helps maintain a difference of about 500 degrees between outside (70 to 80-degree normal Florida temperature) and inside (-423° F of liquid hydrogen) is only about 1-1/2 inches thick around the hydrogen tank.

A unique feature of the second stage is its common bulkhead, a single structure which is both the top of the liquid oxygen tank and the bottom of the liquid hydrogen tank. This bulkhead was a critical item in the development of the stage. The relatively thin bulkhead, consisting of two aluminum facing sheets separated by a phenolic honeycomb core insulation, must maintain a temperature difference of 126 degrees between the two sides. The insulation which accomplishes this varies from one-tenth of an inch thickness at the girth to 4-3/4 inches thickness at the apex of the bulkhead. Development of the common bulkhead resulted in a weight saving of approximately 4 tons and more than 10 feet in stage length.

STRUCTURE

The second stage structure consists of an interstage, which links it with the first stage; a thrust structure and aft skirt assembly, which supports and houses the five J-2 engines; an ellipsoidal liquid oxygen tank; a bolting ring, which attaches the liquid oxygen tank to the second stage structure; six aluminum cylinder walls, which are welded

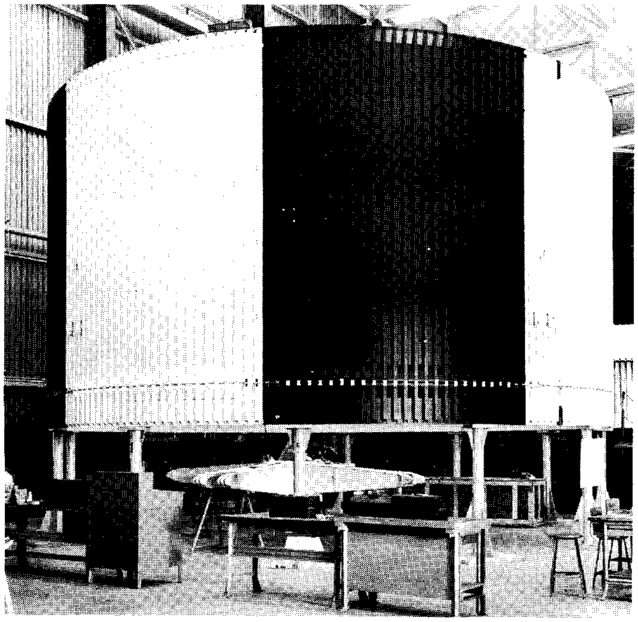


Second Stage Subassemblies

together to form the liquid hydrogen tank; a forward domed bulkhead; and the forward skirt, which connects with the Saturn V third stage. Another important part of the structure is the 60-foot systems tunnel located on the outside of the liquid hydrogen cylinder walls through which all electrical wires between the aft skirt and the forward skirt are routed.

Interstage

The interstage, fabricated at NAA's Tulsa plant, is a semimonocoque structure. Semimonocoque means that the skin has a minimum of internal framework. The interstage is slightly over 18 feet in height and 33 feet in diameter. The structure has internal circumferential supporting frames and external hat



Interstage

sections positioned vertically to provide structural rigidity.

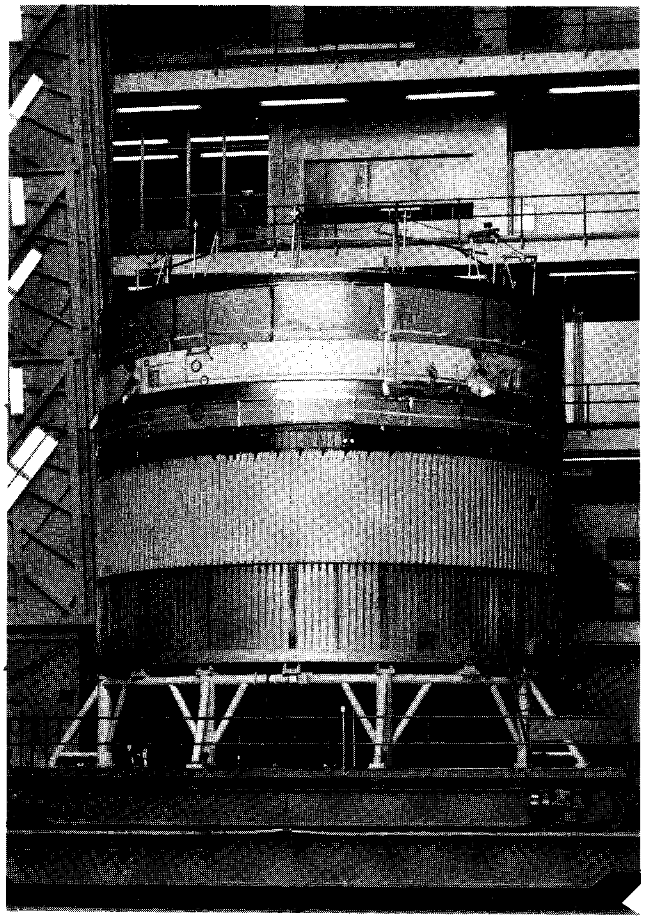
After first stage burnout and initial separation, eight rocket motors attached equidistantly around the interstage are fired for approximately 4 seconds. These motors, called ullage motors (an old brewer's term referring to the gaseous zone in a tank above the liquid), provide positive acceleration and therefore pressure to force the stage's propellants into the feed lines to the J-2 engines. This is. called the ullage maneuver. The interstage is separated from the second stage approximately 30 seconds after it separates from the first stage. The two-step separation of the interstage is called dual-plane separation.

Aft Skirt

Like the interstage, the aft skirt (as well as the thrust structure and forward skirt) is manufactured at NAA's Tulsa facility and delivered to Seal Beach for final assembly. The aft skirt is 7 feet in height and is semimonocoque construction fabricated from aluminum alloy. It is fabricated in four panels with external stringers and subassembled internal frames.

Thrust Structure

The thrust structure consists of four panels of riveted skin-stringer and internal frame construction, which, when assembled, forms an inverted cone decreasing in size from the 33-foot diameter of the upper ring to approximately 18 feet at the



S-5

Stacking Stage—Aft skirt, thrust structure, and common bulkhead move on transfer table to new station for further buildup of stage.

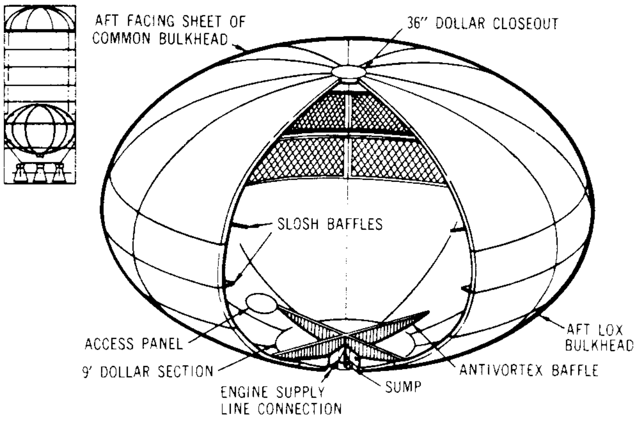
lower ring. Four support rings along with an outer skin stiffened with hat sections comprise the basic structure. In addition, eight thrust longerons (two to each panel) extend upward along the conical surface of the thrust structure. The lower circumferential ring rests directly over the line of thrust of each of the four outboard engines while the center engine support beam assembly is directly over the thrust line of the center engine. A rigid heat shield mounted around the five J-2 engines to a frame connecting to the thrust structure protects the base area of the stage against recirculation of hot engine exhaust gases and heat from the exhaust. This heat shield is of lightweight construction protected by low-density ablative (heat-resistant) material.

Although assembled separately, the aft skirt and thrust structure when joined become a structural entity and together support the five engines and withstand and distribute the thrust and boost structural loads.

In addition to engines and engine accessories, the interstage, aft skirt, and thrust structure house electrical and mechanical equipment such as signal conditioners and controllers, telemetry electronics, flight control electronics, service and connecting umbilicals, electrical power control units, power distribution panels and batteries, inverters, propellant management electronics, propellant plumbing, ordnance installations, and hydraulic pumps and accumulators. Equipment that is not required after second-plane separation is in the interstage which is separated 30 seconds after ignition. Equipment necessary for flight operations is located on the aft skirt, thrust structure, and forward skirt.

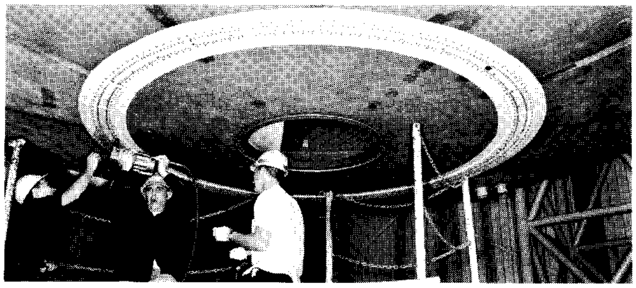
Liquid Oxygen Tank

The liquid oxygen (LOX) tank is an ellipsoidal container 22 feet high and fabricated from ellipsoidal-shaped top and aft halves. The top half of the LOX tank is known as the common bulkhead and is actually two bulkheads separated by phenolic honeycomb insulation and bonded together to form both the upper portion of the liquid oxygen tank and the lower portion of the liquid hydrogen tank.



Second Stage LOX Tank

All of the LOX tank bulkheads are formed by welding together 12 high-energy-formed curved sections (gores), each approximately 20 feet long and 8 feet



S-11

Tank Fabrication—Workmen close out dollar section of propellant tank.

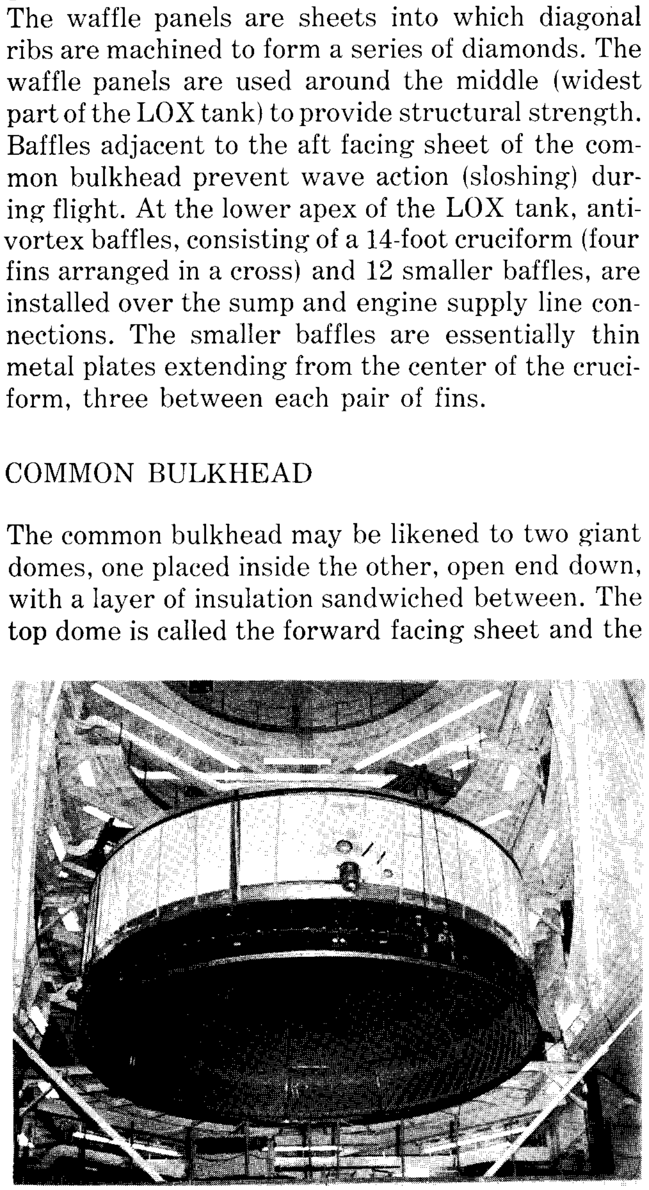
wide. When the gores are welded together, an opening is formed at the apex of the bulkhead. The apex is closed by welding the 12 gores to a circular section called a dollar section.

AFT LOX BULKHEAD

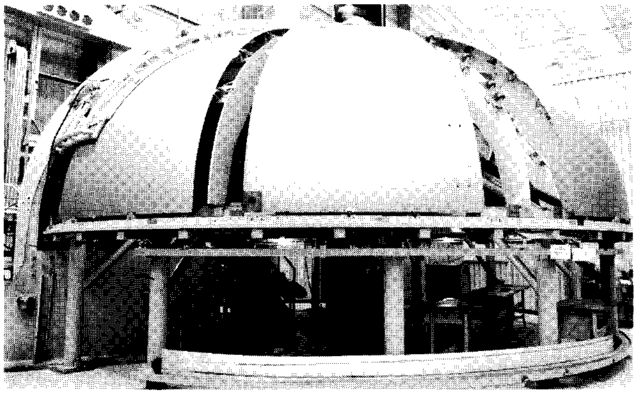
The aft LOX bulkhead, like the aft facing sheet of the common bulkhead, is composed of 12 thin aluminum gores welded to mechanically milled waffle panels. The waffle panels are sheets into which diagonal ribs are machined to form a series of diamonds. The waffle panels are used around the middle (widest part of the LOX tank) to provide structural strength. Baffles adjacent to the aft facing sheet of the common bulkhead prevent wave action (sloshing) during flight. At the lower apex of the LOX tank, antivortex baffles, consisting of a 14-foot cruciform (four fins arranged in a cross) and 12 smaller baffles, are installed over the sump and engine supply line connections. The smaller baffles are essentially thin metal plates extending from the center of the cruciform, three between each pair of fins.

COMMON BULKHEAD

The common bulkhead may be likened to two giant domes, one placed inside the other, open end down, with a layer of insulation sandwiched between. The top dome is called the forward facing sheet and the bottom, the aft facing sheet. The forward facing sheet has a J-shaped periphery, which is welded to the No. 1 liquid hydrogen tank cylinder. In final assembly, a 15-inch, 12-section bolting ring is bolted to the aft skirt and the No. 1 liquid hydrogen tank cylinder. A total of 636 bolts attach the bolting ring to the liquid oxygen tank.


S-8

Bulkhead—Common bulkhead shows aft facing sheet (in sling preparatory to mating).

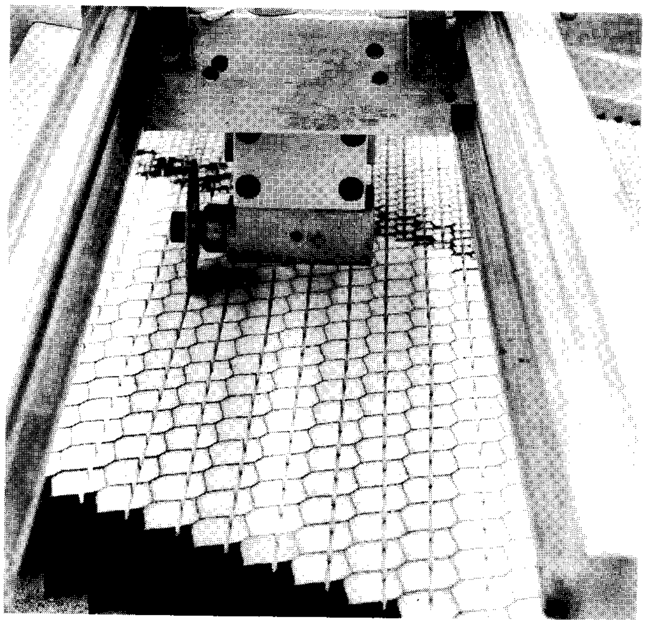

S-13

Gore Section of Aft Facing Sheet of Common Bulkhead Before Assembly

Insulating and joining the forward and aft facing sheets into a common bulkhead is a process of several operations. First the aft facing sheet is placed on a bonding fixture and numerous sections of honeycomb phenolic insulation are fitted and tapered to exact but varying thicknesses. Then the insulation is cemented to the aft facing sheet in a multistage bonding operation which includes chemical processing of the aft facing sheet, application of adhesive, and pressurizing and curing in the autoclave. After mating the forward facing sheet over the insulated aft facing sheet, impression checks are made to assure a perfect fit. The forward facing sheet is then chemically processed, the insulation placed on the exposed top of the aft facing sheet is prepared with adhesive, and the entire bulkhead assembly is joined and placed in the autoclave for pressurizing and curing. In both bonding operations, checks are performed with ultrasonic equipment to ensure that the adhesive has completely covered the surface.

Liquid Hydrogen Tank

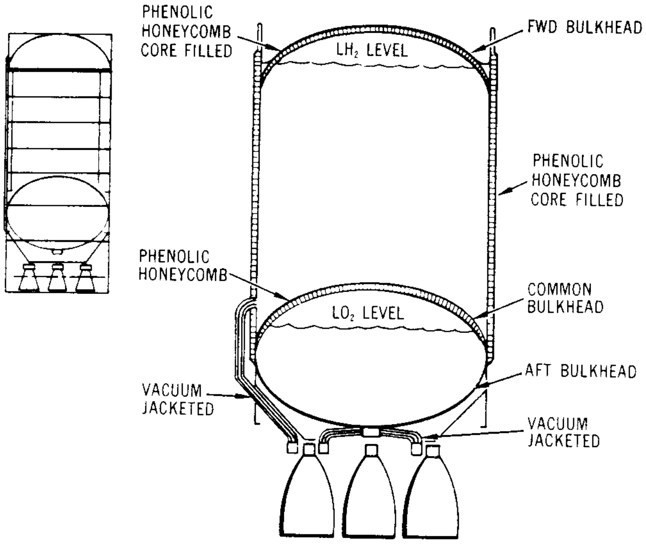
The liquid hydrogen cylinder walls comprise the main bulk of the second stage. Five of the cylinder walls measure slightly over 8 feet in height each, while the sixth, the No. 1 cylinder, is 27 inches high. Each of the six cylindrical sections is comprised of four curved, machined aluminum skins. Numerically machine-milled into the inside of the curved skins are stringers and ring frames. Riveted to the circumferential ring frames are flanged aluminum frames which extend inward for approximately 7 inches. In addition to structural rigidity, the frames act as slosh baffles for the liquid hydrogen.



S-7

Honeycomb Insulation

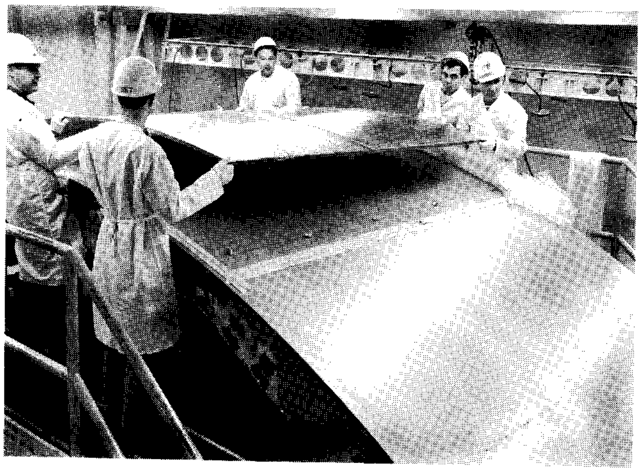
Special lightweight insulation and insulating techniques had to be developed to contain the cryogenic propellants of the second stage. The stage insulation helps maintain the liquid hydrogen at -423 degrees Fahrenheit and the liquid oxygen at -297 degrees Fahrenheit.



Second Stage Insulation

Insulation

The insulation varies in thickness in different parts of the stage. On the cylinder walls of the liquid hydrogen tank, it is only about 1-1/2 inches thick. On the common bulkhead, which separates the liquid hydrogen and liquid oxygen tanks, it varies from a tenth of an inch thick at the edges to about 4-3/4 inches thick at the apex of the bulkhead.



S-10

Insulating—Workers apply insulation to LH_2 cylinder panel.

The LH_2 tank wall insulation is formed of a phenolic honeycomb filled with a heat-resistant foam of isocyanate. The honeycomb is sealed top and bottom with a phenolic laminate and a layer of Tedlar plastic film. The helium is forced through passages (grooves) cut into the foam and honeycomb. The purge is continuous from the start of hydrogen loading until just before launch.

Systems Tunnel

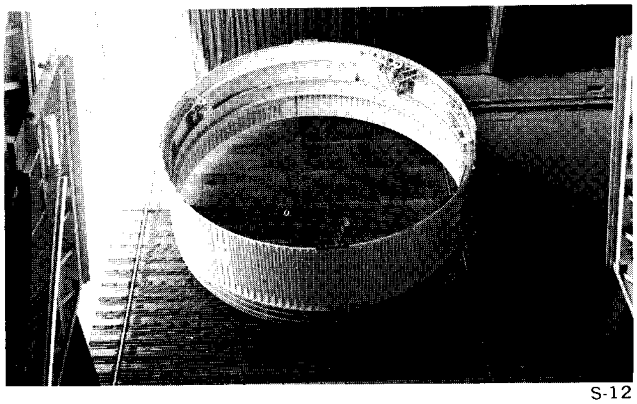
The systems tunnel is semicircular, approximately 22 inches wide, and almost 60 feet long. It is attached vertically to the outside wall cylinders, protecting and supporting instrumentation, wiring, and tubing, which connect system components located at both ends of the stage.

Forward Skirt Assembly

Of semimonocoque construction, the forward skirt is assembled from four curved sections with a height of 11-1/2 feet and has four internal support rings. Hat sections attached vertically to the outer skin stiffen the completed assembly and provide structural support for the third stage and Apollo payload. The forward ring has provisions for attachment to the mating ring of the third stage while skin and vertical members of the skirt attach to the forward end of the liquid hydrogen tank structure.

Final Assembly

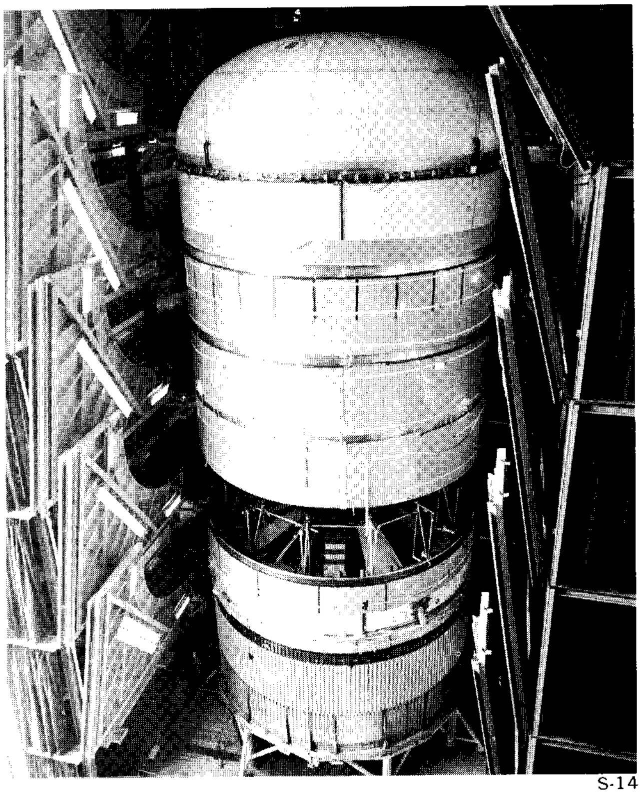
The second stage of the Saturn V launch vehicle assumes its shape in the vertical assembly building of NAA's Seal Beach facility. Assembly in the vertical position is based on a building-block concept. In this position, subassembly loading, circumferential exactness, and station locating is benefited by the even gravitational force exerted during each assembly operation. Constant checks and verification of station planes and stage alignment are maintained during each joining procedure by special scopes, levels, and traditional plumb bobs.


S-12

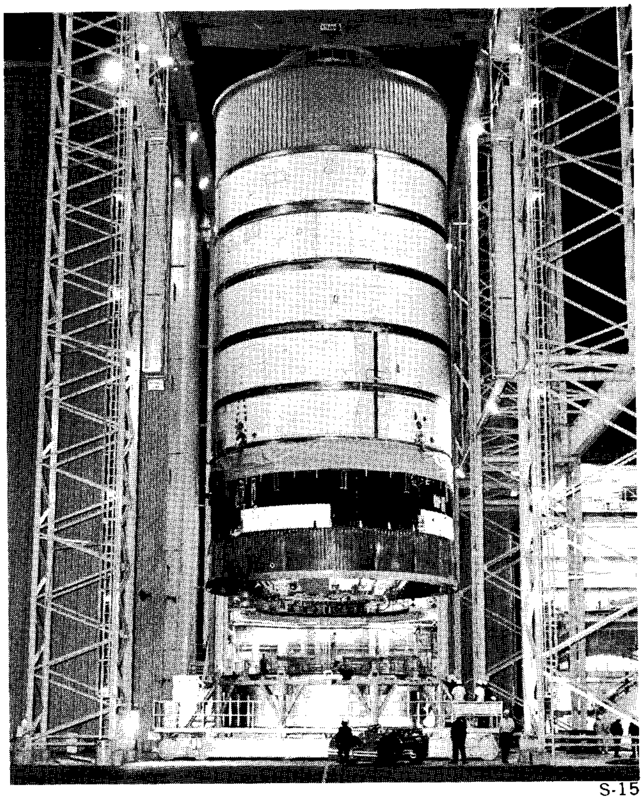
Second Stage Forward Skirt

Another reason for vertical assembly involves the welding of cylinders and bulkhead. If the stage were welded while in a horizontal position, temperature diversion over the circumference could result in harmful expansion near the top of the stage.

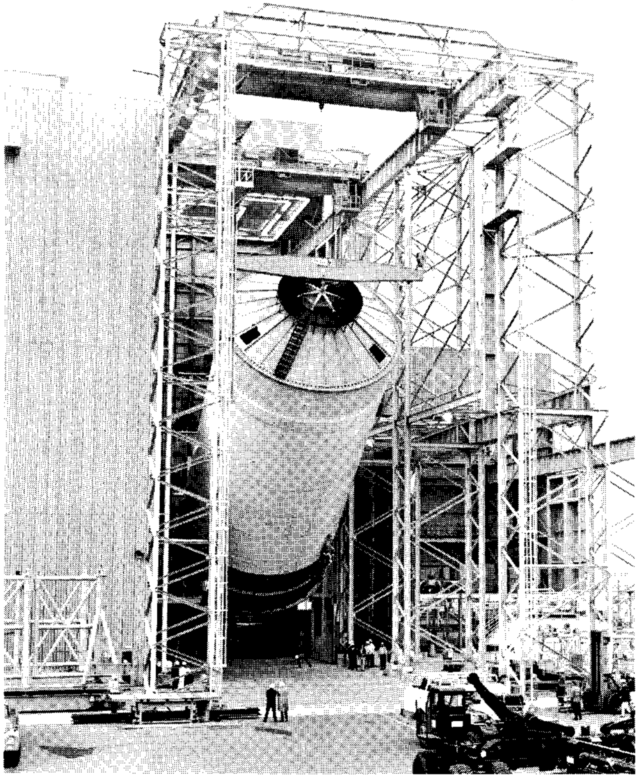
To facilitate movement of the huge components and of the stage itself, a motorized transfer table rolls from outside to inside the building. Essentially, the assembly sequence begins with the welding of the lower two cylinders. Then the common bulkhead is welded to that assembly. Next the uppermost cylinder is welded to the LH_2 forward bulkhead. The aft LOX bulkhead and the aft facing sheet of the common bulkhead are welded together to form the liquid oxygen tank, and the thrust structure and aft skirt are then assembled to it. The remaining cylinders are then welded to the stage, and the forward skirt is then mated to the stage stack. The interstage is fit-checked to the thrust structure before interstage systems are installed. Throughout the assembly and welding operations, hydrostatic, X-ray, dye penetrant, and other tests and quality control devices are performed to ensure that specifications are met. The liquid hydrogen portion of the second stage as well as the liquid oxygen tank are given a thorough cleaning after assembly. After each bulkhead is welded to its components, it is hydrostatically tested. After completion of stack weld operations, the entire stage is pneumostatically tested. After completion of these tests, the liquid hydrogen and liquid oxygen tanks are thoroughly cleaned.


S-14

Vertical Assembly of Stage

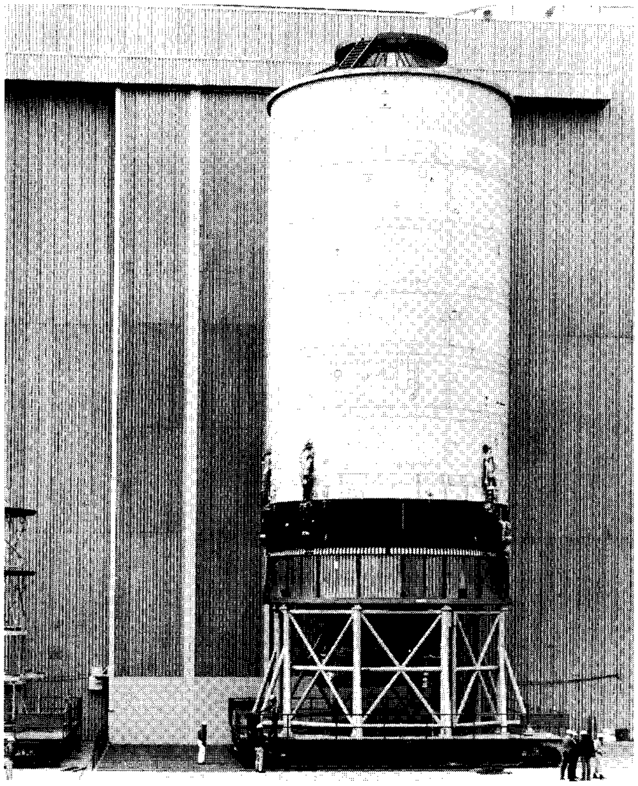

S-15

Move is Made—Flight stage is moved onto transporter to new station in vertical assembly building at Seal Beach.



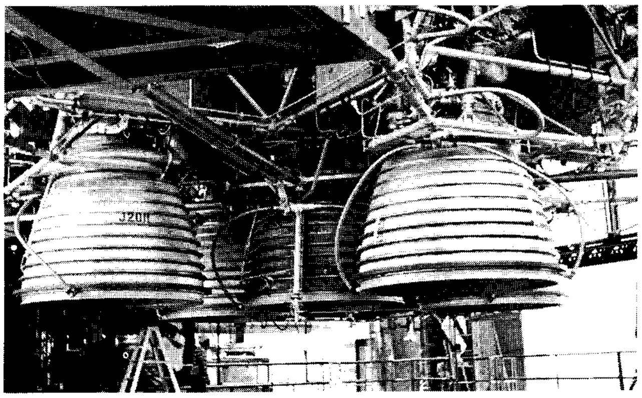
S-16

Repositioning—Second stage is turned horizontally for checkout operation.



S-18

Stage Complete—Flight stage moves on transfer table from assembly building to checkout building.



S-17

Engine Installation—J-2 engines are mounted in stage.

After assembly, the stage is moved to a vertical checkout building for final checks on all stage systems.

PROPELLANT SYSTEM

The propellant system is composed of seven subsystems: purge, fill and replenish, venting, pressurization, propellant feed, recirculation, and propellant management.

Purge Subsystem

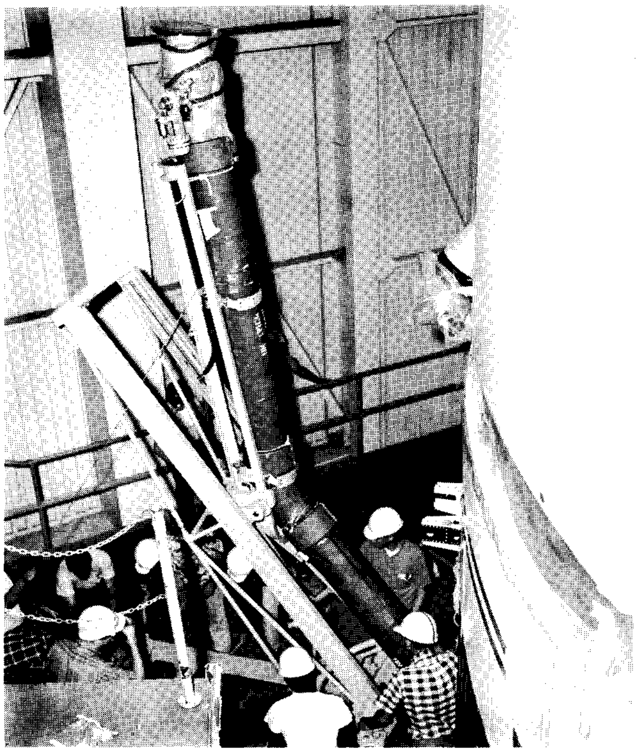
The purge subsystem uses helium gas to clear the propellant tanks of contaminants before they are loaded. The important contaminants are oxygen in the liquid hydrogen tank (liquid hydrogen will freeze oxygen which is impact-sensitive) and moisture in the liquid oxygen tank.

The tanks are purged with helium gas from ground storage tanks. The tanks are alternately pressurized and vented to dilute the concentration of contaminants. The operation is repeated until samples of the helium gas emptied from the tanks show that contaminants have been removed or reduced to a safe level.

Fill and Replenish Subsystem

Filling of the propellant tanks on the second stage is a complex and precise task because of the nature of the cryogenic liquids and the construction of the stage.

Because the metal of the stage is at normal outside temperature, it must be chilled gradually before pumping the ultra-cold propellants into the tanks. The filling operation thus starts with the introduction of cold gas into the tanks, lines, valves, and other components that will come into contact with the cryogenic fluids. The cold gas is circulated until



S-19

Channel Installed—Feed line from LH_2 tank to one of the five engines is installed.

the metal has become chilled enough to begin pumping in the propellants. The filling and replenishing subsystem operation consists of five phases:

Chilldown—Propellants are first pumped into the tank at the rate of 500 gallons per minute for LOX and 1,000 gallons per minute for LH_2. Despite the preliminary chilling by cold gas, the tanks are still so much warmer than the propellants that much of the latter boils off (converts to gaseous form) when it first goes into the tank. Filling continues at this rate until enough of the propellants remain liquid so that the tanks are full to the five per cent level.

Fast Fill—As soon as tank sensors report that the liquid has reached the five per cent level, the filling rate is increased to 5,000 gallons per minute for LOX and 10,000 gallons per minute for LH_2. This rate continues until the liquid level in the tank reaches the 98 per cent level.

Slow-Fill—Propellant tanks are filled at the rate of 1,000 gallons per minute for both LOX and LH_2 until the 100 per cent level is reached.

Replenishment—Because filling begins many hours before a scheduled liftoff and the cryogenic liquids are constantly boiling off, filling continues almost up to liftoff (160 seconds before liftoff for LOX and 70 seconds before liftoff for LH_2). Tanks are filled at the rate of up to 200 gallons per minute for LOX and up to 500 gallons per minute for LH_2, depending on signals from sensors in the tanks on the liquid level.

101 Per Cent Shutdown—A sensor in each tank will send a signal to indicate that the 101 per cent level (over the proper fill level) has been reached; this signal causes immediate shutdown of filling operations.

Filling is accomplished through separate connections, lines, and valves. The ground part of the connections is covered by special shrouds in which helium is circulated during filling operations. This provides an inert atmosphere around the coupling between the ground line and the tanks.

The coupling of the fill line and the tanks is engaged manually at the start of filling operations; it is normally disengaged remotely by applying pneumatic pressure to the coupling lock and actuating a push-off mechanism. A backup method involves a remotely attached lanyard in which the vertical rise of the vehicle will unlock the coupling. The fill valves are designed so that loss of helium pressure or electrical power will automatically close them.

Liquid oxygen is the first propellant to be loaded onto the stage. It is pumped from ground storage tanks. Liquid hydrogen is transferred to the stage by pressurizing the ground storage tanks with gaseous hydrogen. The liquid hydrogen tank is chilled before the liquid oxygen is loaded to avoid structural stresses.

After filling is completed, the fill valves and the liquid oxygen debris valves in the coupling are closed, but the liquid hydrogen debris valve is left open. The liquid oxygen fill line is then drained and purged with helium. The liquid hydrogen line is purged up to the coupling. When a certain signal is received (first stage thrust-commit), the liquid hydrogen debris valve is closed and the coupling is separated from the stage.

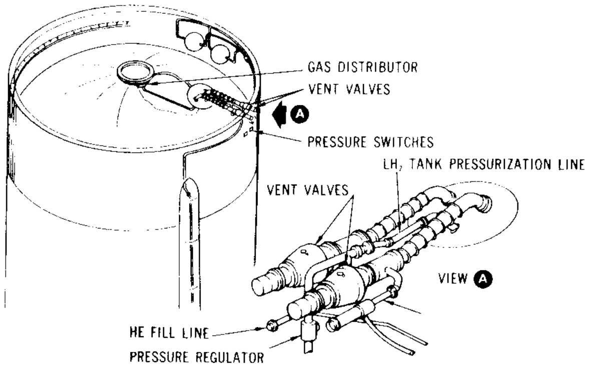
The tanks can be drained by pressurizing them, opening the valves, and reversing the filling operation.

Venting Subsystem

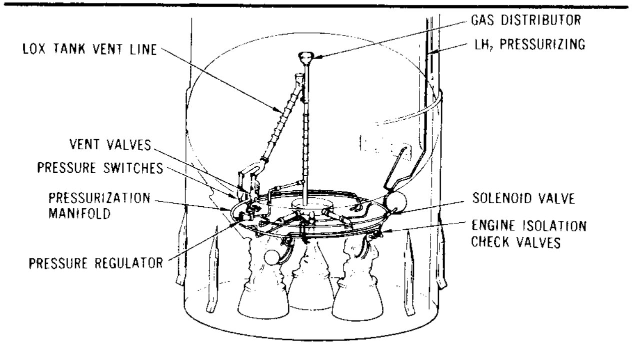
The venting subsystem is used during loading and flight operations. While the propellant tanks are being loaded, the vent valves (two for each tank) are opened by electrical signals from ground equipment to allow the gas created by propellant boil-off to leave the tanks. The valves are spring-loaded to be normally closed, but a relief valve will open them if pressure in the tanks reaches an excessive level. Each valve is capable of venting enough gas to relieve the pressure in its tank; two are provided in each tank as backup.

Pressurization

Pressurization of the propellant tanks is a three-stage process. Before launch, pressurization is obtained with gaseous helium from ground support equipment. After J-2 engine start, the pressurization is maintained with gaseous oxygen and gaseous hydrogen converted from the liquid oxygen and hydrogen.



LH_2 PRESSURIZATION SYSTEM—FWD



LH_2 & LOX PRESSURIZATION SYSTEM AFT

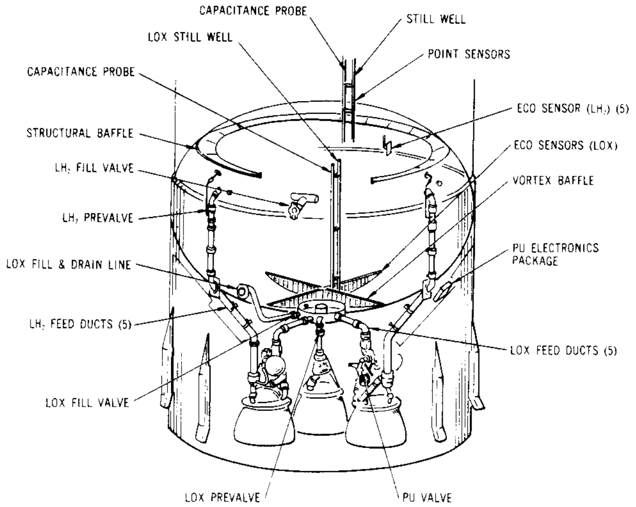
S-20

Propellant Pressurization System

Before launch, both the liquid hydrogen and liquid oxygen tanks are pressurized with gaseous helium which flows directly into the stage pressurization lines from ground storage tanks. Pressure switches, which sense ullage pressure, maintain the required pressurization level (37 to 39 psia for liquid oxygen and 31 to 33 psia for liquid hydrogen). This pressurization is maintained until liftoff; boil-off of the liquid propellants maintains adequate pressure until the stage's engines are ignited. A fitting on the upper manifold of the engines permits gaseous hydrogen (converted from its liquid state) to flow through a common manifold, a pressurization line, and regulator back to pressurize the liquid hydrogen tank to the desired levels. Some of the liquid oxygen is bled through a heat exchanger before reaching the combustion chamber, converted to a gaseous state, and diverted to a pressurization line and a regulator where it flows back to pressurize the liquid oxygen tank. The flow of pressurant gas into the LH_2 tank is automatically stepped up after 250 seconds of a J-2 engine firing, and the greater flow of gas and the increase of pressure continues for the rest of the firing period.

Propellant Feed Subsystem

The purpose of the propellant feed subsystem is simple: transfer the liquid propellants from their tanks to the five J-2 engines.



S-21

Feed System Components

Each tank has five prevalves which control or stop the flow through separate feed lines to the engines. Solenoids control helium pressure to open and close the valves; if pressure or electrical power is lost, the valves will automatically stay open.

The feed lines (except the center engine liquid oxygen line) are 8 inches in diameter and are vacuum-jacketed and insulated. The center engine liquid oxygen line is also 8 inches in diameter but is not insulated. Thermocouples (temperature measuring devices) in the vacuum jackets permit periodic vacuum checks; rupture discs in the jackets relieve excessive pressure. The feed lines also have bellows to allow for thermal expansion and freedom of movement.

Recirculation Subsystem

The main purpose of the recirculation subsystem is to keep the liquid propellants in the engine pumps. The subsystem, by keeping the propellants moving through lines, valves, and pumps, also keeps these parts chilled.

The LH_2 recirculation subsystem pumps the propellants through the feed lines and valves and back to the LH_2 tank through a single return line. The pumps are powered from a 56-volt DC battery system located in the interstage; the batteries are ejected with the interstage approximately 30 seconds after first plane separation. Before liftoff, power for the LH_2 recirculation subsystem is supplied by ground equipment.

The LOX recirculation system works on the basis of a thermal syphon; heat entering the system is used to provide pumping action by means of fluid density differences across the system. Helium gas is used to supplement the density differences and thereby improve the pumping action.

Recirculation of oxygen begins at the start of tanking; LH_2 recirculation begins just before launch. The propellants continue to circulate through first stage firing and up until just before the first stage and second stage separate. While the subsystems are operating, the LH_2 prevalves which lead to the combustion chambers are closed; as soon as the recirculation subsystem stops, the LH_2 prevalves open and the engines ignite.

Propellant Management System

The propellant management system controls loading, flow rates, and measurement of the propellants. It includes propellant utilization, propellant loading, propellant mass indication, engine cutoff, and propellant level monitoring subsystems.

PROPELLANT UTILIZATION SUBSYSTEM

The propellant utilization subsystem controls the flow rates of liquid hydrogen and liquid oxygen in such a manner that both will be depleted simultaneously. It controls the mixture ratio so as to minimize propellant residuals (propellant left in the tanks) at engine cutoff. Propellant utilization bypass valves at the liquid oxygen turbopump outlets control flow of liquid oxygen in relation to the liquid hydrogen remaining. Control of the engine mixture ratio increases the stage's payload capability. The propellant utilization subsystem is interrelated with the propellant loading subsystem and uses some of the same tank sensors and ground checkout equipment.

PROPELLANT LOADING SUBSYSTEM

The loading subsystem is used to control propellant loading and maintain the quantity of propellants in the tanks. Capacitance probes (sensors) running the full length of the propellant tanks sense liquid mass in the tanks and send signals to an airborne computer, which relays them to a ground computer to control loading. They also send signals to an airborne computer for the propellant utilization subsystem's control of flow rates.

PROPELLANT MASS INDICATION SUBSYSTEM

The propellant mass indication subsystem is integrated with the propellant loading subsystem and is used to send signals to the flight telemetry system for transmission to the ground. It utilizes propellant loading sensors to determine propellant levels.

ENGINE CUTOFF SUBSYSTEM

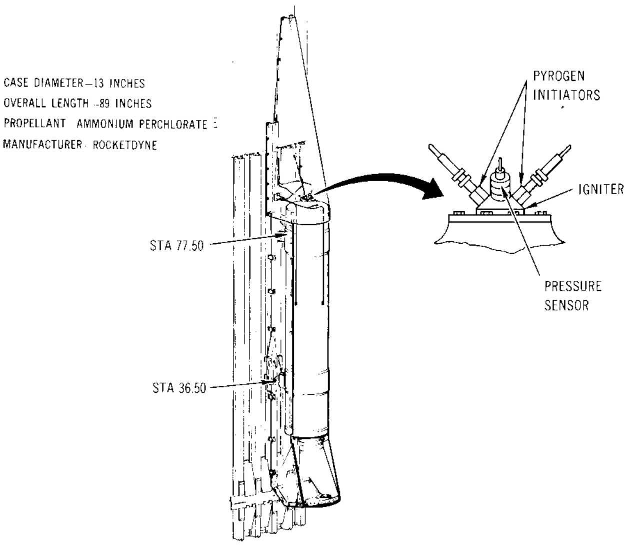
The main function of the engine cutoff subsystem is to signal the depletion point of either propellant. It is an independent subsystem and consists of five sensors in each propellant tank and associated electronics. The sensors will initiate a signal to shut down the engines when two out of five sensors in the same tank signal that propellant is depleted.

PROPELLANT LEVEL MONITORING SUBSYSTEM

The propellant level monitoring subsystem checks the level of propellants in both tanks to provide checkpoints for the sensors used in the propellant utilization and loading subsystems and to monitor propellant levels during firing. These functions are performed by sensors mounted on continuous stillwells adjacent or parallel to the full-length capacitance probes in each tank. There are 14 sensors on each stillwell to indicate various levels in the tanks.

ULLAGE MOTORS

The solid propellant ullage motors are used to provide artificial gravity by momentarily accelerating the second stage forward after first stage burnout. This moment of forward thrust is required in the weightless environment of outer space to make certain that the liquid propellant is in proper position to be drawn into the pumps prior to starting of the second stage engines.



S-22

Ullage Motor

Eight ullage motors are utilized on the stage where they are attached around the periphery of the interstage structure between the first and second stages. Each ullage motor measures 12.5 inches in diameter by 89 inches long and each provides 22,500 pounds of thrust for approximately 4 seconds. The motors utilize Flexadyne solid propellant in a formulation developed specifically to provide high performance and superior mechanical properties under operating conditions encountered in space. Ullage motor nozzles are canted 10 degrees to reduce exhaust impingement against the interstage structure.

THERMAL CONTROL SYSTEM

Thermal control is provided by a ground-operated system which maintains proper temperatures for the equipment containers in the forward and aft skirt areas. Tempered air is used to cool the containers before propellant loading. With preparation for loading, the air is changed to nitrogen for container inerting and heating. Separate thermal control systems are provided for the forward and aft skirt areas. Each of the units contains a single manifold connected to each container, individual fixed-flow orifices, and individual relief holes from each container. Container insulation and thermal inertia preclude excessive temperature changes.

FLIGHT CONTROL SYSTEM

Flight control of the second stage is maintained by gimbaling the four rocket thrust engines for thrust vector (direction) control. These are the four outboard engines; the fifth J-2 engine located in the center of the cluster is stationary.

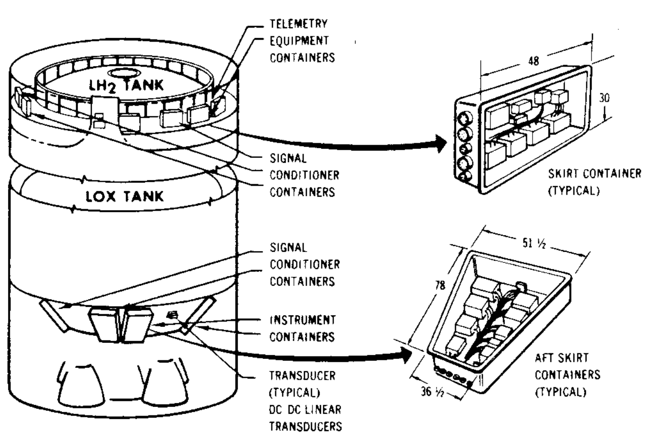
Each outboard engine has a separate engine actuation system to provide the force to position the engine. Gimbaling is achieved by hydraulic-powered actuators controlled by electrical signals generated through a flight control computer located in the instrument unit just above the third stage. Hydraulic power for operating each of the gimbaling actuators is supplied by individual engine-driven hydraulic pumps. Each system is self-contained and operates under a pressure of 3,500 psi. The components of each hydraulic system are attached to the thrust structure above each of the outboard engines. The main hydraulic pump is driven by the liquid oxygen turbopump on the respective engine. Two servoactuators that control each engine programmed for gimbaling are located on the engine outboard side. One is on the pitch plane, and the other on the yaw plane. Each actuator will gimbal the engine plus or minus 7 degrees in pitch or yaw and plus or minus 10 degrees in combination to correct for roll errors at a minimum rate of 8 degrees per second.

During flight, the guidance system continuously determines an optimum vehicle steering command based on the vehicle's position, velocity, and acceleration. This system, located in the instrument unit, has a guidance signal processor which delivers attitude correction signals to the flight control computer in the instrument unit. These signals are shaped, scaled, and summed electronically. These summed error signals are then directed to the servoactuator amplifiers, which, in turn, drive their respective servoactuators in the second stage. These signals cause the servoactuators to position the engines.

MEASUREMENT SYSTEM

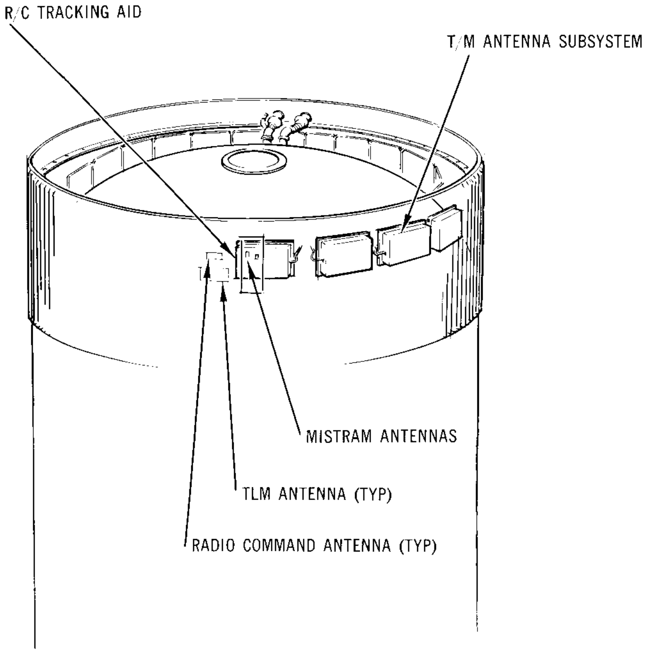
A wide variety of transducers and signal conditioners is used in the instrumentation system, which feeds signals to a high-level telemetering system for transmission to the ground. The various instrumentation sensors monitor pressure, temperature, and propellant flow rates within the tanks. Other sensors record the amount of vibration and noise, and flight position and acceleration.

Tied into the measurement system are telemetry and radio frequency subsystems which transmit the performance signals to ground receiving stations for immediate (real-time) and postflight vehicle performance evaluation. Antennas which serve the telemetry and radio frequency subsystems are flush-mounted on the forward skirt and are omnidirectional in coverage.



S-23

Telemetry System



S-24

Radio Frequency System

ELECTRICAL SYSTEM

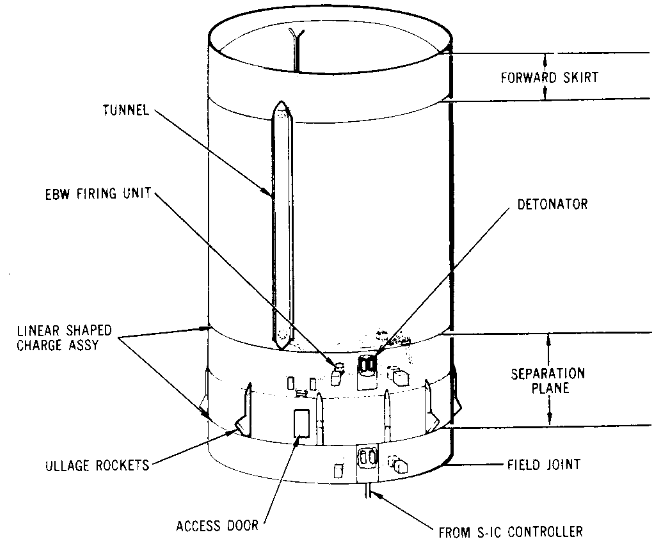
In flight, the second stage electrical system is powered by four 28-volt DC batteries which operate four DC bus systems. The main DC bus powers electrical controls for the pressurization and propellant management systems, J-2 engine control, and the electrical sequence controller. The instrumentation DC bus powers instrumentation and telemetry. The ignition and recirculation bus systems provide the electrical supply for those flight operations.

Both the main and instrumentation bus systems are powered by individual 28-volt DC silver-oxide/zinc batteries. The recirculation DC bus system is powered by two series-connected 28-volt DC silver-oxide/zinc batteries. This supplies electrical power to five recirculation pump-motor inverters. The inverters convert the 56 volts to three-phase, 400 cps, quasi-square wave power to the AC induction motors on the liquid hydrogen recirculation pump systems. The ignition system receives its electrical power from a tap on the recirculation system flight batteries through a power transfer switch.

Each of the flight electrical power bus systems has a power transfer switch—an electro-mechanical device for transferring the systems from ground service equipment power (prelaunch) to stage battery power for flight. Before flight, the electrical power system and its electrical controls are energized from regulated ground service equipment power.

ORDNANCE SYSTEM

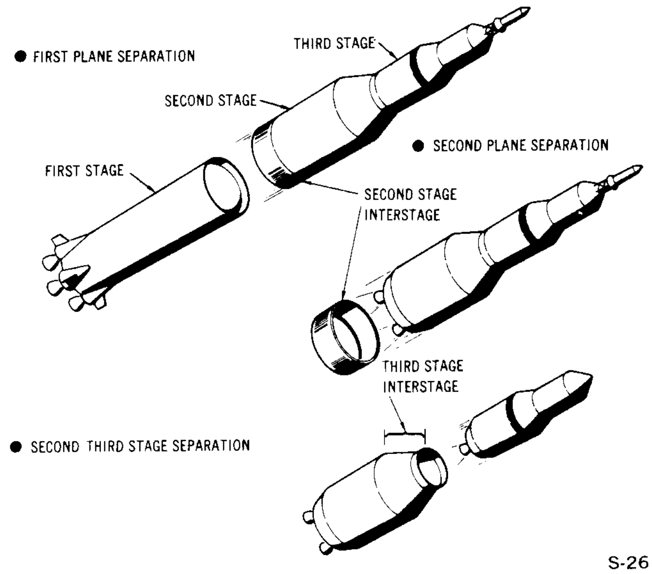
The separation of the first and second stages is a dual-plane separation. With depletion of the first stage propellants, an engine cutoff signal is initiated. A linear-shaped charge utilizing exploding bridge-wire initiators physically severs the two stages.



S-25

Ordnance System

Simultaneously, retrorockets on the first stage are ignited to decelerate the first stage and complete the separation, and in order to assure good propellant flow to the five J-2 engines of the second stage after first stage separation, eight solid-propellant ullage motors located on the second stage's aft interstage are fired to establish positive vehicle acceleration and proper propellant settling. When the outboard engines of the second stage reach 90 per cent of maximum thrust, a signal is transmitted which initiates interstage separation. An explosive charge separates the interstage from the aft skirt of the second stage.



Separation System

Approximately 10 seconds before second stage propellant depletion, a signal activates the separation system which will sever the second stage from the third. An interstage connecting the second and third stage has four retrorockets which are fired to decelerate the second stage.

GROUND SUPPORT

Ground support operations play an important part in getting the second stage ready for operation. Among the vital operations in this area are checkout (performed mostly with complex electronic equipment and computerized routines which stimulate stage systems and analyze responses), leak detection and insulation purge, and engine compartment conditioning.

Leak Detection and Insulation Purge

The purpose of this system is to detect hydrogen, oxygen, or air leaks; to dilute and remove leaking gases; and to prevent air from liquifying during tanking operations.

Any operation involving liquid hydrogen can be extremely hazardous; liquid hydrogen in the presence of oxygen can explode or create a fire. The low-temperature atmosphere of liquid hydrogen causes air to liquify and solidify against the hydrogen tank wall if there is any leak in the tank insulation. The organic portion of the insulation will become impact-sensitive when drenched in liquid air or oxygen; insulation saturated with cryopumped air will add weight to the stage and could cause damage during draining because of a pressure buildup created by the liquified air returning to a gas. For these reasons, detection, control, and elimination of any hydrogen leaks from the stage and ground equipment are of great importance.

The leak detection system checks out the liquid hydrogen tank, tank insulation, and the common bulkhead. The areas to be checked are divided (tank wall, forward bulkhead, and common bulkhead), each with inlet and outlet taps. A gas analyzer determines the concentration of hydrogen in the purge gas (helium) after it has been forced through the insulation, and thus indicates any leakage.

From the start of hydrogen loading until launch, the insulation and core of the common bulkhead are continuously purged of hazardous gases.

Vacuum equipment is used for evacuation to prevent pressure buildup in the insulation and bulkheads by removing trapped gases. The insulation purge prevents air from entering the insulation in the event of damage during cryogenic operations.

Engine Compartment Conditioning

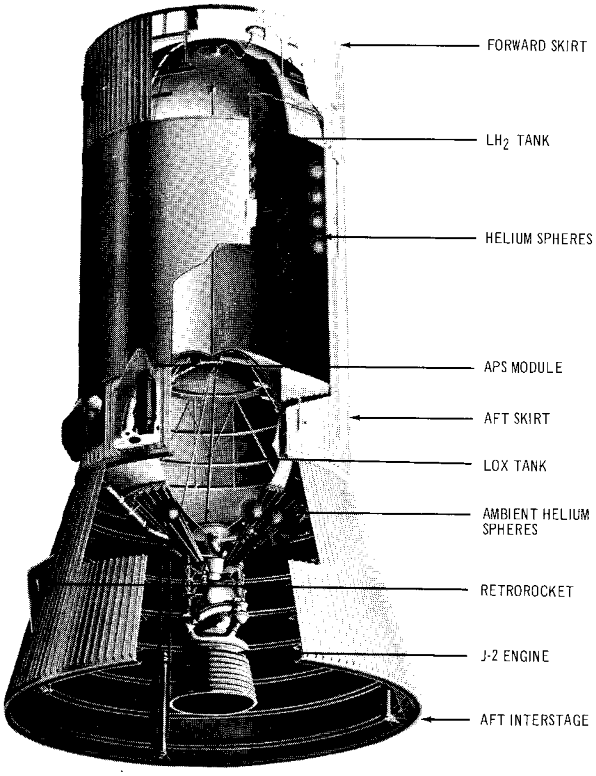
The purpose of this system is to purge the engine and interstage areas of explosive mixtures and to maintain proper temperature in critical regions of the aft compartment of the second stage. The compartment is purged before tanking and while the propellants are loaded.

The system consists of a 13-inch diameter feed line, manifold, ducts, and a series of vents surrounding the engine compartment and skirt area. The system provides temperature control for the hydraulic systems and certain components on the J-2 engines. The purge gas is forced through orifices in the manifold to the following areas requiring warming: the area between the thrust structure and the liquid oxygen tank, the bottom of the thrust structure including the lower surface of the thrust cone, the aft skirt and interstage, and the top surface of the heat shield.

The vent holes are located under the supporting hat sections on the outside of the aft skirt; this prevents wind, rain, and dust from entering the engine compartment. The vents are located so that the flow pattern provides good thermal control and expels hazardous gases.

The aft skirt and interstage are purged with warm (80 to 250 degrees) nitrogen. The nitrogen is sent through the feed line into the manifold, and then through ducts to the temperature-sensitive areas. By maintaining a 98 per cent nitrogen atmosphere in the engine compartment, desired temperatures are maintained and the danger of fire or explosion resulting from propellant leaks are minimized.

THIRD STAGE FACT SHEET



DAC-13067

NOTE:

Figures given for weights and contents are average. These may vary to meet requirements for the differing missions. In those cases where the numbers in the fact sheet differ with the text, the fact sheet contains more current information.

WEIGHT: 33,600 lb. (dry) including 7,700-lb. aft interstage
265,600 lb. (loaded)

DIAMETER: 21 ft. 8 in.

HEIGHT: 58 ft. 7 in.

BURN TIME: 1st burn—2.75 min. (approx.)
2nd burn—5.2 min. (approx.)

VELOCITY: 1st burn—17,500 miles per hour at burnout (approx.)
2nd burn—24,500 miles per hour (approx. typical lunar mission escape velocity)

ALTITUDE AT BURNOUT: 115 miles after 1st burn and into a translunar injection on 2nd burn

MAJOR STRUCTURAL COMPONENTS

AFT INTERSTAGE	THRUST STRUCTURE	COMMON BULKHEAD
AFT SKIRT	PROPELLANT TANK	FORWARD SKIRT

MAJOR SYSTEMS

PROPULSION: One bipropellant J-2 engine
Total Thrust: 225,000 lb. (maximum)
Propellants: LH$_2$—69,500 gal. (39,750 lb.)
LOX—20,150 gal. (192,250 lb.)

HYDRAULIC: Power for gimbaling J-2 engine

ELECTRICAL: One 56 VDC and three 28 VDC batteries, providing basic power for all electrical functions

TELEMETRY AND INSTRUMENTATION: A pulse code modulated/frequency modulator (PCM/FM) subsystem, providing transmission of flight data to ground stations

ENVIRONMENTAL CONTROL: Provides temperature-controlled environment for components in aft skirt, aft interstage, and forward skirt

ORDNANCE: Provides explosive power for stage separation, retrorocket ignition, ullage rocket ignition and jettison, and range safety requirements

FLIGHT CONTROL: Provides stage attitude control and propellant ullage control

THIRD STAGE

STAGE DESCRIPTION

Basically, the Saturn V third stage, the S-IVB, is an aluminum air-frame structure powered by a single J-2 engine, which burns liquid oxygen and liquid hydrogen. The engine has a maximum thrust of 225,000 pounds. The structure has a bipropellant capacity of 228,000 pounds of fuel and oxidizer.

STAGE FABRICATION AND ASSEMBLY

The third stage structure consists of a forward skirt assembly, propellant tank assembly, thrust structure assembly, aft skirt assembly, and aft interstage assembly. The propellant tank assembly consists of a single tank separated by a common bulkhead into a fuel compartment and an oxidizer compartment.

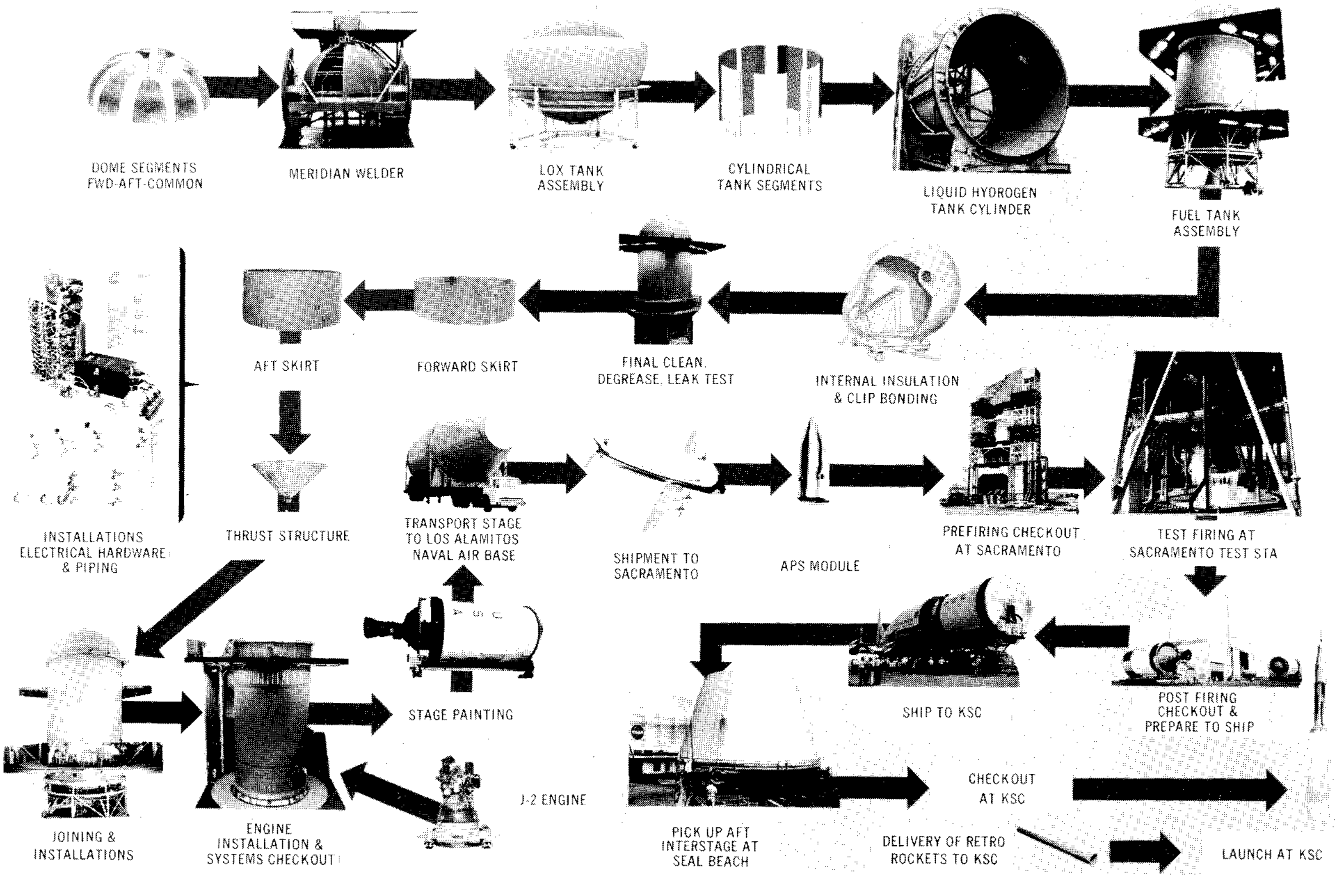
Forward Skirt Assembly

The forward skirt is a cylindrical aluminum skin and stringer structure that provides a hard attach point for the instrument unit. In addition, the forward skirt provides an interior mounting structure for electrical and electronic equipment that requires environmental conditioning, as well as range safety and telemetry antennas mounted around the exterior periphery. Environmental conditioning for electronic equipment is provided by cold plates which utilize a coolant supplied from the IU thermoconditioning system.

Propellant Tank Assembly

Structural elements of the propellant tank assembly are a cylindrical tank section, common bulkhead, aft dome, and forward dome. Seven segments are machined from aluminum alloy plate to form the tank section. A waffle pattern is then machine-milled into each segment to reduce weight and provide shell stiffness. The formed segments are joined into a complete cylinder by single-pass internal weld on a Pandjiris welding machine.

Aft and forward domes are made by forming "orange peel" segments on a stretch press. Orange peel segments are then joined in a dome welder. Each



DAC-16183

Third Stage Production Sequence

dome assembly rotates in the fixture under a stationary welding head which is automatically positioned by a servo-controlled sensing element. To complete the hemisphere, a 43-inch "dollar" segment is bolted in the top center of the dome. Subsequently, all fittings for various tank connections are installed by machine weld.

COMMON BULKHEAD

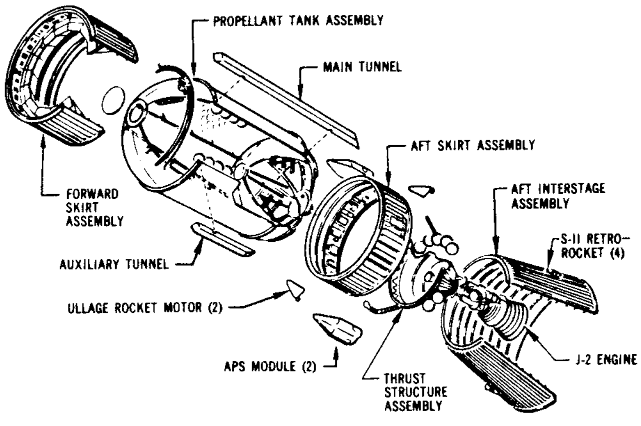
The common bulkhead, which forms the physical separation between the LOX and hydrogen tanks, is a 130-inch-radius hemisphere consisting of two aluminum domes separated and insulated by a fiberglass honeycomb core. The honeycomb core is bonded between the two domes under heat and pressure. Edges of two peripheral tees are welded together to provide a seal for the core. Joining of the common bulkhead and the aft dome completes the LOX tank subassembly. A slosh-baffle located within the LOX tank breaks up any wave action of the oxidizer during flight. The baffle is made up of four rings supported by "A" frames.

Thrust Structure Assembly

The thrust structure distributes J-2 engine thrust over the entire tank circumference. In addition, hydraulic system components, propellant feed lines, propellant tank ambient helium pressurization spheres, pneumatic components, and miscellaneous components which support engine operation are mounted on the thrust structure assembly.

Aft Skirt Assembly

The aft skirt is a cylindrical structure fabricated of aluminum, stringer-stiffened skin panels and provides structural interface between the aft interstage and propellant tank assembly. After second stage burnout, the second stage separates from the third stage at a separation plane located on the aft skirt assembly. Two auxiliary propulsion system (APS) engine modules, two ullage rocket modules, stage separation systems, an aft umbilical connector plate, and associated system support equipment are located on the aft skirt assembly.

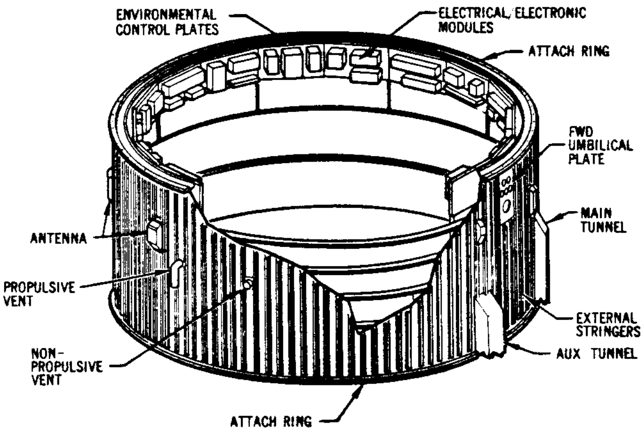


D-NRV-1

Third Stage Exploded View

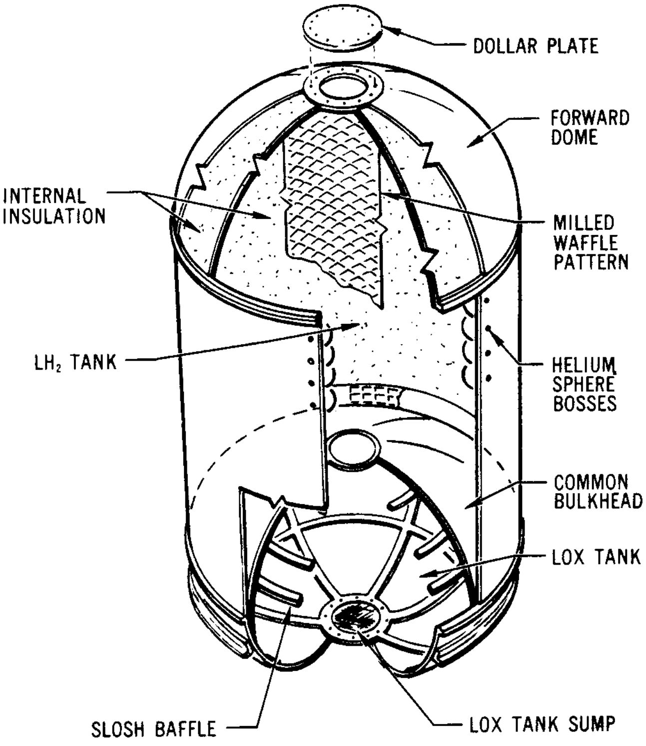
Aft Interstage Assembly

The aft interstage is a truncated cone-shaped structure fabricated of aluminum skin and stringers. It attaches to the third stage aft skirt and provides the structural interface to the second stage. It also houses the second stage retrorockets.



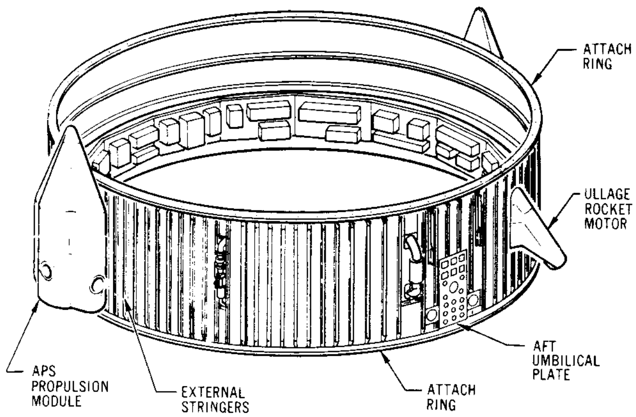
D-NRV-28

Forward Skirt Assembly



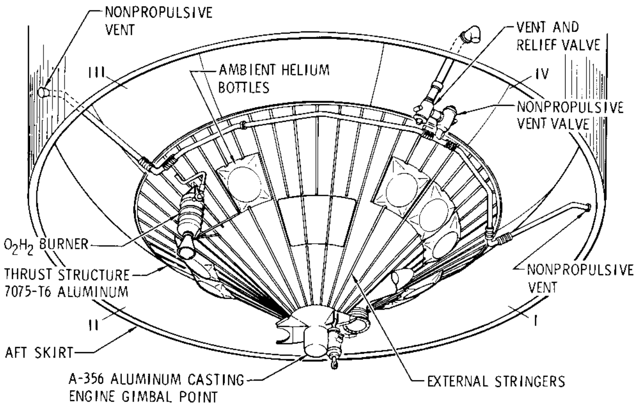
D-PB-110

Propellant Tank Assembly



Aft Skirt Assembly

D-NRV-4



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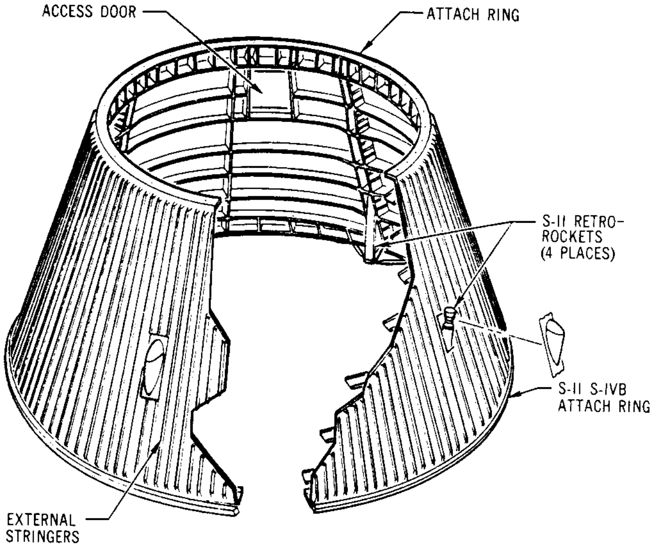
Thrust Structure Assembly

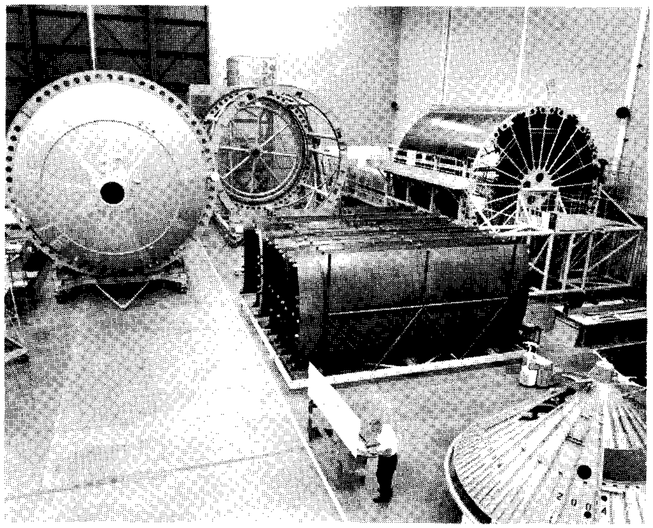


D-NRV-2

Aft Interstage Assembly

Final Assembly

Final assembly of the third stage propellant tank structural components is accomplished in the assembly and welding tower. The assembly is then removed from the tower and transported to the insulation chambers building where the LH_2 tank insulating tiles are fitted and installed, a glass cloth liner is placed on the insulation, and sealant is added. Propulsion system components, internally mounted in the LH_2 tank, are installed following the completion of tank insulation. The hydrogen tank contains a slosh baffle and wave deflector system, which contributes to fuel ullage control during flight, and eight cold helium storage spheres for LOX tank pressurization. The structure is then returned to the assembly tower where the forward and aft skirts and thrust structure are installed.



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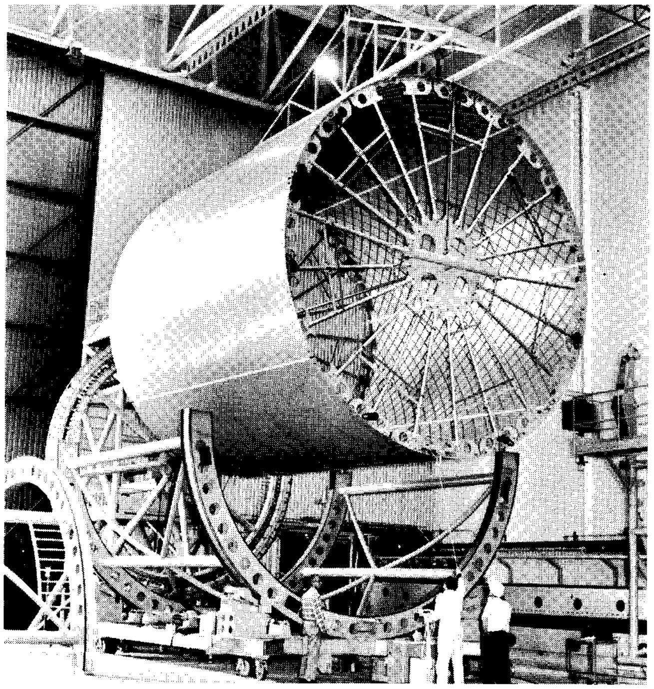
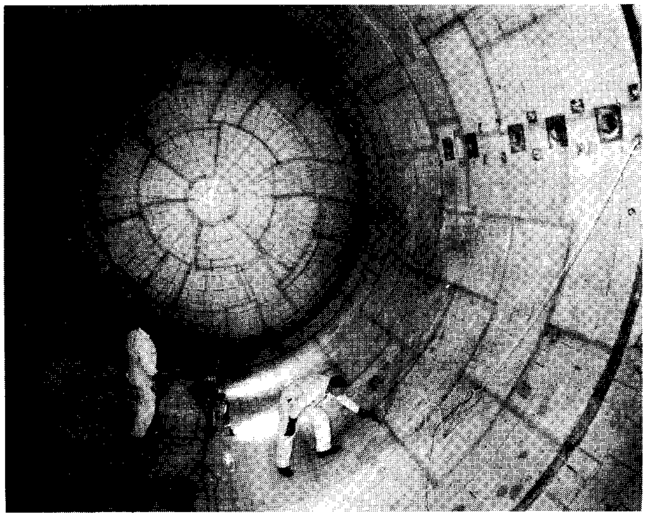
Propellant Tank Assembly Area at Huntington Beach



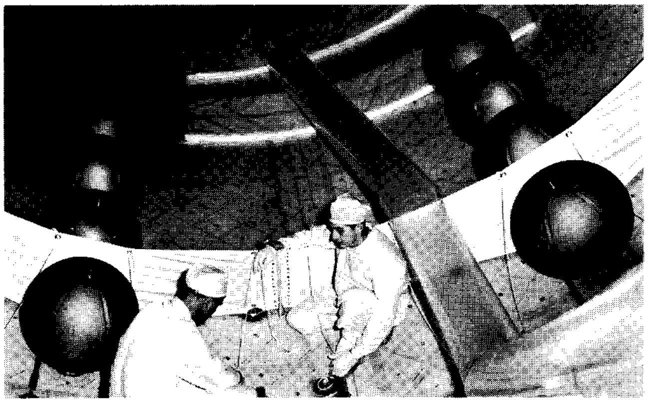
D-NRV-33

Tank Section in Trim and Welding Jig



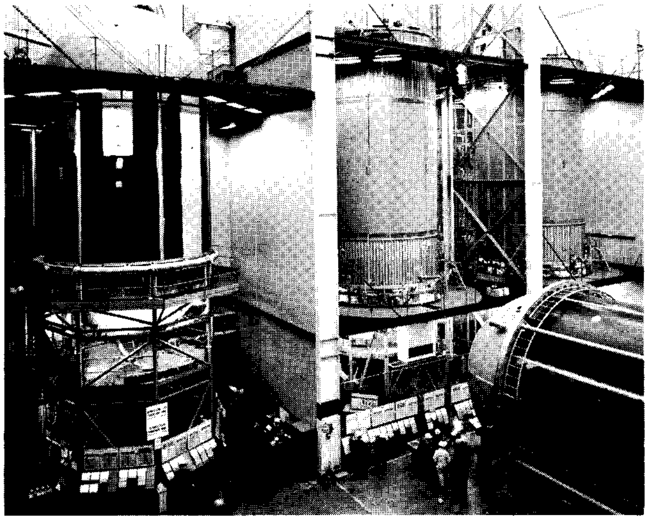
D-NRV-39

LH_2 Tank—Workmen apply resin to insulation tiles in LH_2 tank.



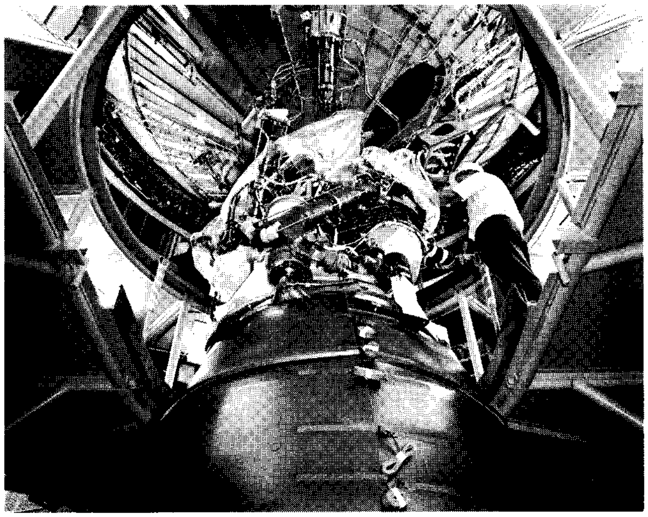
D-NRV-40

Slosh Baffle—Horizontal rings are installed inside LH_2 tank for propellant stabilization during flight.



D-NRV-42

Third stage vehicles reach end of assembly sequence with final assembly and checkout in 115-foot vertical towers.



D-NRV-32

Engine Installed—J-2 engine is attached to stage in final assembly tower at Huntington Beach.

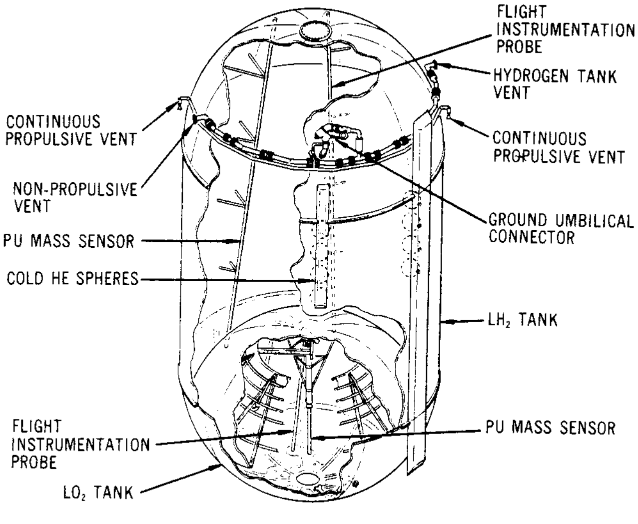
Final installation of various subsystem components is performed in a checkout tower, along with the installation and alignment of the J-2 engine. The stage is in a vertical position in the tower where a complete stage checkout of subsystems and systems is conducted except for actual ignition of engine. After satisfactory checkout, the stage is removed from the tower, placed on a dolly, and ground support rings are installed at each end of the stage. It is then painted, weighed, and prepared for shipment to the Douglas Sacramento Test Center for simulated and static firing of APS engines and J-2 engine.

THIRD STAGE SYSTEMS

Major systems required for third stage operation are the propulsion system, flight control system, electrical power and distribution system, instrumentation and telemetry system, environmental control system, and ordnance systems.

Propulsion System

The propulsion system consists of the J-2 engine, propellant system, pneumatic control system, and propellant utilization system. The J-2 engine burns LOX as an oxidizer and LH_2 as fuel at a nominal mixture ratio of 5:1. Both fuel and oxidizer systems utilize tank pressurization systems and have vent and relief capabilities to protect the propellant tanks from overpressurization. The pneumatic control system regulates and controls both the oxidizer and fuel systems. The propellant utilization (PU) system assures simultaneous and precise fuel and oxidizer depletion by controlling engine mixture ratio.



D-NRV-5

Propulsion System Components

J-2 ENGINE

The engine system consists of the J-2 engine, propellant feed system, start system, gas generator system, control system, and a flight instrumentation system. The propellant feed system utilizes independently driven, direct-drive fuel and oxidizer turbopumps to supply propellants at the proper mixture ratio to the engine combustion chamber. Additional information on the J-2 engine system may be found in the J-2 Engine section.

PROPELLANT SYSTEM

The propellant system consists of related stage subsystems to support an initial J-2 engine propulsive burn phase and an engine restart capability to provide a second J-2 engine propulsive burn phase for the third stage. It includes the oxidizer system, fuel system, pressurization system, repressurization system, tank venting system, and chill-down recirculation system.

Oxidizer System

LOX is loaded into the LOX tank at a temperature of -297° Fahrenheit through a LOX fill and drain line assembly which disconnects automatically at the time of vehicle liftoff.

The LOX tank capacity is 2,828 cubic feet which provides tankage for approximately 191,000 pounds (20,000 gallons) of usable LOX. The tank is pressurized with gaseous helium at 38 to 41 psia, and is maintained at this pressure during liftoff, boost, and stage engine operation.

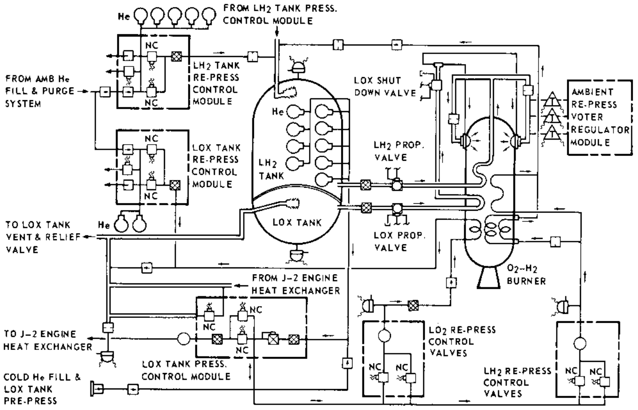
Fill and Drain—The LOX filling operation consists of purging and chilldown of the tank and filling in four stages: slow fill, fast fill, replenish (topping), and pressurization. The ground-controlled combination vent and relief valve is pneumatically opened at the start of the fill operation.

During slow fill, LOX is loaded at a rate of 500 gpm until five per cent of full level is attained; then fast fill at 1,000 gpm is initiated. When 98 per cent of the LOX has been loaded, the fill rate is reduced to a rate of 0 to 300 gpm. When the LOX tank is 100 per cent loaded, the full level is maintained until liftoff by a replenish flowrate of 0 to 30 gpm, as required to compensate for LOX boil-off.

If for any reason the LOX tank becomes overpressurized during fill, such as a vent malfunction, or by an excessive LOX fill flow, a pressure switch signals the LOX ground fill valve to close.

The LOX tank is capable of being unloaded by reversing the flow through the fill system under tank pressure and/or from gravity effect. Drain capacity is at 500 gpm at 33 psia.

LOX Tank Pressurization—The LOX tank is pressurized at 38 to 41 psia from a ground supply of cold helium regulated to -360° Fahrenheit. After liftoff and until engine ignition, the LOX tank pressure is maintained from nine helium storage spheres located in the LH_2 tank that have been charged to 3,100 ± 100 psi at -410° Fahrenheit. During J-2 engine burn, the engine heat exchanger heats and expands a portion of the helium flow before it is fed into the LOX tank. An ullage tank pressure switch controls inflight pressurization by opening or closing cold helium flow to the heat exchanger as required. In case of regulator failure in flight, a pressure switch, plenum chamber, and solenoid valve act as a backup pressure regulator.



D-NRV-6A

Oxidizer Tank Pressurization System

LOX Tank Repressurization—During the coast phase, prior to J-2 engine second ignition, the LOX tank pressurization system is inoperative, allowing LOX tank pressure to decay. However, before the second engine ignition the LOX tank is pressurized by the LOX tank repressurization system. The repressurization system utilizes cold helium from the nine spheres located in the fuel tank. An O_2/H_2 burner heats and expands the helium prior to its entry into the LOX tank. The helium flow is controlled by two parallel-redundant valves which are actuated by a LOX tank pressure switch. Repressurization flow is initiated if LOX tank pressure is below 38 psia and terminated at 41 psia.

Two ambient helium spheres located on the thrust structure provide redundancy to the O_2/H_2 burner repressurization system operation. Should the burner malfunction, or other circumstances require another orbit prior to second ignition, the ambient system, controlled by the LOX repressurization control module, can repressurize the LOX tank for a second time.

At second ignition and continuing through the second burn phase, the LOX tank is pressurized with cold helium gas heated by the engine heat exchanger and supplied by spheres contained within the LH_2 tank.

LOX Tank Vent-Relief System—The LOX tank vent-relief system consists of a tee assembly with a pneumatically operated vent valve and a backup relief valve. Pneumatic operation is provided by the LOX vent actuation module using helium gas from the pneumatic control system. The vent-relief valve is opened during the ground fill operation and closed prior to pressurization. During fill operations, the vent valve is capable of venting all LOX vapor.

The relief valve backup system automatically relieves at 45 psia and reseats at 42 psia. During liftoff and nonpowered stage flight, pressure relief or venting is not anticipated. However, the vent system becomes operational in the event of LOX tank overpressurization.

LOX Feed System—Prior to vehicle liftoff and prior to engine restart for the second burn phase, all LOX feed system components of the J-2 LOX turbopump assembly must be "chilled" to operating temperature for proper operation. Chilldown of the LOX system is accomplished by a closed-loop, forward-flow recirculation system. On command from the IU, a prevalve in the LOX feed duct closes and a bypass shutoff valve opens. An auxiliary, electrically driven centrifugal chilldown pump, mounted in the LOX tank, starts and LOX chilldown circulation begins. LOX is circulated from the LOX tank, through the low pressure feed duct, to the J-2 engine LOX pump and bleed valve, then back to the LOX tank through return lines. The pump is capable of delivering a minimum flowrate of 31 gpm at 15 psia. Recirculation chilldown continues through the boost phase and up to the time for J-2 engine ignition. In the event of an emergency, the chilldown system shutoff valve closes upon command from the IU.

A low pressure supply duct supplies LOX from the tank to the engine at a nominal flowrate of 391 pounds per second at -297° Fahrenheit and 37 psia.

The main LOX feed valve is a 4-inch butterfly-type valve and opens in two distinct steps: the first, a partially opened position; the second, a fully opened position. The LOX feed valve is solenoid-controlled.

A signal from the engine sequencer energizes the LOX feed valve, as required, to obtain steady-state operation. During steady-state operation, LOX feed is regulated by a propellant utilization valve which controls the oxidizer flow to the engine. A complete description of engine operation may be found in the J-2 Engine section.

Fuel System

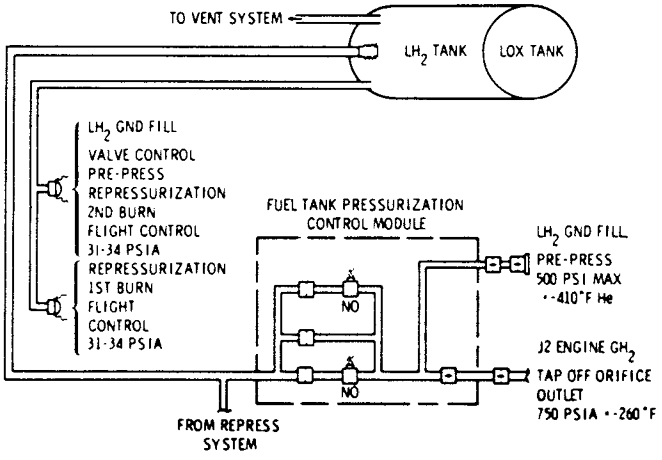
LH_2 is loaded into the insulated fuel tank at a temperature of -423° Fahrenheit through the LH_2 fill and drain valve assembly which is automatically disconnected at time of vehicle liftoff. The tank capacity is 10,446 cubic feet providing tankage for approximately 37,000 pounds (63,000 gallons) of usable fuel. The tank is pressurized from a ground source of helium at 31 to 34 psia. During liftoff, boost, and stage engine operation, pressure is maintained in the fuel tank at 31 to 34 psi.

Fill and Drain—The LH_2 loading operation consists of purging, chilldown of the tank, and filling in four stages: slow fill, fast fill, replenish (topping), and pressurization.

Immediately prior to LH_2 input into the tank, a combination vent and relief valve is pneumatically opened. Loading is then initiated at 500 gpm until five per cent of full level is reached; then fast fill begins. During fast fill, LH_2 is supplied to the tank at 3,000 gpm. When 98 per cent of the loading is completed, the loading rate is curtailed to between 0 to 500 gpm. When the LH_2 tank is 100 per cent loaded, the full level is maintained by a replenish flowrate of 0 to 300 gpm as required to compensate for LH_2 boil-off. During the final topping operation, the fuel tank venting system is closed and the tank is simultaneously pressurized from the ground source of helium. If overpressurization of the tank should occur during fill or during the boost phase, a relief valve, which is spring-loaded to open at 37 psia and close at 34 psia, is actuated to relieve excess pressure.

The LH_2 tank is capable of being unloaded through the fill system. LH_2 unloading is accomplished by reversing the flow through the fill system under tank pressure and/or from gravity effect.

Fuel Tank Pressurization—During initial tank pressurization, an external tank connection is made to a ground supply of helium. The helium is supplied to the fuel tank at -360° Fahrenheit at 600 psig. When the tank ullage pressure reaches a maximum of 31 to 34 psia, a pressure switch sends a signal to deactivate the ground pressurization valve indicating that a satisfactory liftoff pressure has been attained, and pressurization is discontinued.



D-NRV-7A

Fuel Tank Pressurization System

During liftoff and prior to J-2 engine start, additional pressurization is not required, as tank ullage pressure will be maintained from fuel boil-off.

At the initiation of J-2 engine start, GH_2 is bled from the J-2 engine at 750 psia, -260° Fahrenheit to provide ullage pressure during fuel depletion. The pressure bled from the engine into the fuel tank is controlled by a fuel tank pressurization control module.

Fuel Tank Repressurization System—During the coast period prior to engine restart, there is no requirement for fuel tank pressurization. Tank pressure will build up within the tank due to LH_2 boil-off which is vented continuously through a propulsive vent system designed to provide a minimum thrust requirement to assure propellant settling. Additional pressure is vented through the fuel tank vent-relief system.

Prior to J-2 engine restart, the propulsive vent system and tank vent-relief system is closed in the pressurization control module. The tank is then repressurized to between 31 to 34 psia with cold helium from the nine spheres located in the fuel tank. Simultaneously with LOX tank repressurization, the O_2/H_2 burner heats and expands the helium prior to its entry into the fuel tank. The flow is controlled by two parallel-redundant valves which are actuated by a fuel tank pressure switch.

Five ambient helium spheres located on the thrust structure provide redundancy to the O_2/H_2 burner repressurization system operation. Should the burner malfunction or other circumstances require another orbit after repressurization, the ambient system, controlled by the fuel repressurization control module, can repressurize the fuel tank a second time.

Following engine restart, the LH_2 tank is again pressurized throughout the second burn phase with GH_2 bled from the engine.

Fuel Tank Vent-Relief System—Venting of the LH_2 tank is accomplished by a vent and relief system capable of relieving all excess pressure accumulated from overpressurization or fuel boil-off during fill and flight operation. During fill, vaporization is vented through a self-sealing disconnect located in the forward skirt. During liftoff and flight, the gases are vented overboard through a nonpropulsive exhaust.

The venting system consists of an actuation control module, vent valve, and nonpropulsive overboard exhaust. Actuation of the vent valve is commanded from an external ground signal during fill operations and from the flight sequencer during liftoff and flight. The vent valve is designed to open in a maximum of 0.1 second upon command.

The relief valve, which provides a backup capability in case of vent valve failure, opens at 38 psia and reseats at 35 psia, and has a flow/relief capability of 2 pounds per second at sea level.

A directional control valve directs excessive pressures through the ground disconnect during fill and directs excessive pressures through the nonpropulsive vent during liftoff and flight. The nonpropulsive vent system extends from the directional control valve into two 4-inch vent lines that terminate into two nonpropulsive exhaust ports. The ports are located 180° apart in the forward skirt area. The ports are arranged to direct the exhaust for total thrust cancellation.

LH_2 Continuous Propulsive Vent System

The continuous vent system is used to provide a thrust force required to position propellants at the aft end of each tank during coast. The system consists of a vent line originating at the vent-relief valve, terminating at two low thrust nozzles located 180° apart, and facing aft on the forward skirt. Continuous venting is controlled and regulated by a pneumatically operated continuous propulsive vent module.

At the completion of the first burn engine cutoff, APS ullage engines are activated to settle the liquid propellants in the aft end of the tanks during the shutdown phase. LH_2 tank pressure is then vented through the continuous propulsive vent system, providing a continuous propulsive thrust to the stage. This maintains control of the propellants within the tanks. The APS engines are shut off after the transition is complete and the propulsive venting continues throughout the coast phase. The continuous propulsive vent module controls venting from a maximum of 45 pounds to a minimum of approximately 7 pounds.

LH_2 Feed System—Prior to vehicle liftoff and prior to engine restart, all LH_2 feed system components of the J-2 turbopump assembly must be chilled to assure proper operation. Chilldown of the LH_2 system is accomplished by a closed loop, forward-flow recirculation system. On command from the IU, the prevalve in the LH_2 feed duct closes and the chilldown shutoff valve opens. An auxiliary, electrically driven LH_2 chilldown pump, mounted in the LH_2 tank, circulates the LH_2 within the system and is capable of a minimum flowrate of 135 gpm at 6.1 psi.

LH_2 is circulated from the LH_2 tank through the low pressure feed duct, through the J-2 engine fuel pump, the fuel bleed valve, and back to the tank through a return line. Recirculation chilldown continues through the boost phase and up to J-2 engine ignition. In the event of an emergency shutdown requirement, the chilldown system shutoff valve is closed upon command from the IU. LH_2 is supplied to the J-2 engine through a vacuum-jacketed, low-pressure duct at a flowrate of 81 pounds per second at -423° Fahrenheit and 32 psia. The duct is located in the fuel tank side wall above the common bulkhead joint and is equipped with bellows to compensate for thermal motion. Signals from the engine sequencer energize the LH_2 feed valve, as required to obtain steady-state operation. A complete description of engine operation may be found in the J-2 Engine section.

DUAL REPRESSURIZATION SYSTEM

The dual repressurization system pressurizes the stage propellant tanks to flight conditions for restart of the J-2 engine. The first repressurization is accomplished by the O_2/H_2 burner system. The O_2/H_2 burner provides the heat to expand the cold helium used to pressurize the propellant tanks during the coast phase.

An ambient helium system provides mission backup capability as well as backup for the O_2/H_2 burner system.

The dual repressurization system consists of four major subsystems: the O_2/H_2 burner, the burner propellant feed systems, the O_2/H_2 burner repressurization system, and the ambient repressurization system.

O_2/H_2 Burner

The O_2/H_2 burner heats cold helium (-410°F) for use as the oxidizer and fuel tank pressurant. The burner utilizes LOX and LH_2 from the S-IVB mainstage propellant tanks. At existing tank pressures, the propellants are fed through vacuum-jacketed, low-pressure ducts to the burner. In the burner the propellants are vaporized as they pass through regenerative coils before being fed into the injector in a gaseous state and burned. Combustion products are exhausted through an exhaust nozzle using a 40:1 expansion ratio. A thrust of 16 to 30 pounds is obtained, which is directed approximately through the center of gravity of the vehicle.

O_2/H_2 Burner Propellant Feed System

The propellant feed system supplies and controls LH_2 and LOX flow from the mainstage propellant tank to the O_2/H_2 burner. Propellant flow is controlled by three valves: the LH_2 control valve, the LOX control valve, and the gaseous oxygen shutdown valve. The LH_2 control valve and the gaseous oxygen shutdown valve close simultaneously to preclude a high temperature transient at burner cutoff. The LOX control valve remains open at burner shutdown to preclude a pressure buildup in the LOX supply duct.

O_2/H_2 Burner Repressurization System

The O_2/H_2 burner repressurization system utilizes helium as the pressurant to implement repressurization of the LOX and LH_2 propellant tanks. Pressurant is supplied to the burner from nine storage spheres. The spheres are located in the LH_2 tank, and contain cold helium charged to 3,100 psia at -410° F. The pressure is reduced to 385±25 psig by the LOX tank pressure control module, and is controlled by two identical valve assemblies. These valve assemblies are located in each supply line, and contain dual redundant solenoid valves (normally closed) for controlling pressurant flow from the storage spheres to the burner inlets. The solenoid valves open and close on a signal from associated mainstage propellant tank pressure switches, which sense tank ullage pressures. The repressurization process is initiated approximately 7.5 minutes prior to second ignition.

Ambient Repressurization System

The ambient repressurization system operates late in the restart preparations sequence. If helium from the ambient system is required (due to O_2/H_2 burner malfunction, for make up gas, or for recycling due to command interrupt decision), it is supplied through an independent control module for each of the propellant tanks. Two 4.5-cubic foot spheres are provided for LOX tank repressurization, and five 4.5-cubic foot spheres are provided for LH_2 tank repressurization. The storage spheres and control modules are located on the thrust structure. The helium storage pressure is 3,000 psi.

PROPELLANT UTILIZATION SYSTEM

The primary function of the PU system is to assure simultaneous depletion of propellants by controlling the LOX flowrate to the J-2 engine. It also provides propellant mass information for controlling the fill and topping valves during propellant loading operations. The system consists of mass sensors, an electronics assembly, and an engine-mounted mixture ratio valve.

During loading operations, the mass of propellants loaded is determined within one per cent by the mass sensors. Tank over-fill sensors act as a backup system in the event the loading system fails to terminate fill operations.

Continuous LH_2 and LOX residual readout signals are provided throughout third stage powered flight. Differences between the fuel and oxidizer mass indications, as sensed by the mass sensors, are continually analyzed and are then used to control the oxidizer pump bypass flowrate, which changes the engine mixture ratio correspondingly. The static inverter/converter supplies the analog voltages necessary to operate the PU system. It is commanded "on" and "off" by a switch selector and sequencer combination.

PNEUMATIC CONTROL SYSTEM

The pneumatic control system provides GHe (gaseous helium) pressure to operate all third stage pneumatically operated valves with the exception of those provided as components of the J-2 engine. GHe is supplied from an ambient helium sphere and pressurized from a ground source before propellant fill operations at 3,100 ± 100 psia at 70° Fahrenheit for valve operation. The sphere is located on the thrust structure and is pre-conditioned to above 70° Fahrenheit from the environmental control system before liftoff.

The pneumatic control system provides regulated pressure at 475 ± 25 psig for operation of the LH_2 and LOX vent-relief valves during propellant loading, LH_2 directional control valve, LOX and LH_2 fill and drain valves during loading, and the GH_2 engine start system vent-relief valve. It also provides operating pressures for the LH_2 and LOX turbopump turbine purge module, LOX chilldown pump purge module control, LOX and LH_2 prevalves, LOX and LH_2 chilldown shutoff valves, the LH_2 continuous propulsive vent control module, and the O_2/H_2 burner propellant valves.

The pneumatic control subsystem is protected from overpressure by a normally open solenoid valve controlled by a downstream pressure-sensing switch. At pressures greater than 600 ± 15 psia, the pressure switch actuates and closes the valve. At pressures below 490 ± 15 psia, the pressure switch drops out and the solenoid opens, thus acting as a backup regulator.

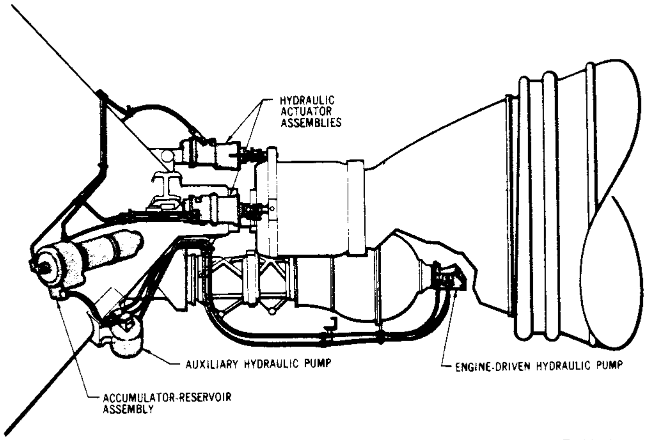
Flight Control System

The flight control system provides stage thrust vector steering and attitude control. Steering is achieved by gimbaling the J-2 engine during powered flight. Hydraulic actuator assemblies provide J-2 engine deflection rates proportional to steering signal corrections supplied by the IU.

Stage roll attitude during powered flight is controlled by firing the APS attitude control engines.

HYDRAULIC SYSTEM

The hydraulic system performs engine positioning upon command from the IU. Major components are a J-2 engine-driven hydraulic pump, two hydraulic actuator assemblies, and an accumulator-reservoir assembly.



D-NRV-10

J-2 Engine Hydraulic System Components

The electrically driven auxiliary hydraulic pump is started before vehicle liftoff to pressurize the hydraulic system. Electric power for the pump is provided by a ground source. At liftoff, the pump is switched to stage battery power. Pressurization of the hydraulic system restrains the J-2 engine in a null position with relation to the third stage centerline, preventing pendulum-like shifting from forces encountered during liftoff and boost. During powered flight, the J-2 engine may be gimbaled up to 7° in a square pattern by the hydraulic system upon command from the IU.

Engine-Driven Hydraulic Pump

The engine-driven hydraulic pump is a variable displacement type pump capable of delivering hydraulic fluid under continuous system pressure and varying volume as required for operation of the hydraulic actuator assemblies. The pump is driven directly from the engine oxidizer turbopump. A thermal isolator in the system controls hydraulic fluid temperature to ensure proper operation.

Auxiliary Hydraulic Pump

The auxiliary hydraulic pump is an electrically driven variable displacement pump which supplies a constant minimum supply of hydraulic fluid to the hydraulic system at all times. The pump is also used to perform preflight engine gimbaling checkouts, hydraulically lock the engine in the null position during boost phase, maintain system hydraulic fluid at operating temperatures during other than the powered phase, and augment the engine-driven hydraulic pump during powered flight. It also provides an emergency backup supply of fluid to the system.

Hydraulic Actuator Assemblies

Two hydraulic actuator assemblies are attached directly to the J-2 engine and the thrust structure and receive IU command signals to gimbal the engine. The actuator assemblies are identical and interchangeable.

Accumulator-Reservoir Assembly

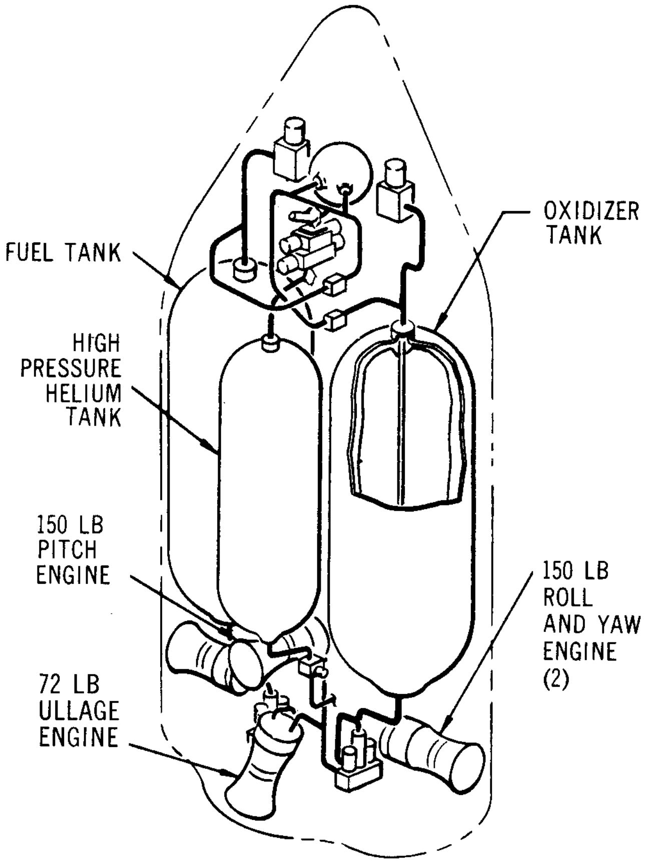
The accumulator-reservoir assembly is an integral unit mounted on the thrust structure. The reservoir section is the storage area for hydraulic fluid; the accumulator section supplies peak system fluid requirements and dampens high-pressure surges within the system.

AUXILIARY PROPULSION SYSTEM

The APS provides auxiliary propulsive thrust to the stage for three-axis attitude control and for ullage control. Two APS modules are mounted 180° apart on the aft skirt assembly. Two solid propellant rocket motors are mounted 180° apart between the APS modules on the aft skirt assembly and provide additional thrust for ullage control.

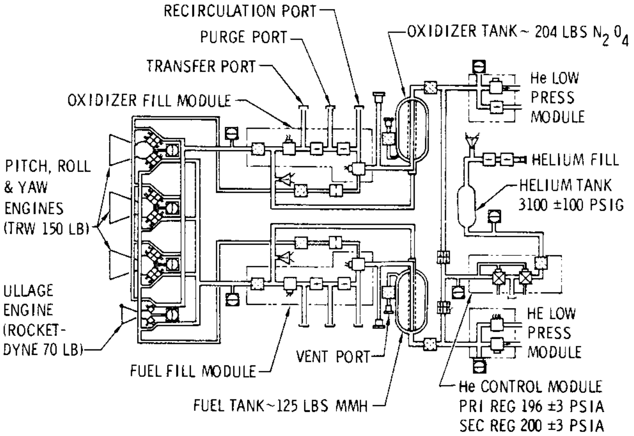
APS Modules

Each APS module contains three 150-pound-thrust attitude control engines and one 72-pound-thrust ullage control engine.



D-NRV-27

Auxiliary Propulsion System Module



APS Schematic

D-NRV-13A

The attitude control engines are fired upon command from the IU in short duration bursts for attitude control of the stage during the orbital coast phase of flight. Minimum engine-firing pulse-duration is approximately 70 milliseconds. The attitude control engines are approximately 15 inches long with exit cones approximately 6.5 inches in diameter. Engine cooling is accomplished by an ablative process.

The ullage control engines are fired also upon command from the IU during the transition between J-2 engine first burn and the coast phase of flight to prevent undesirable propellant movement within the tanks. Firing continues for approximately 50 seconds until activation of the LH_2 continuous propulsive vent system. The ullage engines are again fired at the end of the third stage coast phase of flight and prior to J-2 engine restart to assure proper propellant positioning at inlets to the propellant feed lines during propellant tank repressurization.

The ullage control engines are similar to the attitude control engines and are approximately 15 inches long with an exit cone approximately 5.75 inches in diameter. Engine cooling is accomplished by an ablative process.

Each APS module contains an oxidizer system, fuel system, and pressurization system. The modules are self-contained and easily detached for separate checkout and environmental testing.

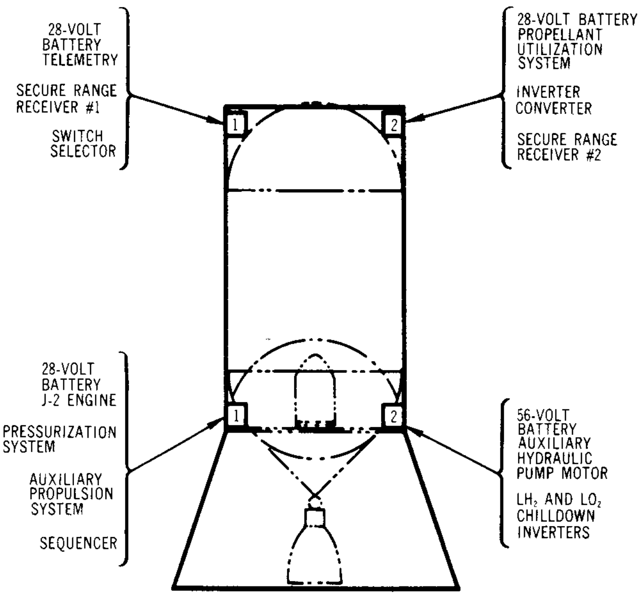
Separate fuel and oxidizer tanks of the bladder type are mounted within the APS module along with a high-pressure helium tank, which provides pressurization for both the propellant tanks and the associated plumbing and control systems.

An ignition system is unnecessary because fuel and oxidizer are hypergolic (self-igniting). Nitrogen tetroxide (N_2O_4), the oxidizer, is stable at room temperature.

The fuel, monomethyl hydrazine ($CH_3N_2H_3$), is stable to shock and extreme heat or cold. The APS module carries approximately 125 pounds of usable fuel and about 178 pounds of usable oxidizer.

Ullage Control

Two solid propellant Thiokol TX-280 rocket motors, each rated at 3,420 pounds of thrust, are ignited during separation of the second and third stages for ullage control approximately 4 seconds before J-2 ignition. This thrust produces additional positive stage acceleration during separation and positions LOX and LH_2 propellants toward the aft end of the tanks. In addition, propellant boil-off vapors are forced to the forward end where they are safely vented overboard. Tank outlets are covered to ensure a net positive suction head (NPSH) to the propellant pumps, thus preventing possible pump cavitation during J-2 engine start. Ullage rockets ignite upon command from the stage sequencer and fire for approximately 4 seconds. At about 12 seconds from ignition, the complete rocket motor assemblies, including bracketry, are jettisoned from the stage upon command from the stage sequencer.



D-NRV-15

Third Stage Basic Electrical Power and Distribution System

Electrical Power and Distribution System

Four battery-powered systems provide electrical requirements for third stage operation. Forward Power System No. 1 includes a 28 VDC battery and power distribution equipment for telemetry, secure range receiver No. 1, forward battery heaters, and a power switch selector located in the aft skirt area.

Forward Power System No. 2 includes a 28 VDC battery and power distribution equipment for the PU assembly, inverter-converter, and secure range receiver No. 2.

Aft Power System No. 1 includes a 28 VDC battery and power distribution equipment for the J-2 engine, pressurization systems, APS modules, TM signal power, aft battery heaters, hydraulic system and stage sequencer.

Aft Power System No. 2 includes a 56 VDC battery and power distribution equipment for the auxiliary hydraulic pump, oxidizer chilldown inverter, and fuel chilldown inverter.

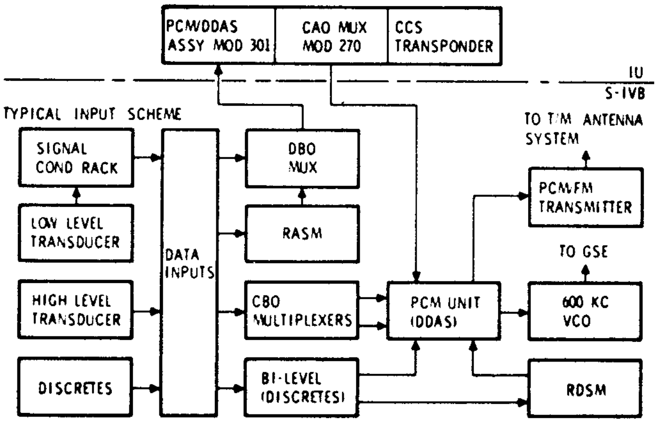
Silver-oxide, zinc batteries used for electrical power and distribution systems are manually activated. The batteries are "one-shot" units, and not interchangeable due to different load requirements.

Electrical power and distribution systems are switched from ground power to the batteries by command through the aft umbilical prior to liftoff.

Telemetry and Instrumentation System

A radio frequency telemetry system is used for transmission of stage instrumentation information to ground receiving stations. One transmitter, using two separate antenna systems, is capable of returning information during third stage flight. The telemetry transmission link consists of one basic modulation scheme: a Pulse Code Modulated/FM (PCM/FM).

A Digital Data Acquisition System (DDAS) airborne tape recorder stores sampled data normally lost during staging and over-the-horizon periods of orbital missions, and plays back information when in range of ground stations.



D-ORM-167A

Basic PCM Digital Data Acquisition System

The PCM/FM system (DDAS) is used during automatic checkout to provide data for the ground checkout computer. The system is also used to provide precise information concerning stage environment and performance of systems during flight.

Environmental Control Systems

AFT SKIRT AND INTERSTAGE THERMOCONDITIONING AND PURGE

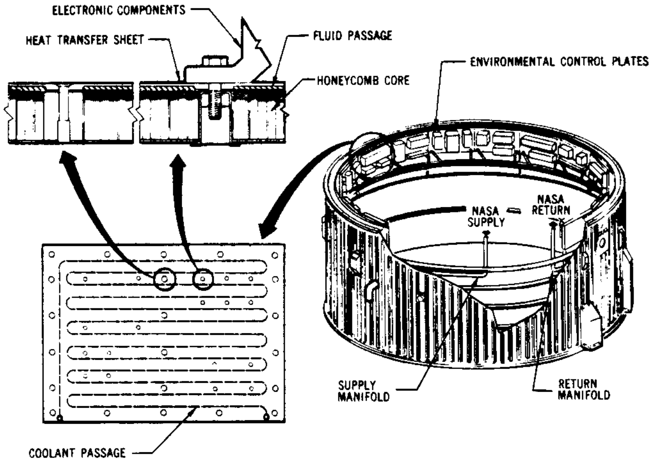
The thermoconditioning and purge system purges the aft skirt and aft interstage of combustible gases and distributes temperature controlled air or gaseous nitrogen around electrical equipment in the aft skirt during the vehicle countdown.

The purging gas, supplied from a ground source through the umbilical, passes over the electrical equipment and flows into the aft interstage area. Some of the gas is directed through each of the auxiliary propulsion modules and exhausts into the interstage. A duct from the skirt manifold directs air or GN_2 to a thrust structure manifold. From the thrust structure manifold supply duct, a portion of air or GN_2 is directed to a shroud covering the hydraulic accumulator reservoir.

Temperature control is accomplished by two dual-element thermistor assemblies located in the gaseous exhaust stream of each of the auxiliary propulsion modules. Elements are wired in series to sense average temperature. Two series circuits are formed, each circuit utilizing one element from each thermistor assembly. One series is used for temperature control, the other for temperature recording.

FORWARD SKIRT THERMOCONDITIONING

Electrical equipment in the third stage forward skirt area is thermally conditioned by a heat transfer system, using "cold plates" on which electronic components are mounted, and through which coolant fluid circulates. Coolant is pumped through the system from the IU and returned. Heat from electrical equipment attached to the cold plates is dissipated by conduction through the mounting feet and the cold plates to the fluid. Refer to the Instrument Unit section for a complete description of the IU environmental conditioning system.



D-NRV-19

Forward Skirt Environmental Control System

FORWARD SKIRT AREA PURGE

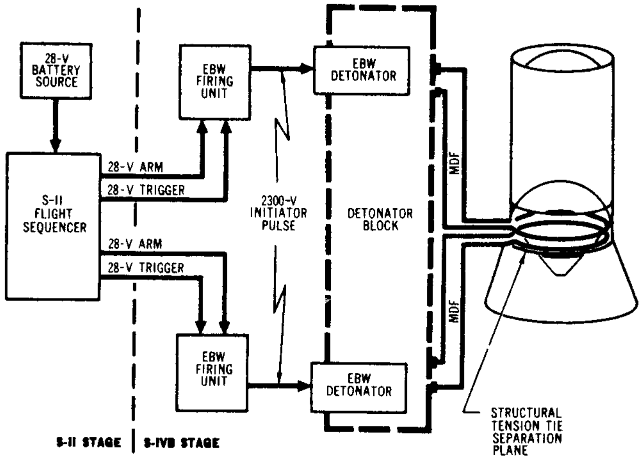
The forward skirt area is purged with gaseous nitrogen to minimize fire and explosion hazards while propellants are loaded or stored in the stage. Gaseous nitrogen is supplied and remotely controlled from a ground source.

Ordnance Systems

The ordnance systems perform stage separation, retrorocket ignition, ullage control rocket ignition and jettison, and range safety functions.

STAGE SEPARATION SYSTEM

The stage separation system consists of a severable tension strap, mild detonating fuse (MDF), exploding bridgewire, (EBW), detonators and EBW firing units.



D-NRV-2

Separation System

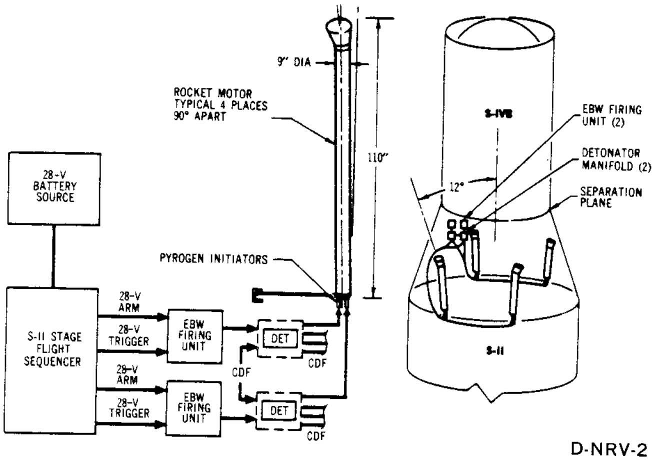
The severable tension strap houses two redundant MDF cords in a "V" groove circumventing the stage between the aft skirt and aft interstage at the separation plane. Ignition of the MDF cords is triggered by a signal from the second stage sequencer through the EBW and EBW firing units about 3 seconds after second stage engine cutoff.

The MDF consists of a flexible metal sheath surrounding a continuous core of high explosive material. Once detonated, the explosive force of the MDF occurs at a rate of 23,000 feet per second.

The EBW detonator is fired to initiate the MDF explosive train. A 2,300 VDC pulse is applied to a small resistance wire and a spark gap. The high voltage electrical arc across the spark gap ignites a charge of high explosive material which in turn detonates the MDF. The high voltage pulse requirement for ignition renders this system safe from random ground or vehicle electrical power. Upon command, each EBW firing unit supplies high voltage and current required to fire a specific EBW detonator.

RETROROCKET IGNITION SYSTEM

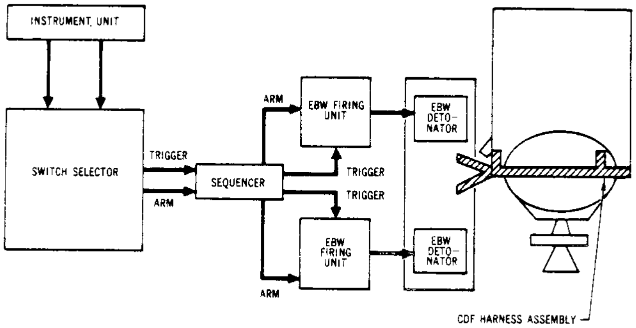
Four solid propellant retrorockets are mounted equidistant around the aft interstage assembly, and when ignited, assure clean separation of the third stage from the second stage by decelerating or braking the spent booster. Each retrorocket is rated for a nominal thrust of 35,000 pounds, weight of 384 pounds, and burn time of about 1.5 seconds.



D-NRV-2

Retrorocket System

A signal from the second stage initiates two EBW firing units located on the aft interstage. The EBW firing units ignite two detonator manifolds, which in turn ignite the retrorockets through redundant pairs of confined detonating fuse (CDF) and pyrogen initiators.



D-NRV-22

Ullage Rocket System

ULLAGE CONTROL ROCKET IGNITION AND JETTISON SYSTEM

Two solid propellant ullage rockets, located on the third stage aft skirt just forward of the stage separation plane, are ignited on signal from the stage sequencer by EBW initiators.

After firing, the burned-out ullage rocket casings and fairings are jettisoned to reduce stage weight. Upon command from the stage sequencer, two forward and aft frangible nuts, which secure each rocket motor and its fairing to the stage, are detonated by confined detonating fuse (CDF), to free the entire assembly from the vehicle.

RANGE SAFETY SYSTEM

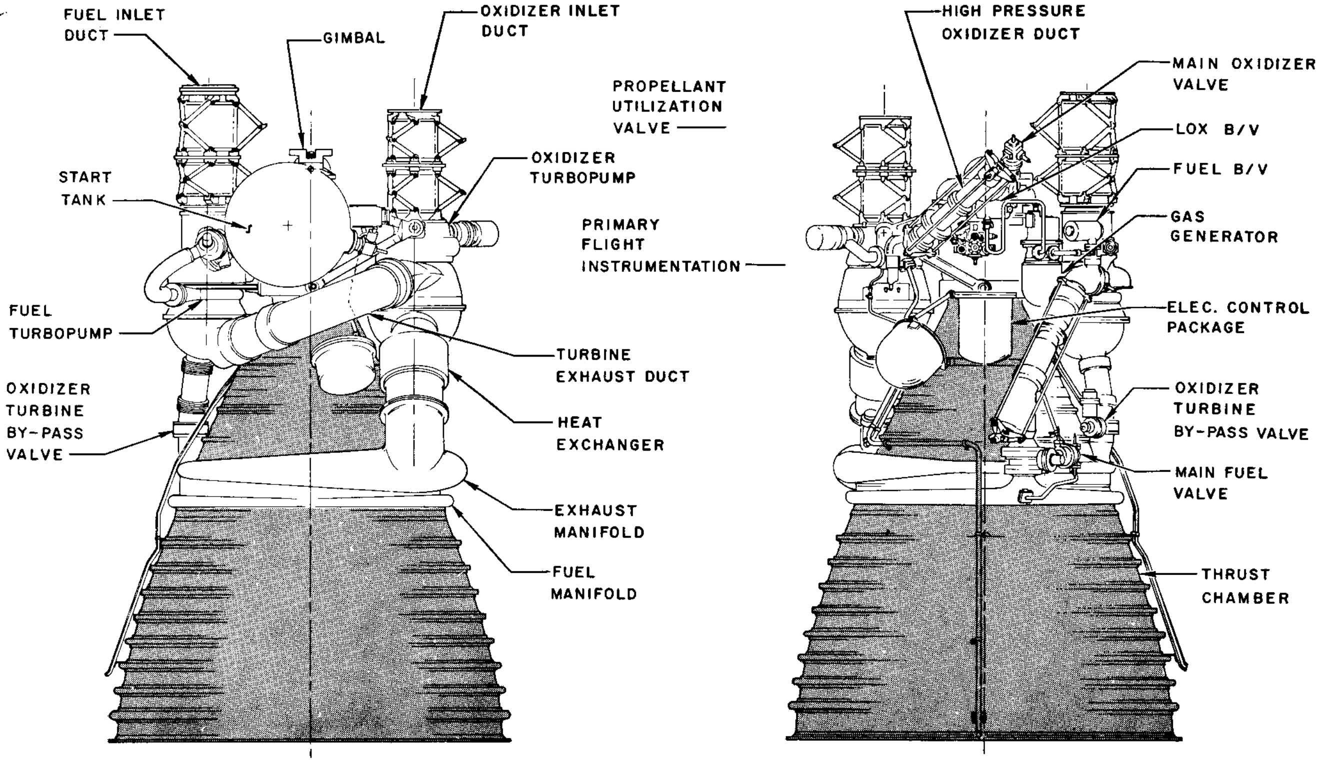
The range safety system terminates vehicle flight upon command of the range safety officer. Redundant systems are used throughout to provide maximum reliability.

Four antennas, mounted around the periphery of the third stage forward skirt assembly, feed two redundant secure range receivers located in the forward skirt assembly. Both receivers have separate power supplies and circuits. A unique combination of coded signals must be transmitted, received, and decoded to energize this destruct system.

A safety and arming device prevents inadvertent initiation of the explosive train by providing a positive isolation of the EBW detonator and explosive train until arming is commanded. Visual and remote indications of SAFE and ARMED conditions are displayed at all times at the firing center. Upon proper command, EBW firing units activate EBW detonators.

A CDF, detonated by the safety and arming device, explodes a flexible linear-shaped charge which cuts through the tank skin to disperse both fuel and oxidizer.

J-2 ENGINE FACT SHEET



R-16

LENGTH	11 ft. 1 in.
WIDTH	6 ft. 8½ in.
NOZZLE EXIT DIAMETER	6 ft. 5 in.
THRUST (altitude)	230,000 lb. (vehicle 503 2nd stage 225,000 lb.)
SPECIFIC IMPULSE	424 sec. (427 at 5:1 mixture ratio)
RATED RUN DURATION	500 sec.
FLOWRATE: Oxidizer	449 lb/sec (2,847 gpm)
Fuel	81.7 lb/sec (8,365 gpm)
MIXTURE RATIO	5.5:1 oxidizer to fuel
CHAMBER PRESSURE (Pc)	763 psia
WEIGHT, DRY, FLIGHT CONFIGURATION	3,480 lb.
EXPANSION AREA RATIO	27.5:1
COMBUSTION TEMPERATURE	5,750°F

J-2 ENGINE

J-2 ENGINE DESCRIPTION

The Rocketdyne J-2 engine is a high performance, upper stage, propulsion system utilizing liquid hydrogen and liquid oxygen propellants and developes a maximum vacuum thrust of 225,000 pounds.

All J-2 engines are identical when delivered and may be allocated to either the second or third stage. Each engine is equipped to be restarted in flight. However, the restart capability will be utilized only in the third stage.

The single J-2 engine used in the third stage is gimbal-mounted so that it can be moved in flight and used to steer the stage. Five J-2 engines are arranged in a cluster in the second stage. The four outboard engines of the five-engine cluster are gimbal-mounted to provide the vehicle with pitch, yaw, and roll control. The center engine is mounted in a fixed position.

Major systems of the J-2 engine include a thrust chamber and gimbal assembly system, propellant feed system, gas generator and exhaust system, electrical and pneumatic control system, start tank assembly system, and flight instrumentation system.

Thrust Chamber and Gimbal System

The J-2 engine thrust chamber serves as a mount for all engine components. It is composed of the following subassemblies: thrust chamber body, injector and dome assembly, gimbal bearing assembly, and augmented spark igniter.

Thrust is transmitted through the gimbal mounted on the thrust chamber assembly dome to the vehicle thrust frame structure. The thrust chamber injector receives the propellants from a dual turbopump system (oxidizer and fuel) under pressure, mixes the propellants, and burns them to impart a high velocity to the expelled combustion gases to produce thrust.

THRUST CHAMBER

The thrust chamber is constructed of stainless steel tubes of 0.012-inch wall thickness. Tubes with thin walls are required for heat transfer purposes. The thrust chamber tubes are stacked longitudinally and furnace-brazed to form a single unit. The chamber is bell-shaped with a 27.5 to 1 expansion area ratio for efficient operation at altitude, and is regeneratively cooled by the fuel. Fuel enters from a manifold located midway between the thrust chamber throat and the exit at a pressure of more than 1,000 psi. In cooling the chamber the fuel makes a one-half pass downward through 180 tubes and is returned in a full pass up to the thrust chamber injector through 360 tubes. (See schematic drawing.)

DOME

The injector and oxidizer dome assembly is located at the top of the thrust chamber. The dome provides a manifold for the distribution of the liquid oxygen to the injector and serves as a mount for the gimbal bearing and the augmented spark igniter.

THRUST CHAMBER INJECTOR

The thrust chamber injector atomizes and mixes the propellants in a manner to produce the most efficient combustion. Six hundred and fourteen hollow oxidizer posts are machined to form an integral part of the injector. Fuel nozzles are threaded and installed over the oxidizer posts forming concentric orifices.

The injector face is porous and is formed from layers of stainless steel wire mesh and is welded at its periphery to the injector body. Each fuel nozzle is swaged to the face of the injector.

The injector receives liquid oxygen through the dome manifold and injects it through the oxidizer posts into the combustion area of the thrust chamber.

The fuel is received from the upper fuel manifold in the thrust chamber and injected through the fuel orifices which are concentric with the oxidizer orifices. The propellants are injected uniformly to ensure satisfactory combustion.

GIMBAL

The gimbal is a compact, highly loaded (20,000 psi) universal joint consisting of a spherical, socket-type bearing with a Teflon/fiberglass composition coating that provides a dry, low-friction bearing surface. It also includes a lateral adjustment device for aligning the chamber with the vehicle.

The gimbal transmits the thrust from the injector assembly to the vehicle thrust structure and provides a pivot bearing for deflection of the thrust vector, thus providing flight attitude control of the vehicle. The gimbal is mounted on the top of the injector and oxidizer dome assembly.


R-10

J-2 Assembly—Hydrogen fueled J-2 rocket engines for upper stages of Saturn V vehicles are completed on this assembly line. J-2 develops a maximum thrust of 225,000 pounds.

AUGMENTED SPARK IGNITER

The augmented spark igniter (ASI) is mounted to the injector face. It provides the flame to ignite the propellants in the thrust chamber. When engine start is initiated, the spark exciters energize two spark plugs mounted in the side of the igniter chamber. Simultaneously, the control system starts the initial flow of oxidizer and fuel to the spark igniter. As the oxidizer and fuel enter the combustion chamber of the ASI, they mix and are ignited.

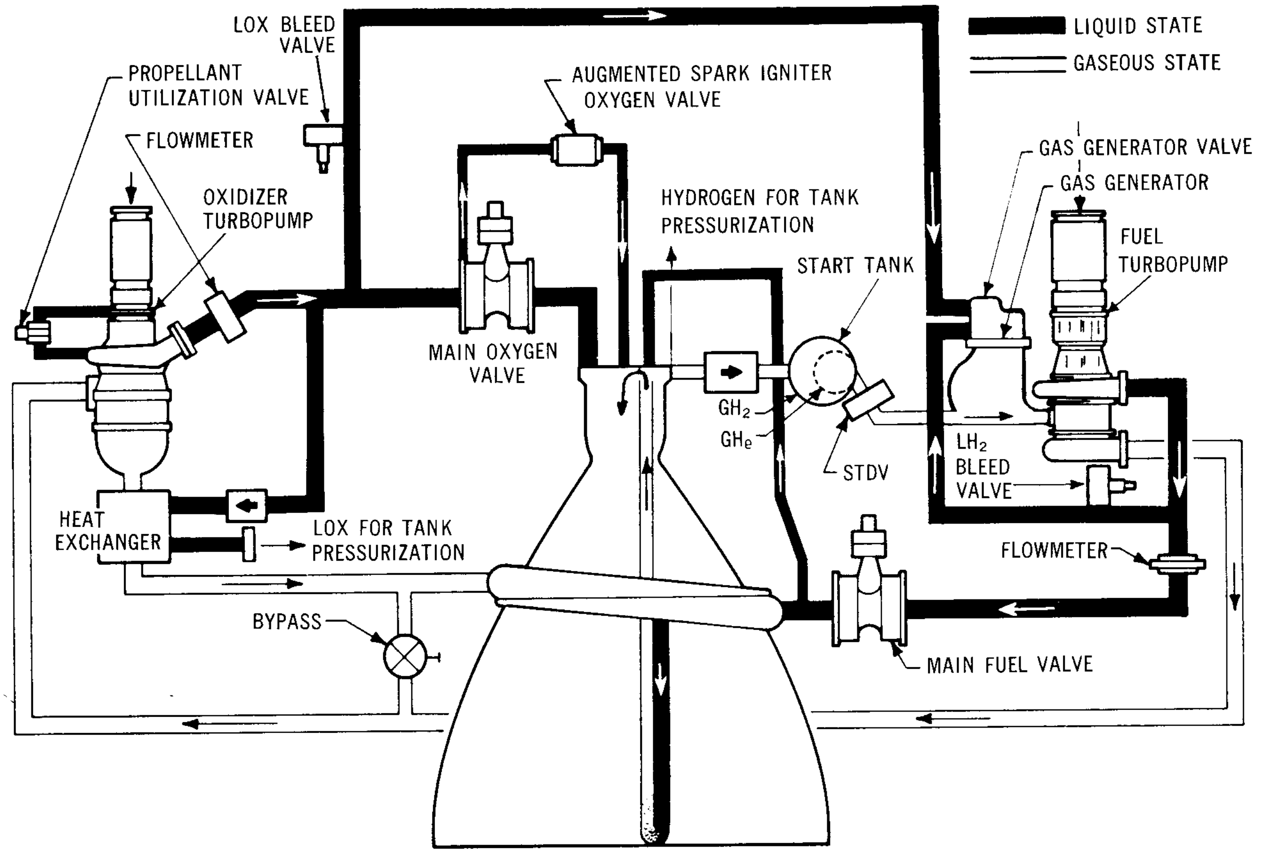
Mounted in the ASI is an ignition monitor which indicates that proper ignition has taken place. The ASI operates continuously during entire engine firing, is uncooled, and is capable of multiple reignitions under all environmental conditions.

Propellant Feed System

The propellant feed system consists of separate fuel and oxidizer turbopumps, main fuel valve, main oxidizer valve, propellant utilization valve, fuel and oxidizer flowmeters, fuel and oxidizer bleed valves, and interconnecting lines.

FUEL TURBOPUMP

The fuel turbopump, mounted on the thrust cham-



R-8

Basic J-2 Engine Schematic

ber, is a turbine-driven, axial flow pumping unit consisting of an inducer, a seven-stage rotor, and a stator assembly. It is a high-speed pump operating at 27,000 rpm, and is designed to increase hydrogen pressure from 30 psia to 1,225 psia through high-pressure ducting at a flowrate which develops 7,800 brake horsepower.

Power for operating the turbopump is provided by a high-speed, two-stage turbine. Hot gas from the gas generator is routed to the turbine inlet manifold which distributes the gas to the inlet nozzles where it is expanded and directed at a high velocity into the first stage turbine wheel.

After passing through the first stage turbine wheel, the gas is redirected through a ring of stator blades and enters the second stage turbine wheel. The gas leaves the turbine through the exhaust ducting. Three dynamic seals in series prevent the pump fluid and turbine gas from mixing. Power from the turbine is transmitted to the pump by means of a one-piece shaft.

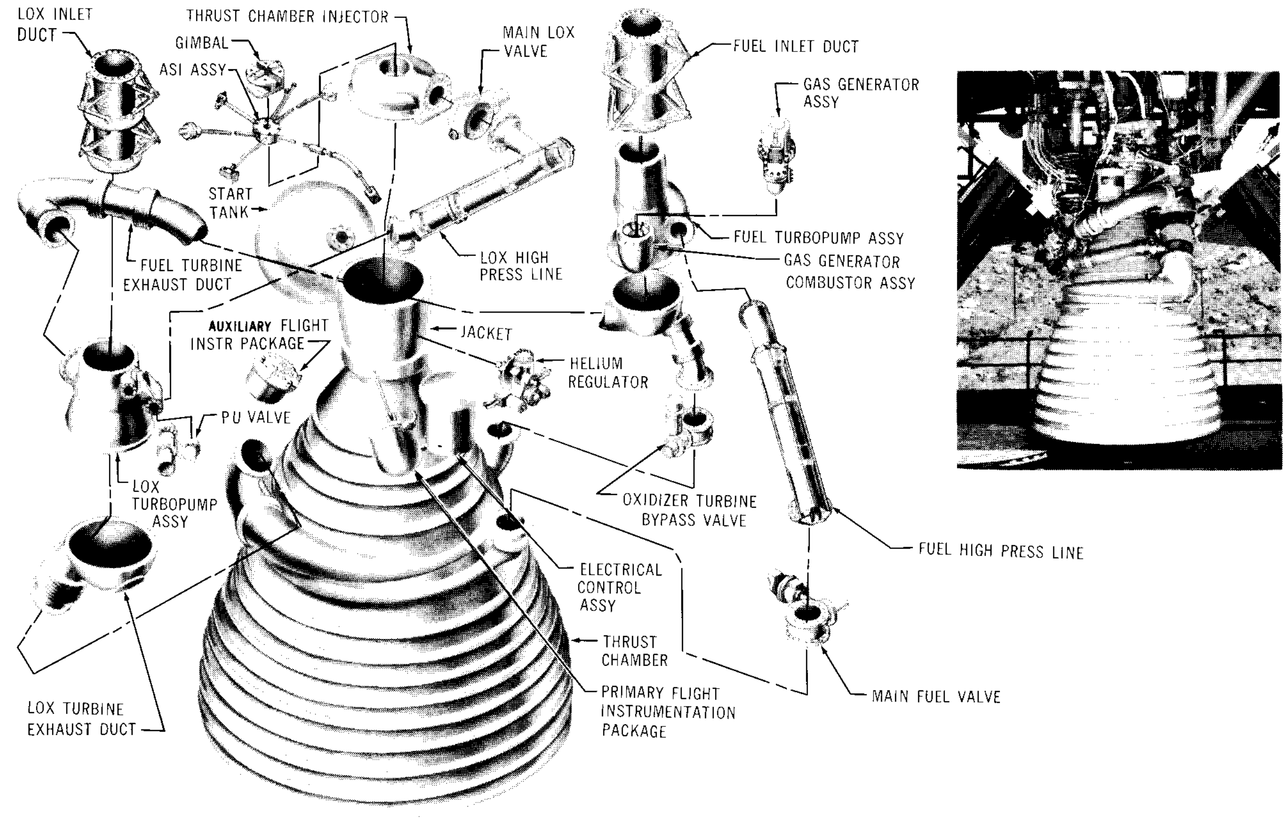
OXIDIZER TURBOPUMP

The oxidizer turbopump is mounted on the thrust chamber diametrically opposite the fuel turbopump. It is a single-stage centrifugal pump with direct turbine drive. The oxidizer turbopump increases the pressure of the liquid oxygen and pumps it through high-pressure ducts to the thrust chamber.

The pump operates at 8,600 rpm at a discharge pressure of 1,080 psia and develops 2,200 brake horsepower. The pump and its two turbine wheels are mounted on a common shaft.

Power for operating the oxidizer turbopump is provided by a high-speed, two-stage turbine which is driven by the exhaust gases from the gas generator. The turbines of the oxidizer and fuel turbopumps are connected in a series by exhaust ducting that directs the discharged exhaust gas from the fuel turbopump turbine to the inlet of the oxidizer turbopump turbine manifold. One static and two dynamic seals in series prevent the turbopump oxidizer fluid and turbine gas from mixing.

Beginning the turbopump operation, hot gas enters the nozzles and, in turn, the first stage turbine wheel. After passing through the first stage turbine wheel, the gas is redirected by the stator blades and enters the second stage turbine wheel. The gas then leaves the turbine through exhaust ducting, passes through the heat exchanger, and exhausts into the thrust chamber through a manifold directly



J-2 Major Component Breakdown

above the fuel inlet manifold. Power from the turbine is transmitted by means of a one-piece shaft to the pump. The velocity of the liquid oxygen is increased through the inducer and impeller. As the liquid oxygen enters the outlet volute, velocity is converted to pressure and the liquid oxygen is discharged into the outlet duct at high pressure.

Bearings in the liquid hydrogen and liquid oxygen turbopumps are lubricated by the fluid being pumped because the extremely low operating temperature of the engine precludes use of lubricants or other fluids.

MAIN FUEL VALVE

The main fuel valve is a butterfly-type valve, spring-loaded to the closed position, pneumatically operated to the open position, and pneumatically assisted to the closed position. It is mounted between the fuel high-pressure duct from the fuel turbopump and the fuel inlet manifold of the thrust chamber assembly. The main fuel valve controls the flow of fuel to the thrust chamber. Pressure from the ignition stage control valve on the pneumatic control package opens the valve during engine start. As the gate starts to open, it allows fuel to flow to the fuel inlet manifold.

MAIN OXIDIZER VALVE

The main oxidizer valve (MOV) is a butterfly-type valve, spring-loaded to the closed position, pneumatically operated to the open position, and pneumatically assisted to the closed position. It is mounted between the oxidizer high-pressure duct from the oxidizer turbopump and the oxidizer inlet on the thrust chamber assembly.

Pneumatic pressure from the normally closed port of the mainstage control solenoid valve is routed to both the first and second stage opening actuators of the main oxidizer valve. Application of opening pressure in this manner, together with controlled venting of the main oxidizer valve closing pressure through a thermal-compensating orifice, provides a controlled ramp opening of the main oxidizer valve through all temperature ranges. A sequence valve, located within the MOV assembly, supplies pneumatic pressure to the opening control part of the gas generator control valve and through an orifice to the closing part of the oxidizer turbine bypass valve.

PROPELLANT UTILIZATION VALVE

The propellant utilization (PU) valve is an electrically operated, two-phase, motor-driven, oxidizer transfer valve and is located at the oxidizer turbopump outlet volute. The propellant utilization valve ensures the simultaneous exhaustion of the contents of the propellant tanks. During engine operation, propellant level sensing devices in the vehicle propellant tanks control the valve gate position for adjusting the oxidizer flow to ensure simultaneous exhaustion of fuel and oxidizer.

An additional function of the PU valve is to provide thrust variations in order to maximize payload. The second stage, for example, operates with the PU valve in the closed position for more than 70 per cent of the firing duration. This valve position provides 225,000 pounds of thrust at a 5.5:1 propellant (oxidizer to fuel by weight) mixture ratio. During the latter portion of the flight, the PU valve position is varied to provide simultaneous emptying of the propellant tanks.

The third stage also operates at the high-thrust level for the majority of the burning time in order to realize the high thrust benefits.

The exact period of time at which the engine will operate with the PU valve closed will vary with individual mission requirements and propellant tanking levels.

When the PU valve is fully open, the mixture ratio is 4.5:1 and the thrust level is 175,000 pounds.

The propellant utilization valve and its servomotor are supplied with the engine. A position feedback potentiometer is also supplied as a part of the PU valve assembly. The PU valve assembly and a stage or a facility-mounted control system make up the propellant utilization system.

FUEL AND OXIDIZER FLOWMETERS

The fuel and oxidizer flowmeters are helical-vaned, rotor-type flowmeters. They are located in the fuel and oxidizer high-pressure ducts. The flowmeters measure propellant flowrates in the high-pressure propellant ducts. The four-vane rotor in the hydrogen system produces four electrical impulses per revolution and turns approximately 3,700 revolutions per minute at nominal flow. The six-vane rotor in the liquid oxygen system produces six electrical impulses per revolution and turns at approximately 2,600 revolutions per minute at nominal flow.

PROPELLANT BLEED VALVES

The propellant bleed valves used in both the fuel and oxidizer systems are poppet-type which are spring-loaded to the normally open position and pressure-actuated to the closed position. Both propellant bleed valves are mounted to the bootstrap lines adjacent to their respective turbopump discharge flanges.

The valves allow propellant to circulate in the propellant feed system lines to achieve proper operating temperature prior to engine start. The bleed valves are engine controlled. At engine start, a helium control solenoid valve in the pneumatic control package is energized allowing pneumatic pressure to close the bleed valves, which remain closed during engine operation.

Gas Generator and Exhaust System

This system consists of the gas generator, gas generator control valve, turbine exhaust system and exhaust manifold, heat exchanger, and oxidizer turbine bypass valve.

GAS GENERATOR

The gas generator is welded to the fuel pump turbine manifold, making it an integral part of the fuel turbopump assembly. It produces hot gases to drive the fuel and oxidizer turbines and consists of a combustor containing two spark plugs, a control valve containing fuel and oxidizer ports, and an injector assembly.

When engine start is initiated, the spark exciters in the electrical control package are energized, providing energy to the spark plugs in the gas generator combustor. Propellants flow through the control valve to the injector assembly and into the combustor outlet and are directed to the fuel turbine and then to the oxidizer turbine.

GAS GENERATOR CONTROL VALVE

The gas generator control valve is a pneumatically operated poppet-type that is spring-loaded to the closed position. The fuel and oxidizer poppets are mechanically linked by an actuator. The gas generator control valve controls the flow of propellants through the gas generator injector.

When the mainstage signal is received, pneumatic pressure is applied against the gas generator control valve actuator assembly which moves the piston and opens the fuel poppet. During the fuel poppet opening, an actuator contacts the piston that opens the oxidizer poppet. As the opening pneumatic pressure decays, spring loads close the poppets.

TURBINE EXHAUST SYSTEM

The turbine exhaust ducting and turbine exhaust hoods are of welded sheet metal construction. Flanges utilizing dual (Naflex) seals are used at component connections. The exhaust ducting conducts turbine exhaust gases to the thrust chamber exhaust manifold which encircles the thrust chamber approximately halfway between the throat and the nozzle exit. Exhaust gases pass through the heat exchanger and exhaust into the main thrust chamber through 180 triangular openings between the tubes of the thrust chamber.

HEAT EXCHANGER

The heat exchanger is a shell assembly, consisting of a duct, bellows, flanges, and coils. It is mounted in the turbine exhaust duct between the oxidizer turbine discharge manifold and the thrust chamber. It heats and expands helium gas for use in the third stage or converts liquid oxygen to gaseous oxygen for the second stage for maintaining vehicle oxidizer tank pressurization. During engine operation, either liquid oxygen is tapped off the oxidizer high-pressure duct or helium is provided from the vehicle stage and routed to the heat exchanger coils.

OXIDIZER TURBINE BYPASS VALVE

The oxidizer turbine bypass valve is a normally open, spring-loaded, gate type. It is mounted in the oxidizer turbine bypass duct. The valve gate is equipped with a nozzle, the size of which is determined during engine calibration. The valve in its open position depresses the speed of the oxygen pump during start, and in its closed position acts as a calibration device for the turbopump performance balance.

Control System

The control system includes a pneumatic system and a solid-state electrical sequence controller packaged with spark exciters for the gas generator and the thrust chamber spark plugs, plus interconnecting electrical cabling and pneumatic lines.

PNEUMATIC SYSTEM

The pneumatic system consists of a high-pressure helium gas storage tank, a regulator to reduce the pressure to a usable level, and electrical solenoid control valves to direct the central gas to the various pneumatically controlled valves.

ELECTRICAL SEQUENCE CONTROLLER

The electrical sequence controller is a completely self-contained, solid-state system, requiring only DC power and start and stop command signals.

Pre-start status of all critical engine control functions is monitored in order to provide an "engine ready" signal. Upon obtaining "engine ready" and "start" signals, solenoid control valves are energized in a precisely timed sequence as described in the "Engine Operation" section to bring the engine through ignition, transition, and into mainstage operation. After shutdown, the system automatically resets for a subsequent restart.

Start Tank Assembly System

This system is made up of an integral helium and hydrogen start tank, which contains the hydrogen and helium gases for starting and operating the engine. The gaseous hydrogen imparts initial spin to the turbines and pumps prior to gas generator combustion, and the helium is used in the control system to sequence the engine valves.

HELIUM AND HYDROGEN TANKS

The spherical helium tank is positioned inside the hydrogen tank to minimize engine complexity. It holds 1,000 cubic inches of helium. The larger spherical hydrogen gas tank has a capacity of 7,257.6 cubic inches. Both tanks are filled from a ground source prior to launch and the gaseous hydrogen tank is refilled during engine operation from the thrust chamber fuel inlet manifold for subsequent restart in third stage application.

Flight Instrumentation System

The flight instrumentation system is composed of a primary instrumentation package and an auxiliary package.

PRIMARY PACKAGE

The primary package instrumentation measures those parameters critical to all engine static firings and subsequent vehicle launches. These include some 70 parameters such as pressures, temperatures, flows, speeds, and valve positions for the engine components, with the capability of transmitting signals to a ground recording system or a telemetry system, or both. The instrumentation system is designed for use throughout the life of the engine, from the first static acceptance firing to its ultimate vehicle flight.

AUXILIARY PACKAGE

The auxiliary package is designed for use during early vehicle flights. It may be deleted from the basic engine instrumentation system after the propulsion system has established its reliability during research and development vehicle flights. It contains sufficient flexibility to provide for deletion, substitution, or addition of parameters deemed necessary as a result of additional testing. Eventual deletion of the auxiliary package will not interfere with the measurement capability of the primary package.

Engine Operation

START SEQUENCE

Start sequence is initiated by supplying energy to two spark plugs in the gas generator and two in the augmented spark igniter for ignition of the propellants. Next, two solenoid valves are actuated: one for helium control, and one for ignition phase control. Helium is routed to hold the propellant bleed valves closed and to purge the thrust chamber LOX dome, the LOX pump intermediate seal, and the gas generator oxidizer passage. In addition, the main fuel valve and ASI oxidizer valve are opened, creating an ignition flame in the ASI chamber that passes through the center of the thrust chamber injector.

After a delay of 1, 3, or 8 seconds, during which time fuel is circulated through the thrust chamber to condition the engine for start, the start tank discharge valve is opened to initiate turbine spin. The length of the fuel lead is dependent upon the length of the Saturn V first stage boost phase. When the J-2 engine is used in the second stage of the Saturn V vehicle, a one-second fuel lead is necessary. The third stage of the Saturn V vehicle, on the other hand, utilizes a three-second fuel lead for its initial start and an eight-second fuel lead for its restart.

After an interval of 0.450 second, the start tank discharge valve is closed and a mainstage control solenoid is actuated to: 1) turn off gas generator and thrust chamber helium purges; 2) open the gas generator control valve (hot gases from the gas generator now drive the pump turbines); 3) open the main oxidizer valve to the first position (14 degrees) allowing LOX to flow to the LOX dome to burn with the fuel that has been circulating through the injector; 4) close the oxidizer turbine bypass valve (a portion of the gases for driving the oxidizer turbopump were bypassed during the ignition phase); 5) gradually bleed the pressure from the closing side of the oxidizer valve pneumatic actuator controlling the slow opening of this valve for smooth transition into mainstage. Energy in the spark plugs is cut off and the engine is operating at rated thrust. During the initial phase of engine operation, the gaseous hydrogen start tank will be recharged in those engines having a restart requirement. The hydrogen tank is repressurized by tapping off a controlled mixture of liquid hydrogen from the thrust chamber fuel inlet manifold and warmer hydrogen from the thrust chamber fuel injection manifold just before entering the injector.

FLIGHT MAINSTAGE OPERATION

During mainstage operation, engine thrust may be varied between 175,000 and 225,000 pounds by actuating the propellant utilization valve to increase or decrease oxidizer flow as described in the section "PU Valve". This is beneficial to flight trajectories and for overall mission performance to make greater payloads possible.

CUTOFF SEQUENCE

When the engine cutoff signal is received by the electrical control package, it de-energizes the mainstage and ignition phase solenoid valves and energizes the helium control solenoid de-energizer timer. This, in turn, permits closing pressure to the main fuel valve, main oxidizer valve, gas generator control valve, and augmented spark igniter valve. The oxidizer turbine bypass valve and propellant bleed valves open and the gas generator and LOX dome purges are initiated.

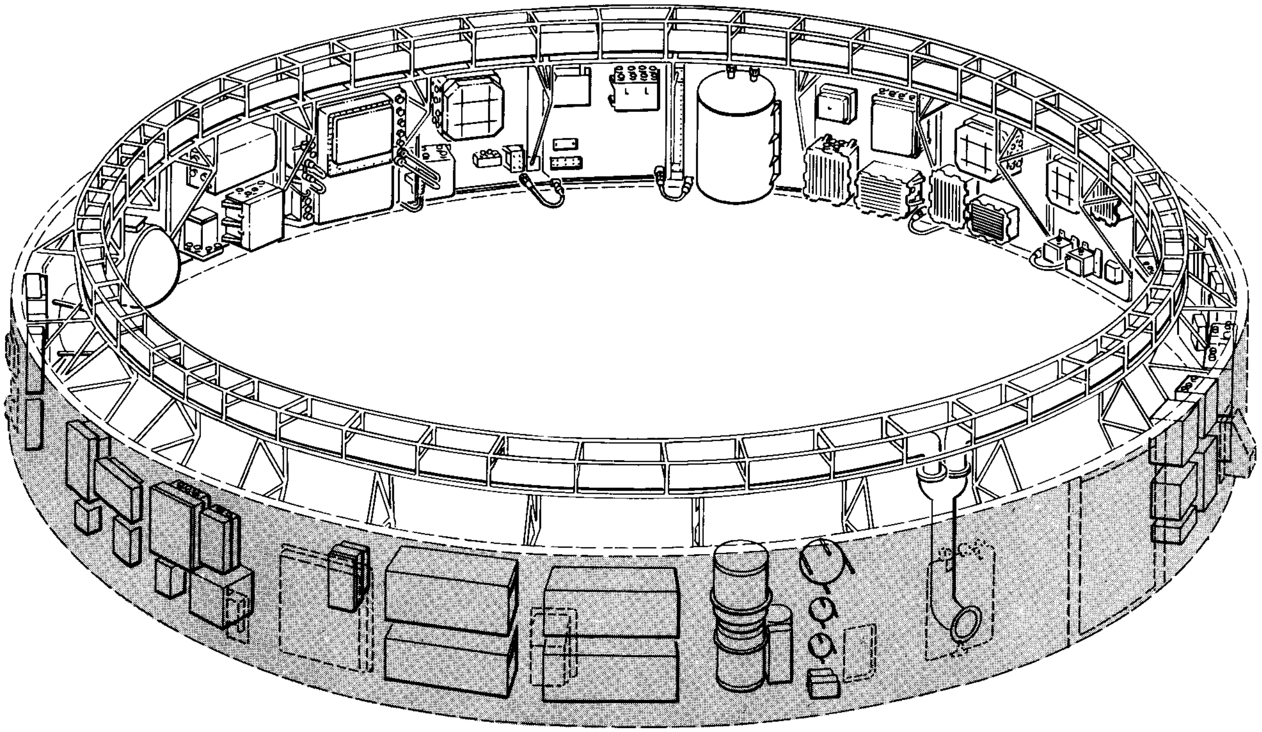
ENGINE RESTART

To provide third stage restart capability for the Saturn V, the J-2 gaseous hydrogen start tank is refilled in 60 seconds during the previous firing after the engine has reached steady-state operation. (Refill of the gaseous helium tank is not required because the original ground-fill supply is sufficient for three starts.) Prior to engine restart, the stage ullage rockets are fired to settle the propellants in the stage propellant tanks, ensuring a liquid head to the turbopump inlets.

Also, the engine propellant bleed valves are opened, the stage recirculation valve is opened, the stage prevalve is closed, and a LOX and LH_2 circulation is effected through the engine bleed system for five minutes to condition the engine to the proper temperature to ensure proper engine operation.

Engine restart is initiated after the "engine ready" signal is received from the stage. This is similar to the initial "engine ready". The hold time between cutoff and restart is from a minimum of 1-1/2 hours to a maximum of 6 hours, depending upon the number of earth orbits required to attain the lunar window for translunar trajectory.

INSTRUMENT UNIT FACT SHEET



IBM-DR-27

DIAMETER: 260 in.
HEIGHT: 36 in.
WEIGHT: 4,500 lb. (Average)

MAJOR SYSTEMS

ENVIRONMENTAL CONTROL SYSTEM: Provides cooling for electronic modules and components within the IU and forward compartments of third stage

GUIDANCE AND CONTROL SYSTEM: Determines course of Saturn V through space and adapts that course to fulfill mission requirements

INSTRUMENTATION SYSTEM: Measures vehicle conditions and reactions during mission and transmits this information to ground for subsequent analysis, as well as providing for ground station-to-vehicle communication

ELECTRICAL SYSTEM: Provides basic operating power for all electronic and electrical equipment in the IU; also monitors vehicle performance and may initiate automatic mission abort if an emergency arises

STRUCTURAL SYSTEM: Serves as a load bearing part of the launch vehicle, supporting both the components within the IU and the spacecraft; composed of three 120-degree segments of thin-wall aluminum alloy face sheets bonded over a core of aluminum honeycomb about an inch thick

INSTRUMENT UNIT

INSTRUMENT UNIT DESCRIPTION

The instrument unit (IU) for Saturn V was designed by NASA at MSFC and was developed from the Saturn I IU. Overall responsibility for the IU has been assigned to IBM's Federal Systems Division for fabrication and assembly of the unit, system testing, and integration and checkout of the unit with the launch vehicle. IBM also assembles and delivers computer programming necessary to support the IU. These programs are used:

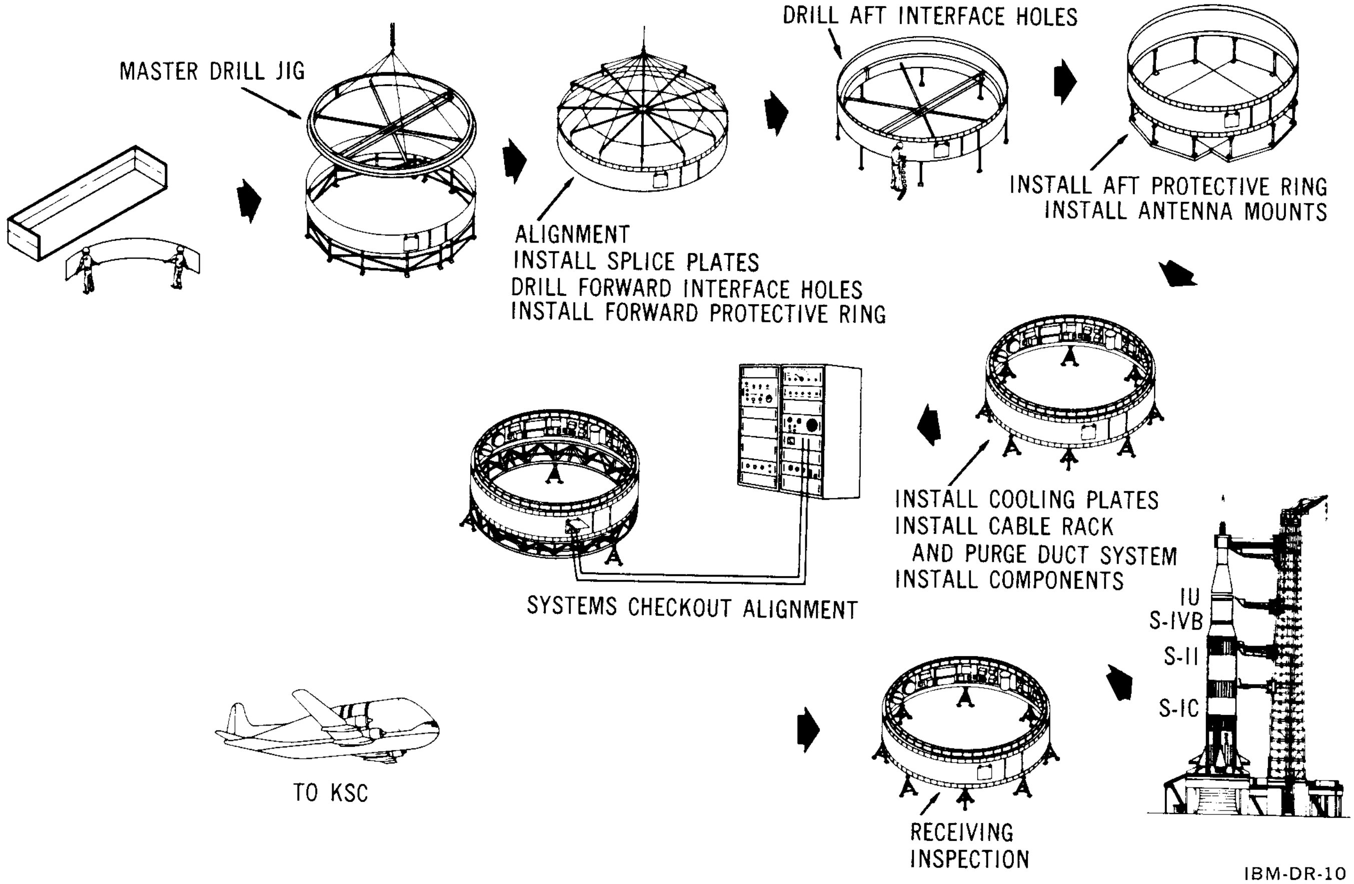
1. In IBM's automated systems checkout computer complex in Huntsville. This system verifies IU system integrity prior to release of the IU to NASA.
2. In IBM's simulation laboratory in Huntsville to verify the flight readiness of the IU's launch vehicle digital computer program, as well as the passive filters contained in the IU's analog flight control computer.
3. To operate the automated launch computer complex at John F. Kennedy Space Center. This computer complex is used to automatically checkout the flight readiness of the vehicle prior to liftoff.
4. To operate the IU's launch vehicle digital computer in flight, as well as programming that will be used by NASA, for postflight analysis of vehicle environment and performance data.

The IU is Saturn V's nerve center. It contains the electronic and electrical equipment needed for guidance, tracking, and origination and communication of vehicle environmental and performance data. The IU also contains environmental control equipment for temperature control, batteries, and power supplies to furnish operating power for electronic equipment.

The stage structure is 260 inches in diameter and 36 inches high and becomes a load-bearing part of the vehicle. It supports the components within the IU and the weight of the spacecraft.

INSTRUMENT UNIT FABRICATION

The structure is manufactured in three, 120-degree segments, each consisting of thin-wall alumi-

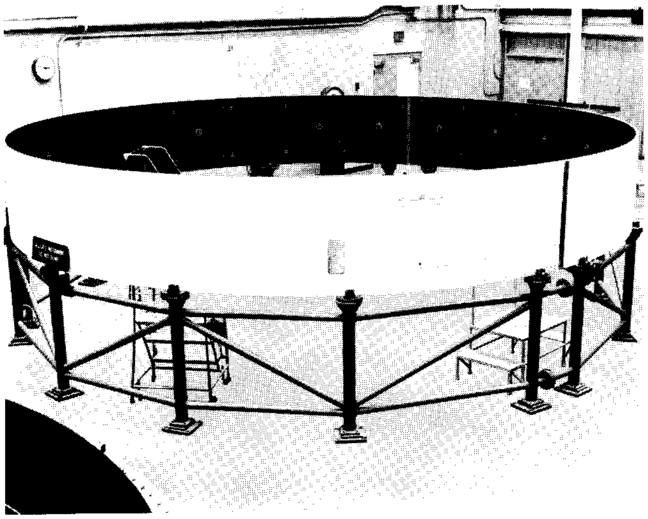


IU Production Sequence

num honeycomb. An aluminum alloy channel ring, bonded to the top and bottom edge of each segment, provides mating surfaces between the IU, the third stage, and the payload adapter. Mounted, inner skin brackets provide attachment points for the environmental control system's cold plates or for cold plate installation.

Segments are aligned and joined by splice plates bolted both inside and outside the joints. A spring-loaded umbilical door provides access to electrical connections between IU equipment and ground test areas. A larger access door, bolted in place, permits personnel to enter the IU after vehicle mating.

Assembly of an IU begins when the three curved structural segments, three feet high by 14 feet long, arrive at IBM's Huntsville, Ala., facility. Each segment weighs only 140 pounds.



IBM-DR-22

Structure Segments—Prior to splicing, mounting brackets for thermal conditioning panels can be seen on interior surface of segments. The exterior of the spring-loaded umbilical door and the access door are visible at right center.

Extremely accurate theodolites, similar to a surveyor's transit, are used to align the segments in a circle prior to splicing. Metal splicing plates join the three segments, and the holes which permit the IU to be joined to mating surfaces of the launch vehicle are drilled at top and bottom edges of the structure for ease in handling. Protective rings are bolted to these edges to stiffen the structure. Vehicle antenna holes are cut after splices are bolted.

After structure fabrication is completed, module and component assembly operations begin. Temperature transducers are fastened to the inner skin, environmental control system (ECS) cold plates are mounted, and a cable tray is bolted to the top of the structure. Components are mounted on the cold plates and ECS system pumps, storage tanks (called accumulators), heat exchangers, and plumbing are installed. Two nitrogen supply systems are installed: one for gas bearings of the inertial platform and the other for pressurization of the ECS. Finally, ducts, tubing, and electrical cables complete the assembly and the IU now weighing in excess of 4,000 pounds is ready for a long series of tests.

INSTRUMENT UNIT SYSTEMS

Environmental Control System

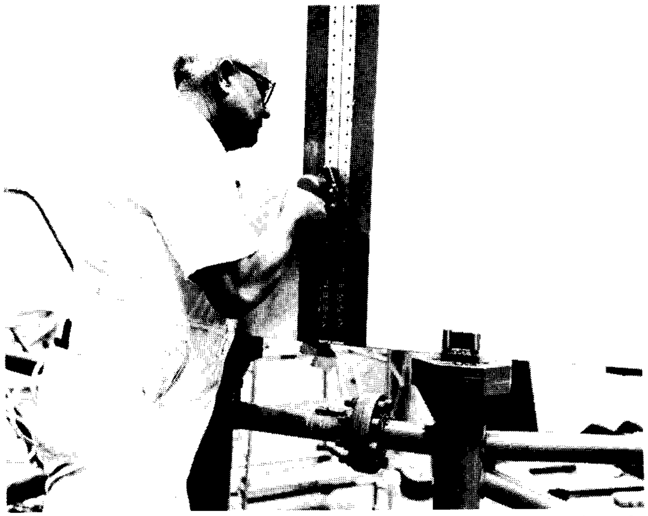
The ECS cools the electronic equipment in the IU and the forward third stage skirt. Sixteen cold plates are installed in each stage.

An antifreeze-like coolant, 60 per cent methanol and 40 per cent water, from a reservoir within the IU is circulated through the cold plates. Heat generated by the mounted components is transferred to the coolant by means of conduction.

Prior to liftoff a preflight heat exchanger serviced by ground support equipment transfers heat from the coolant. Approximately 163 seconds after liftoff, ECS's sublimator-heat exchanger takes over the job of temperature control.

Some of the more complex components like the guidance computer, flight control computer, and the ST-124-M platform, have coolant fluid circulated through them to provide more efficient heat removal.

In the vacuum of space the warmed coolant, after leaving the cold plates, is routed through a device called a sublimator. Water, from an IU reservoir,

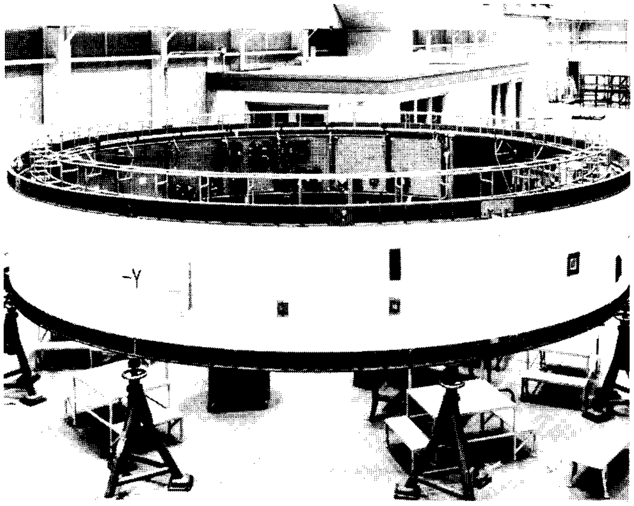


IBM-DR-16

Splice Joint Operations—Final grinding of a splice joint ensures a smooth surface prior to splice plate assembly.

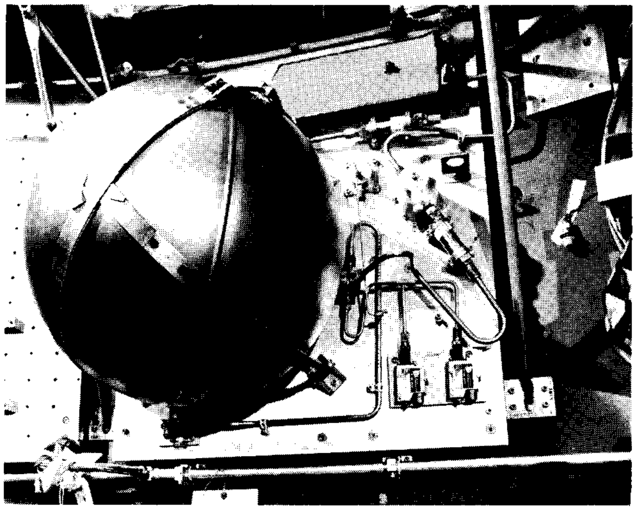
goes to the sublimator and is exposed through a porous plate to the low temperature and pressure of outer space where it freezes, blocking the pores in the plate. The heat from the coolant, transferred to the plate, is absorbed by the ice converting it directly into water vapor (a process called sublimation).

The system is self-regulating. The rate of heat dissipation varies with the amount of heat input, speeding up or slowing down as heat is generated. If the coolant temperature falls below a pre-set level, an electronically controlled valve causes the coolant mixture to bypass the sublimator until the temperature rises sufficiently to require further cooling.



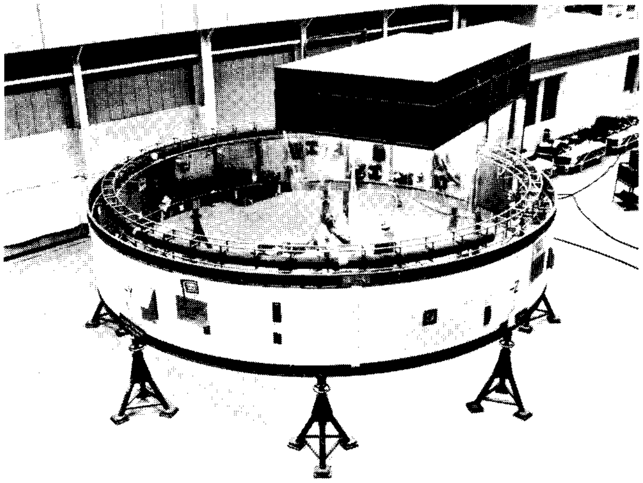
IBM-DR-19

Instrument Unit Assembly in IBM Manufacturing Area—Splicing operations and assembly of the tubular cable tray are complete, the cold plates have been installed, and installation of components is underway.



IBM-DR-21

GN_2 Storage Sphere—In place next to the ST-124-M inertial platform, the sphere holds 2 cubic feet of gas used for gas bearings of the platform. Also visible are a pressure regulator, heat exchanger for warming gas, and pressure indicators.



IBM-DR-23

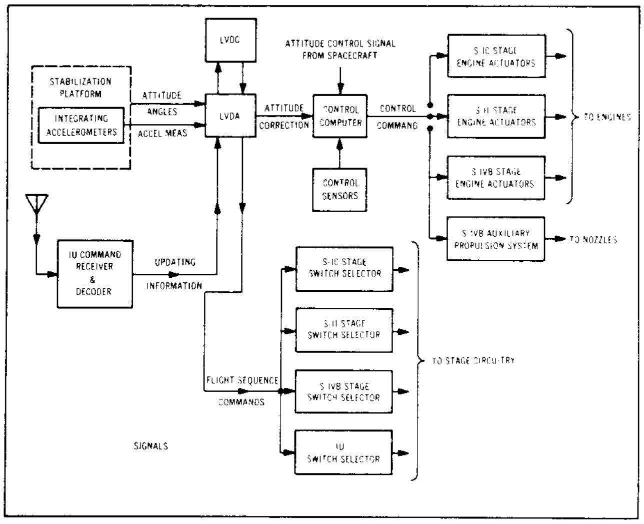
Environment Control—A mobile clean room protects against contamination during assembly of environmental control system components. Gaseous nitrogen will be circulated from a ground supply through the duct partially assembled in the cable tray to purge the IU following vehicle fueling.

Nitrogen gas provides artificial pressure for both coolant solution and sublimator water reservoirs during orbit.

A coolant circulating pump along with the necessary valves and piping to control flow complete the environmental control equipment.

Guidance and Control

The IU's guidance and flight control systems nav-

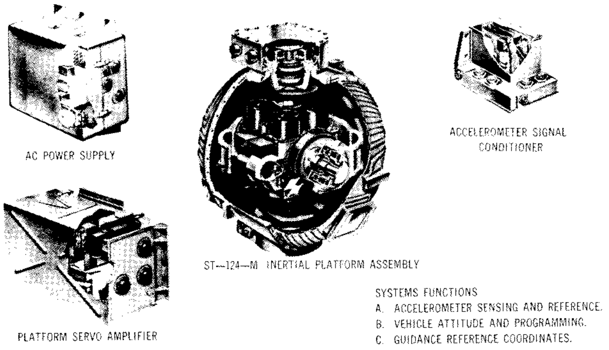


IBM-DR-8

Block Diagram of Guidance and Control System igate (determine vehicle position and velocity), guide (determine attitude correction signals), and control (determine and issue control commands to the engine actuators) the Saturn V vehicle.

Completely self-contained, these systems measure acceleration and vehicle attitude, determine velocity and position and their effect on the mission, calculate attitude correction signals, and determine and issue control commands to the engine actuators. All this is done to place the vehicle in a desired attitude to reach the required velocity and altitude for mission completion.

Major components are an inertial platform, the launch vehicle digital computer (LVDC), the launch vehicle data adapter (LVDA), an analog flight control computer, and control and rate gyros.



IBM-DR-4

ST-124-M Inertial Platform System

Prior to liftoff, launch parameters go to the LVDC.

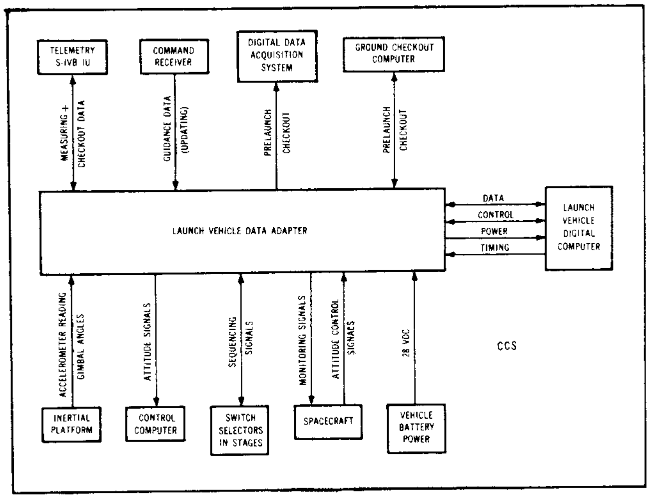
About five seconds before liftoff, the inertial guidance platform and the LVDC are released from ground control. As the vehicle ascends, the guidance platform senses and measures vehicle acceleration and attitude and sends these measurements to the LVDC via the LVDA.

The LVDC integrates these measurements with the time since launch to determine vehicle position relative to starting point and destination. It then computes the desired vehicle attitude, using data stored in its memory, and the difference between the desired attitude and the actual becomes the generated attitude correction signal.

This signal is sent to the analog flight control computer, where it is combined with information from rate gyros. Using this data, the flight control computer determines and issues the command to gimbal the engines and change the thrust direction.

Each mission has at least three phases: atmospheric-powered flight, boost period after initial entry into space, and the coasting period.

Atmospheric boost causes the greatest vehicle load because of atmospheric pressure. During this time the guidance system is primarily checking vehicle integrity and is programmed to minimize this pressure.



IBM-DR-6

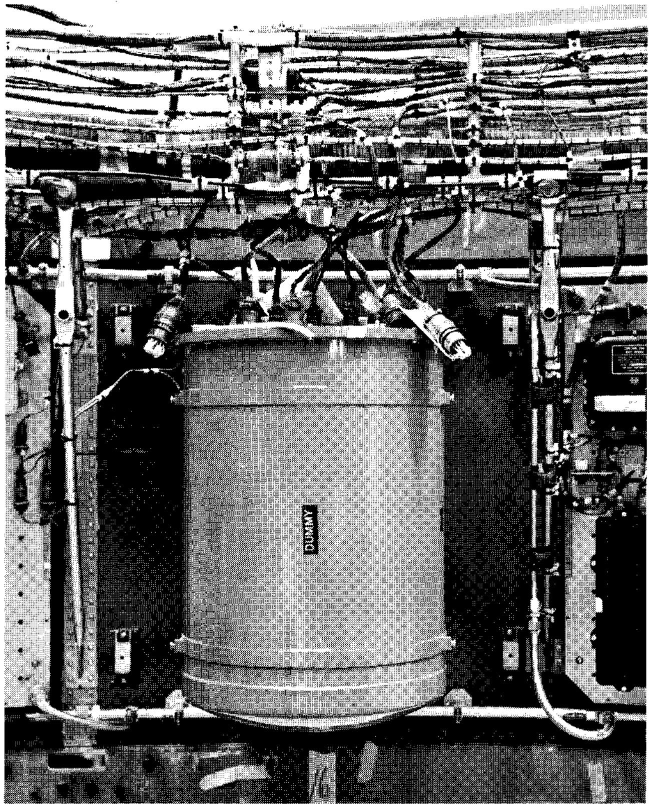
Guidance and Control—The LVDC and LVDA portion of the guidance system is shown in this block diagram. The LVDC receives information from all parts of the vehicle via the LVDA, and in turn issues commands.

The vehicle maintains liftoff orientation long enough to clear the launch equipment, and then it performs a roll maneuver to get to the flight azimuth direction.

The time tilt program is applied after the roll maneuver. The pitch angle is regulated by the tilt program, and is independent of navigation measurements. However, navigation measurements and computations are performed throughout the flight, beginning at the time the platform is released (i.e., five seconds before liftoff). First stage engine cutoff and stage separation are commanded when the IU receives a signal that the tank's fuel level has reached a predetermined point. During second stage powered flight the LVDC guides the vehicle via the best path to reach the mission objectives.

During orbit, navigation and guidance information in the LVDC can be updated by data transmission from ground stations through the IU radio command system.

Approximately once every two seconds, the LVDC, using iterative or "closed loop" guidance, figures vehicle position and vehicle conditions required at the end of powered flight (velocity, altitude, etc.) and generates the attitude correction signals to gimbal the engines so that the vehicle reaches its predetermined parking orbit.



IBM-DR-17

IU Interior During Assembly—The large, cylindrical component simulates size and shape of the flight control computer and is used to check cable lengths and mounting arrangement.

Second stage engine cutoff comes when the IU is signaled that stage propellant has reached a predetermined level, and then the stage is separated. By this time, the vehicle has already reached its approximate orbital altitude, and the third stage burn merely gives it enough push to reach a circular parking orbit.

TRIPLE RELIABILITY

To ensure the accuracy and reliability of guidance information, critical LVDC circuits are provided in triplicate. Known as triple modular redundancy (TMR), the system corrects for failure or inaccuracy by providing three identical circuits. Each circuit produces an output which is voted upon. In case of a discrepancy, the majority rules, and a random failure or error can be ignored. In addition, the LVDC has a duplexed memory, and if an error is found in one portion of the memory, the required output is obtained from the other and correct information read back into both memories, thus correcting the error.

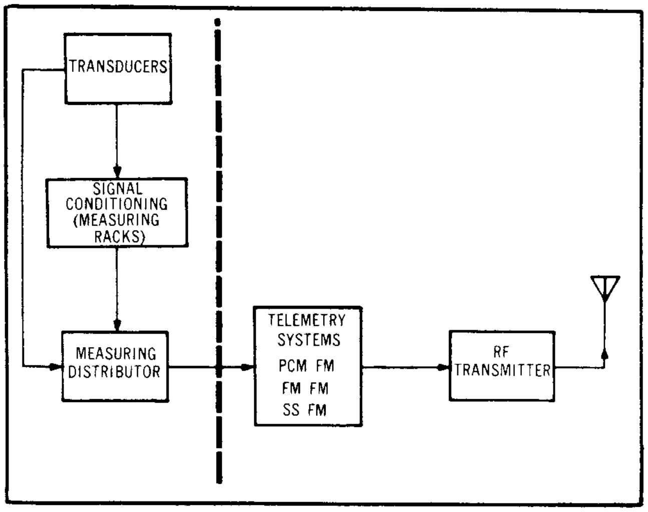
The ST-124-M inertial platform provides signals representing vehicle attitude. Since a signal error could produce vast changes in ultimate position, component friction must be minimized. Therefore, the platform bearings are floated in a thin film of dry nitrogen supplied at a controlled pressure and flowrate from reservoirs within the IU.

PRELAUNCH FUNCTIONS

In addition to guidance computations, other functions are performed by the LVDC and the LVDA. During prelaunch, the units conduct test programs. After liftoff they direct engine ignition and cutoff, direct stage separations, and conduct reasonableness tests of vehicle performance. During earth orbit, the computer directs attitude control, conducts tests, isolates malfunctions, and controls transmission of data, plus the sequencing of all events.

Instrumentation

A basic requirement for vehicle performance analysis and for planning future missions is knowing what happened during all phases of flight and just how the vehicle reacted. The IU's measuring and telemetry equipment reports these facts. Measuring sensors or transducers are located throughout the vehicle monitoring environment and systems' performance.



IBM-DR-1

Measuring and Telemetry

Measurements are made of mechanical movements, atmospheric pressures, sound levels, temperatures, and vibrations and are transformed into electrical signals. Measurements also are made of electrical signals, such as voltage, currents, and frequencies which are used to determine sequence of stage separation, engine cutoff, and other flight events and to determine performance of onboard equipment.

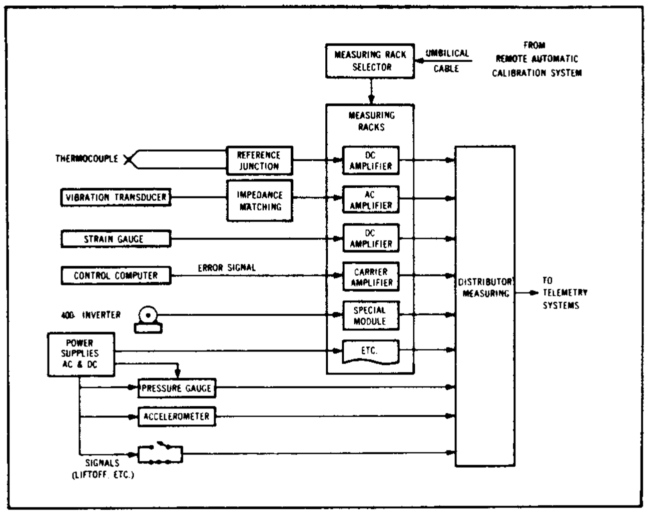
In all, the IU makes several hundred measurements. A wide variety of sensors are used to obtain all kinds of information required: acoustic transducers monitor sound levels; resistor or thermistor transducers monitor temperature environments; bourdon-tube or bellows transducers measure pressures; force-balance, or piezoelectric accelerometers measure force levels at critical points; flow meters determine rates of fluid flow.

Various measuring devices produce a variety of outputs, and before these outputs can be effectively utilized, they must be standardized to some extent. Signal conditioning modules are employed to adapt transducer outputs to a uniform range of 0-5 VDC.

Different types of data require different modes of transmission, and the telemetry portion of the system provides three such modes: SS/FM, FM/FM, and PCM/FM. Each type of information is routed to the most suitable telemetry equipment; a routing is performed by the measuring racks within the IU.

To get the most out of the transmission equipment, multiplexing is employed on some telemetry channels. Information originated by various measuring devices is repeatedly sampled by multiplexers, or commutators, and successive samples from different sources are transmitted to earth.

Information sent over any channel represents a series of measurements made at different vehicle points. This time-sharing permits large chunks of data to be handled with a minimum amount of equipment. The LVDC also helps in data transmission. For instance, when the vehicle is between ground receiving stations, the LVDC stores important PCM data for later transmission. Once the vehicle leaves the earth's atmosphere, sound levels requiring air for continuance no longer exist. The LVDC signals a measuring distributor to switch from unimportant measurements to those more critical to the mission. And during stage separation retro-rocket firing, when flame attenuation distorts or destroys telemetry transmissions, signals are automatically recorded by an onboard tape recorder, and transmitted later.



IBM-DR-2

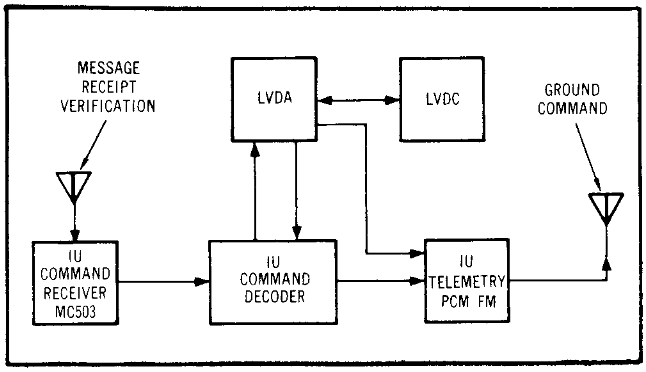
Typical Saturn Measuring System

In order to monitor vehicle performance, ground controllers must know the vehicle's precise position at all times. The RF section of the instrumentation system provides this capability, as well as linking the IU's guidance and control equipment during flight.

TRACKING SYSTEM

Several tracking systems are used to follow vehicle trajectory during ascent and orbit. Consolidation of this data not only increases data reliability, but gives the best trajectory information.

Vehicle antennas and transponders, which increase ground-base tracking systems' range and accuracy, make up the IU's tracking equipment.



IBM-DR-7

Saturn V Instrument Unit Command System

A pulse or series of pulses of RF energy sent by ground stations to the vehicle's general direction will interrogate the airborne transponder. In response, the transponder produces a pulse or series of pulses. Triangulation between precisely located ground stations determines point of origin of these reply pulses and fixes location of the vehicle.

Three tracking systems are employed in conjunction with the Saturn V IU: AZUSA, C-band radar, and the S-band portion of the command and communication system (CCS). Two C-band transponders are employed to provide tracking capabilities for this system independent of vehicle attitude. A single transponder is employed with the AZUSA system.

Real-time navigation, needed to update the guidance system, is received in the IU by a radio command link. But before it is sent, and before it is accepted in the IU, both ground equipment and IU instruments scrutinize update information for accuracy. The slightest error in transmission could conceivably produce a greater problem than if the original data had been left alone.

The message goes from antenna to command receiver for amplification and demodulation. Then it is routed to the decoder for breakout into the original pattern of digital bits.

The first validity check is here. If there is an error in a bit, or a bit is missing, the entire message is rejected. Accepted commands get further checking in the command decoder and in the LVDC.

First the vehicle address is checked in the command decoder. This is important because commands for both IU instruments and the spacecraft use similar command links. If the spacecraft address is recognized, the IU ignores the message.

Passing this test, the message is sent to the LVDC. Upon receipt, the LVDC tests the message to determine if it is proper. If it is, then the command decoder releases a pulse via the telemetry link to the ground station verifying message acceptance. If the message fails the test, the LVDC rejects it and telemeters an error message.

Depending on the mission, several types of messages can be processed. For example: commands to perform updating, commands to perform tests, commands to perform special subroutines or special modes of operation, a command to dump or clear certain sectors of the computer memory, or a command to relay a particular address in the computer memory to the ground. Provisions have been made to expand the number of types of messages if experience indicates this is necessary.

SWITCH SELECTOR

All stages, and the IU, are equipped with a switch selector. This unit has electronic and electromechanical components which decode LVDC/LVDA sequence commands and switches them to the proper circuits en each stage. This system has several advantages: reduction of stage interface lines, increased flexibility with respect to timing and sequencing, and conserving the discrete output circuitry in the LVDC/LVDA. Sequencing commands can come as fast as every 100 milliseconds.

Stage power isolation is maintained in the switch selector by using relays as the input circuit. The relays are driven by IU power, while the decoding circuitry and driver output are powered from the parent stage. Input and output are coupled through relay contacts. These contacts drive a diode matrix used in decoding the 8-bit input code to select the output driver, producing the switch selector output.

There also is a check and proceed system built into the switch selector. After the switch selector relays have been "picked," the complement of the received message is fed back to the LVDA/LVDC where it is checked. If the feedback is good, a read command is issued. If there is disagreement, a new message is sent which accomplishes the same function. (Note: For redundancy, two messages' codes are assigned for each switch selector output).

Electrical System

The electrical system powers the IU's equipment. As with most of the IU's systems, the electrical system is divided into two sectors: prelaunch and flight. Ground sources provide power through the umbilical lines before launch. At approximately 25 seconds prior to liftoff, power is transferred to the four 28 VDC IU batteries. Each battery has a 350 ampere hour capacity, and loads are equally distributed to drain.

Two special power supplies are provided: a 5-volt master measuring voltage supply converts 28 VDC main supply to a highly regulated 5 VDC for reference and supply voltage to the measuring components, and a 56-volt power supply for operation of the guidance system's ST-124-M inertial platform and the platform's AC power supply.

In order to get the most out of the battery stored power during flight, the LVDC and LVDA turn off unused or unimportant circuits in favor of more important applications as the mission progresses.

EMERGENCY DETECTION EQUIPMENT

The Saturn V is equipped with a myriad of equipment designed to detect malfunctions. Some of this equipment checks engine thrust, and monitors guidance computer status, attitude rates, angle of attack, and abort request.

This emergency detection information is flashed to the IU where it is routed to the emergency detection distributor (EDS) in the electrical system. The EDS distributor is an interconnector and switching point and has the logic circuits which determine the emergency. In case of a malfunction, the equipment will turn on a light in front of the astronauts. If the spacecraft abort selector switch is in the automatic abort position, the abort will take place without further crew participation; the action cannot be vetoed by the astronauts. However, if the selector switch is in the manual position, the crew, consulting with NASA flight controllers, decides when to abort a mission.

FACILITIES

INTRODUCTION

Assembly, test, and launch facilities for the Saturn V consist of a combination of facilities which existed before the onset of the program as well as many specifically created for its execution.

Included in these facilities are installations set up by the National Aeronautics and Space Administration to meet the greatly increased size and complexity of the Saturn program.

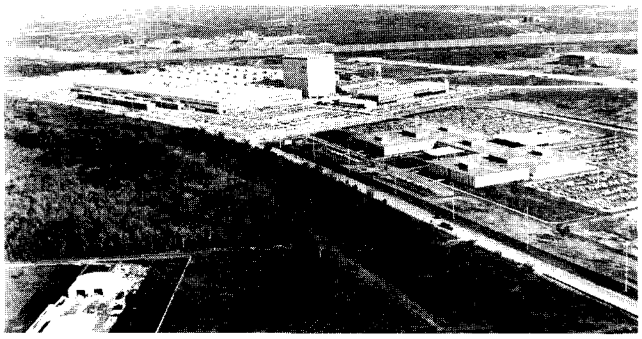
The Marshall Space Flight Center includes installations at Huntsville, Ala., where vehicle development is the prime responsibility; Michoud Assembly Facility, New Orleans, La., where the first stage is fabricated and assembled; and Mississippi Test Facility, Bay St. Louis, Miss., which is responsible for test operations. Launch facilities are located at the NASA Kennedy Space Center, Fla.

Because of the giant size of Saturn launch vehicles and the difficulties in transporting them, fabrication and test facilities were located within easy water shipment to the launch site.

At all of these NASA installations are employes of the prime contractors which build the various stages and components of the Saturn V. Other facilities, including the home bases of the major contractors and subcontractors, are located across the nation.

BOEING FACILITIES

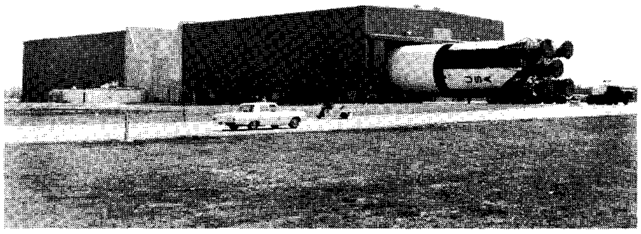
The Boeing Company manufactures the Saturn V first stage at the 900-acre NASA Michoud Assembly Facility in New Orleans. The facility has about 2,000,000 square feet of manufacturing floor space and about 730,000 square feet of office space. About 60 per cent of the manufacturing area is occupied by Boeing.



B-4793-9

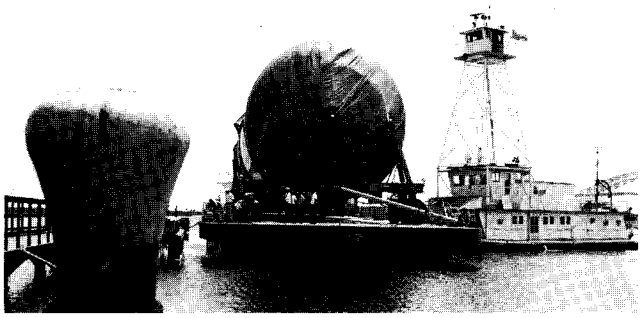
Michoud—The Michoud Assembly Facility is the fabrication site of the first stage booster. Dominating the skyline is the Vertical Assembly Building.

The plant is arranged for logical and efficient flow of materials from the loading dock through to final assembly. Paralleling the material flow are the rework and modification area and the test and laboratory areas. There are 50,000 square feet of tooling area in the plant.



B-9847-7

Stage Test—Before leaving Michoud, the completed booster undergoes a simulated firing during which all systems function in the Stage Test Building.



B-14265-5

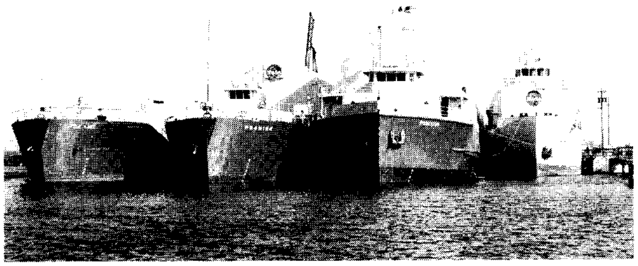
Barge Slip—First stages are loaded onto barges at Michoud and travel by waterways from New Orleans to Huntsville, Mississippi Test Facility, and Kennedy Space Center.

The environmentally controlled portion of the minor assembly area contains facilities for heat treatment, chemical cleaning, conversion coating, and welding of pre-formed metal sections received at the loading dock. Final assembly of the propellant tanks and the joining of the major components into the complete stage occur in the Vertical Assembly Building (VAB).

The VAB is a single-story structure rising the equivalent of 18 stories. A 180-ton overhead crane is used to stack the five large cylindrical segments of the first stage into a vertical assembly position. A $50 million program included the construction of three buildings—the VAB, the Stage Test Building, and the Engineering and Office Building—as well as the renovation of existing facilities at Michoud.

Checkout of the stage's electrical and mechanical systems is performed in the four giant test cells of the Stage Test Building. Each of the test cells is 83 by 191 feet with 51 feet of clear height. Each has separate test and checkout equipment.

Stages leave and enter Michoud by waterways connecting to the Mississippi River or the Gulf of Mexico.



B-8930-10

Unique Vessels—Four of six special barges used to carry Saturn rocket stages are shown moored side-by-side at the Michoud Assembly Facility. From left are the Little Lake, the Promise, the Poseidon, and the Palaemon.

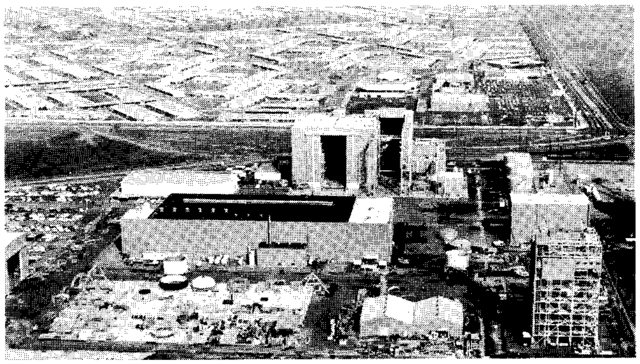
SECOND STAGE FACILITIES

The second stage of the Saturn V is manufactured and tested in facilities located from one end of the nation to the other.

The main fabrication and testing facilities are located in Seal Beach, Calif., about 15 miles south of Downey, which is the headquarters of SD operations. SD subcontracts important elements of work to other North American facilities in Los Angeles and Tulsa and McAlester, Okla. The complex of NASA buildings at Seal Beach all built especially for the second stage, were complemented in mid-1967 with three North American Rockwell-owned buildings which house all the second stage administrative, engineering, and support personnel who currently are located at Downey.

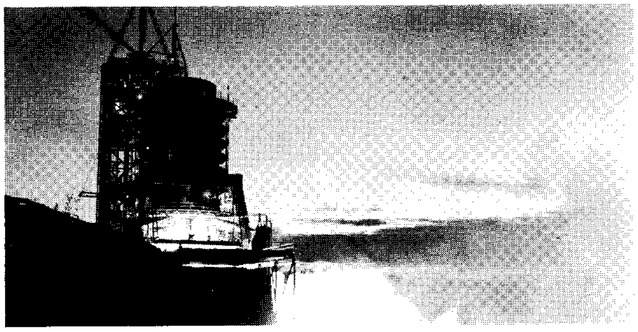
The Seal Beach facility includes a bulkhead fabrication building, 125-foot-high vertical assembly building, 116-foot-tall vertical checkout building, pneumatic test and packaging building, and a number of other structures.

The bulkhead fabrication building is a large, highly specialized structure designed solely for the construction and assembly of the second stage's three bulkheads. Among other tooling it contains an autoclave about 40 feet in diameter with a 40-foot dome for curing the large stage bulkheads.



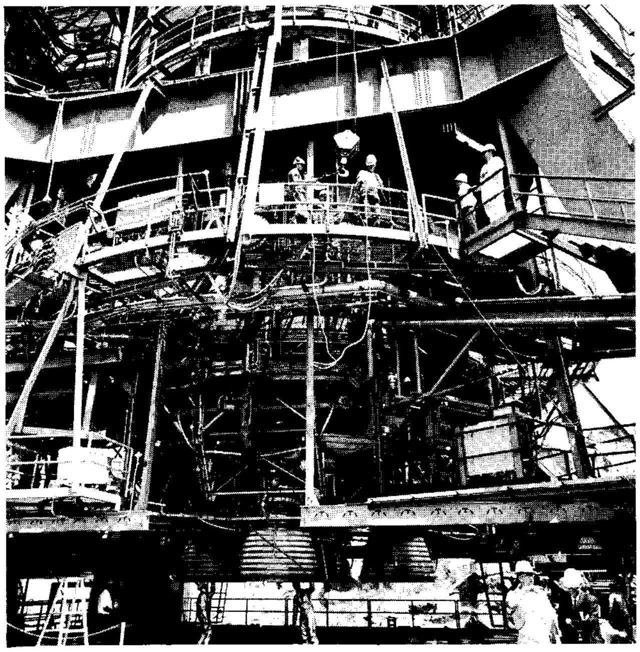
S-27

Over-all View—NASA Seal Beach facilities used by NAR include in-process storage building (left); bulkhead fabrication building (center); vertical assembly building (far right); pneumatic test and packaging building (right center); and structural test tower (right front).



S-28

Night firing of Test Second Stage at Santa Susana



S-29

Space Truck Readied—The five engines of the Saturn V second stage dwarf technicians preparing the "battleship" vehicle for hot firing at North American's Santa Susana static test lab.

The vertical assembly building, where the stage is assembled, contains six work stations at which successive major parts of the stage are added.

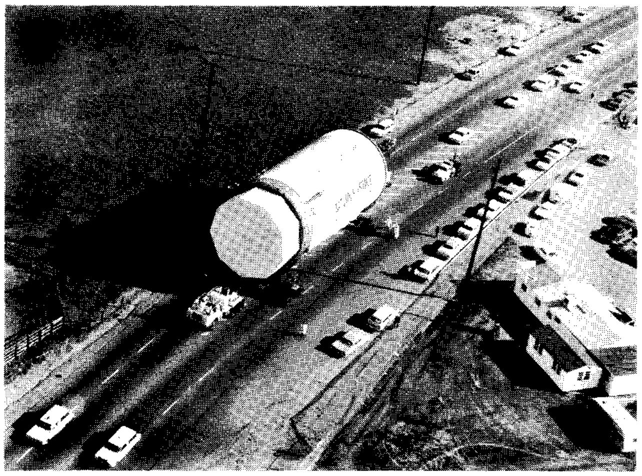
After assembly, the stage is moved to the vertical checkout building, where some final installations are made and where its systems and components are given final tests.

The last stop for a stage is the pneumatic test and packaging building, where stages are turned horizontally for pneumatic tests, painted, and prepared for shipping.

Other buildings at Seal Beach provide for such things as processing and storage of subassemblies and machine and tool shop services.

Second stage engine (J-2) testing is performed at the Rocketdyne Santa Susana facility. The Coca test area at Santa Susana operated by SD was rebuilt for the second stage and is where battleship test firings are conducted.

SD also operates facilities at both Mississippi Test Facility and Kennedy Space Center to provide management and operational support services.



S-30

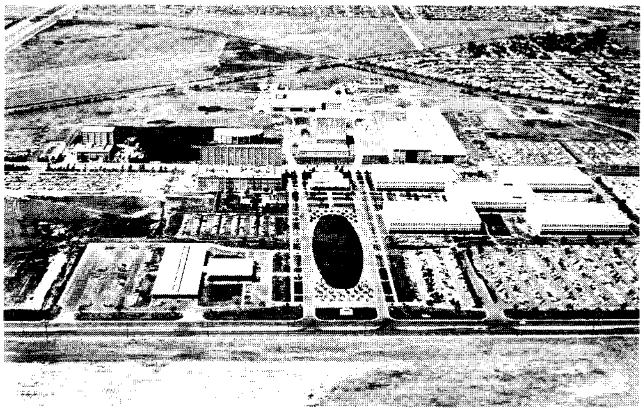
Wide Load—A second stage is transported from Seal Beach to the Navy dock for shipment to Mississippi Test Facility.

MCDONNELL DOUGLAS FACILITIES

The McDonnell Douglas Astronautics Company Space Systems Center at Huntington Beach, Calif., is the master facility for engineering and production of the third stage of Saturn V. The center is headquarters for the Missile and Space Systems Division and for direction of Saturn activities in other company facilities at Santa Monica and Sacramento, Calif., and at Cape Kennedy, Fla.

Fabrication

Initial component fabrication for the Saturn V third stage is accomplished at the Santa Monica plant. It produces parts and subassemblies ranging from micro-miniature electronic components to the complete liquid oxygen tank.



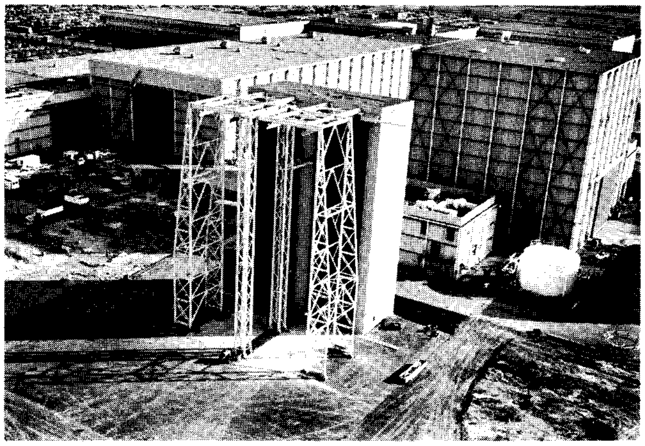
D-NRV-43

Aerial View of Space Systems Center, Huntington Beach, Calif.

Final assembly and factory checkout of the third stage takes place at the Space Systems Center. High-bay manufacturing area is provided for production of propellant tanks, skirts, and interstages. Eight tower positions are available for vertical assembly and checkout of completed vehicles.

The bulk of research and development testing in the third stage program is carried out in laboratory facilities at the Santa Monica plant. There components and subassemblies are put through a complete qualification test program in some 80 different laboratories.

Other test facilities include the Space Simulation Laboratory at the Space Systems Center, where major Saturn subassemblies are subjected to space-



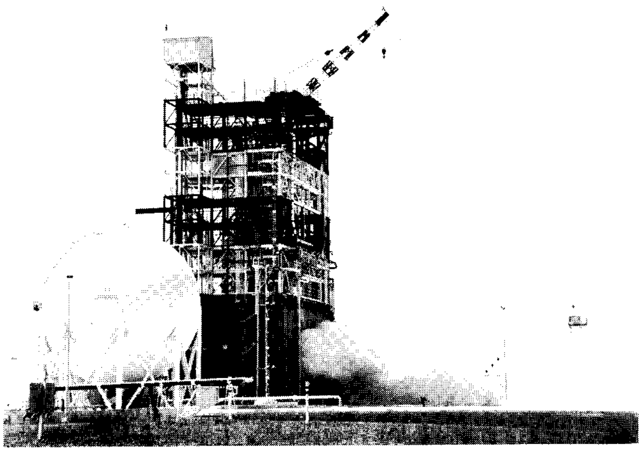
D-012220

Assembly and Checkout Tower, Douglas Space Systems Center like conditions by being placed inside a 39-foot diameter vacuum chamber for extended periods of time. The chamber is capable of simulating the vacuum at an altitude of 500 miles above the earth. Structural tests on major vehicle structures such as the propellant tank, skirt sections, and interstage are conducted in the Structural Test Laboratory at the Space Systems Center.

Two vertical checkout towers at the Space Systems Center provide for the final factory tests on finished third stages, prior to shipment from the plant for test firing. The vertical checkout laboratory is equipped with two complete sets of automatic checkout equipment.

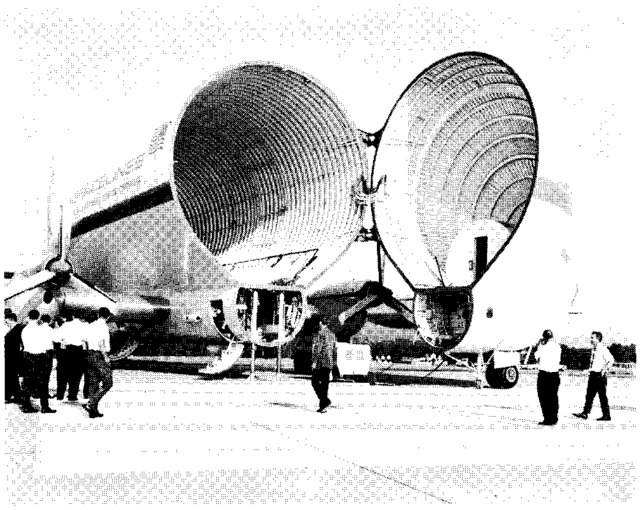
Actual ground test firings of the stages are accomplished at the Douglas Sacramento Test Center, where each stage is delivered following the completion of assembly and checkout at the Huntington Beach plant.

Primary Saturn facilities at Sacramento include a pair of 150-foot-high steel and concrete test stands where the stages are put through the final vehicle acceptance test—a full-duration, full-power static firing, simulating actual launch operations.


D-NRV-48

Static Test Firing of Third Stage at Sacramento

The Super Guppy, the world's largest airplane, is the primary means of transporting the third stage from the Douglas Huntington Beach plant to the Sacramento Test Center, and from Sacramento to KSC. Developed by Aero Spacelines, Inc., for transport of large space hardware, the plane has an inside diameter of 25 feet and a total length of 141 feet. Tail height is 46 feet—almost five stories above the ground. Cubic displacement of the fuselage is 49,790 cubic feet, approximately five times that of most present jet transports. The airplane is powered by four turbo-prop engines, producing a total of 28,000 horsepower.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Super Guppy

IBM FACILITIES

Three IBM-owned buildings at Huntsville comprise the Space Systems Center where component testing, fabrication, assembly, and systems checkout of the instrument unit are completed. Assembly and the majority of the testing activity take place in a 130,000-square-foot building located in Huntsville's Research Park.

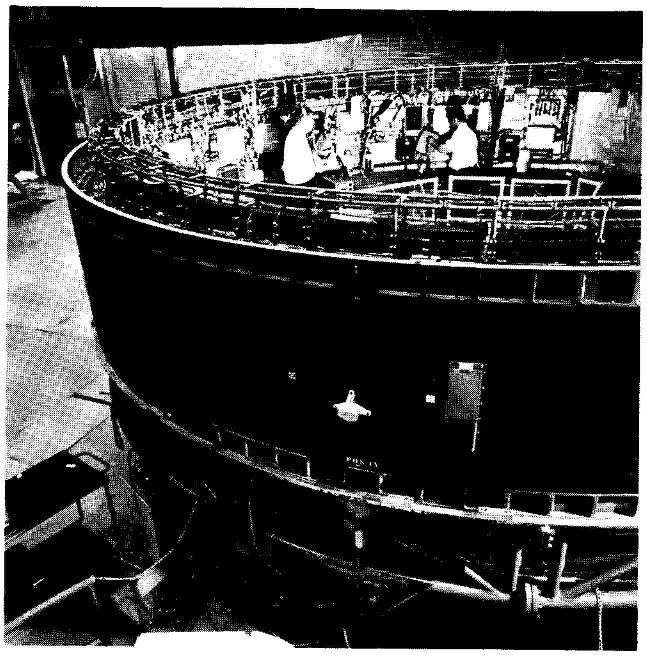
As units are received, they are inspected and then moved to one of the testing laboratories where they are subjected to detailed quality and reliability testing. From component testing, the parts move


IBM-DR-24

IU Assembly and Test—All instrument unit assembly work and the majority of testing are done in this IBM-owned building in Huntsville's Research Park. The rear of the building is the high-bay area where assembly operations take place.

to inventory until called out for assembly.

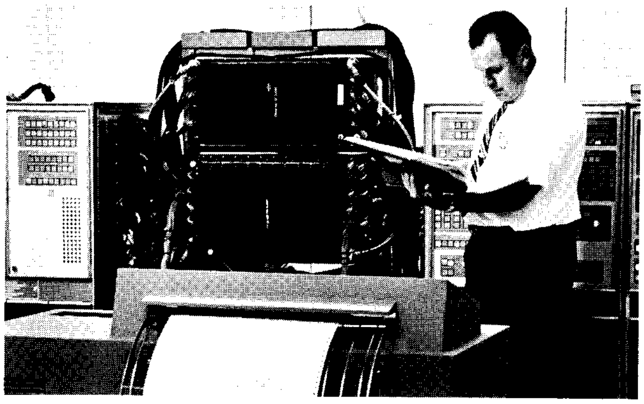
Following assembly operations, the IU is moved to one of two systems checkout stands—one for uprated Saturn I vehicles, the other for Saturn V.



IBM-DR-25

Automatic Checkout—IBM technicians monitor systems checkout tests as another technician optically adjusts the inertial guidance platform, prior to a simulated mission.

A complete systems checkout is performed automatically. Hooked by underground cables, two digital checkout computer systems examine the IU. Each of the IU's six subsystems is tested before the IU is tested as an integrated unit. With independent computers, systems tests for two instrument units can be conducted simultaneously.



IBM-DR-26

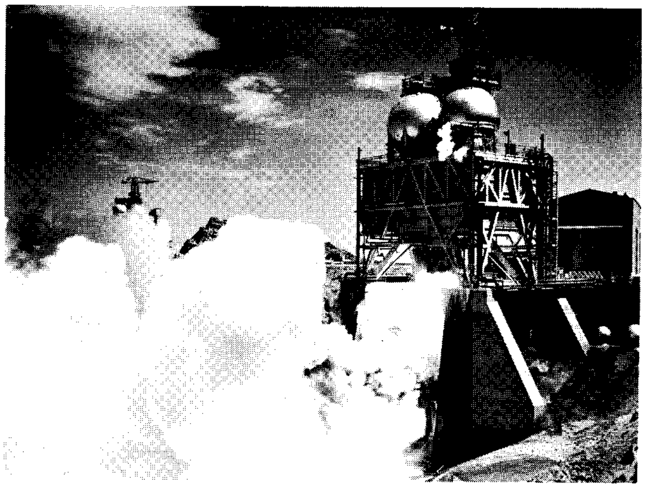
Simulation Laboratory—Saturn V flight guidance and navigation programs as well as launch computer programs are tested in IBM's Engineering Building at Huntsville. Here a technician checks a computer readout of a simulated mission.

ENGINE FACILITIES

F-1 and J-2 engines for the Saturn V launch vehicle are manufactured at Rocketdyne's main complex in Canoga Park, Calif. F-1 static testing is conducted at the Edwards Field Laboratory located at the NASA Rocket Engine Test Site, Edwards, Calif., about 125 miles northeast of Los Angeles, and the J-2 is tested at Rocketdyne's Santa Susana Field Laboratory located about 10 miles from Canoga Park.

R-11

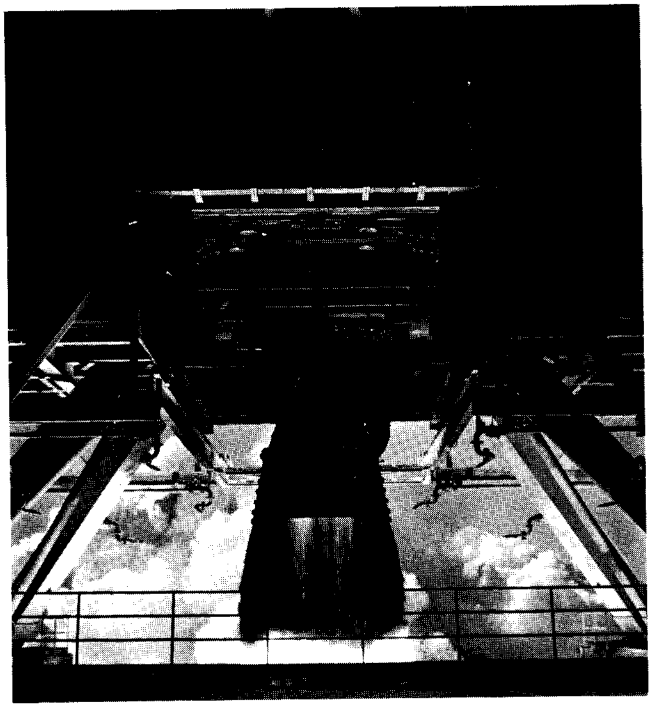
F-1 Test Stands—Three of six stands for testing F-1 rocket engines or components at full thrust are visible in this aerial view of NASA Rocket Engine Test Site.



R-12

F-1 Test Firing—An F-1 rocket engine developing 1,500,000 pounds of thrust is tested at NASA Rocket Engine Test Site. The stand is one of six in the complex.

Manufacturing of components and final assembly of both engines are carried out in eight buildings in the Canoga complex. These facilities are equipped with general purpose machine tools for precision and heavy machining as well as some 20 numerically controlled machines for performing programmed multiple machining operations. Also included are two of the largest gas-fired brazing furnaces of their type for brazing of thrust chamber tubes and injectors, eight units for ultrasonic cleaning, 21 installations for Gamma and X-ray inspection, more than 50 environmentally controlled areas for ultraclean assembly operations, sheet metal preparation, precision cleaning, and receiving inspection.



R-13

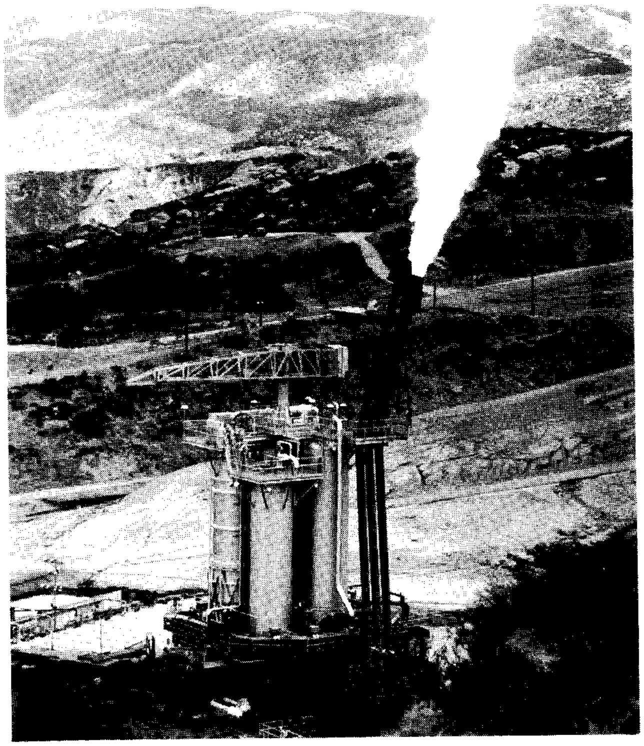
F-1 Flight Engine Firing

An Engineering Development Laboratory provides specialized facilities to support manufacturing programs. These facilities include a high-flow water test facility for checking propellant systems, 12 concrete cells for conducting hazardous tests, 28 environmental test chambers, a photo-elastic laboratory, two pneumatic flow benches, six vibration test rooms, and others for checking components as well as complete engines.

Research and development testing of F-1 turbomachinery, gas generators, heat-exchangers, seals, and splines is conducted on two test stands and three components test laboratories at Santa Susana.

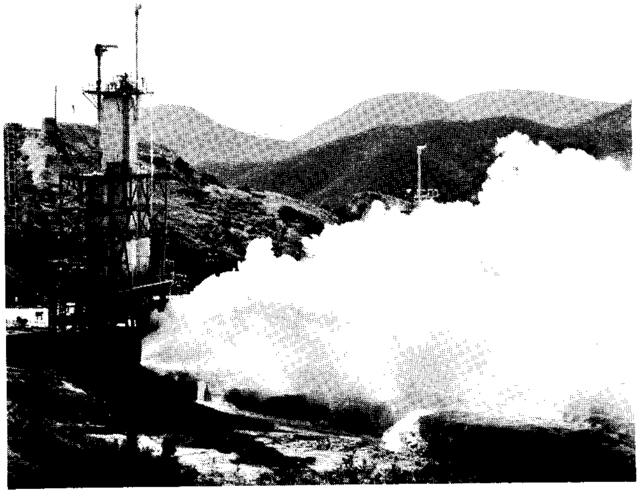
Six large test stands, with a total of eight test positions, and associated shops and support facilities at the Edwards Field Laboratory are used for testing complete F-1 engines as well as injectors.

Six large engine test stand positions at the Santa Susana Field Laboratory are used for testing the J-2. One of these stands is equipped with a steam injection diffuser for altitude simulation testing. J-2 turbopumps, gas generators, valves seals, bearings, and other components are tested in 22 test cells in five component test laboratories in Santa Susana.



R-14

Pump Tests—Flames from gases burned during test of an F-1 engine turbopump shoot more than 150 feet in air.



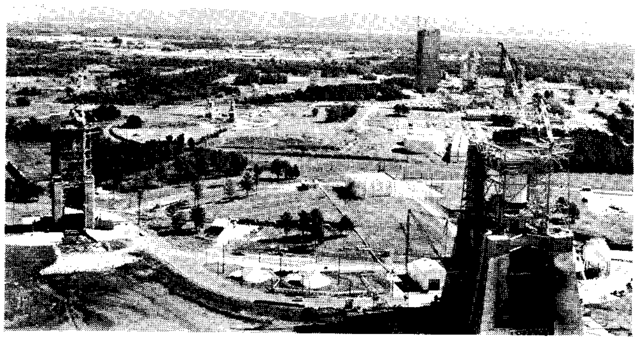
R-15

J-2 Testing—A hydrogen fueled J-2 rocket engine is tested under ambient altitude conditions at Santa Susana.

HUNTSVILLE FACILITIES

New Saturn V facilities built at the Marshall Space Flight Center at Huntsville, Ala., include the first stage static test stand, an F-1 engine test stand, the Saturn V launch vehicle dynamic test stand, a J-2 engine facility, and ground support and component test positions.

The Marshall Center built a $39 million addition to its Test Laboratory for captive testing the Saturn V booster and F-1 engines. The Test Laboratory addition is called the West Test Area. The largest structure in the area is the first stage test stand. Completed in 1964, the stand has an overall height of 405 feet.



H-5-31176

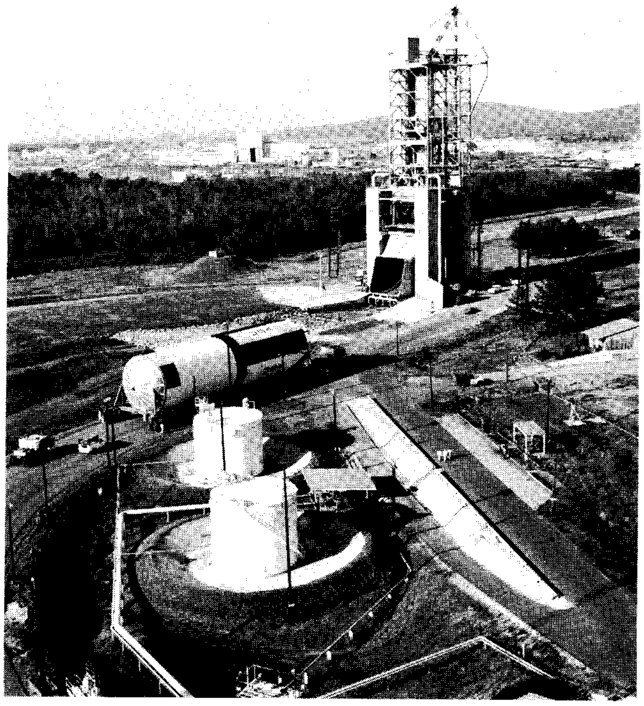
Growth at Huntsville—The growth of rocket testing facilities at the Marshall Space Flight Center is contrasted here by the size of the first Redstone Arsenal test stand, second from left, and stands at right built for the Saturn V program.

The nearby single engine test stand is being used for research and development tests of the 1.5 million pound thrust F-1 engine.

Control and monitoring equipment for the first stage and F-1 engine stands is located in the area's central blockhouse. Water needed to cool the flame deflectors of the two stands is pumped from a nearby high-pressure industrial water station.

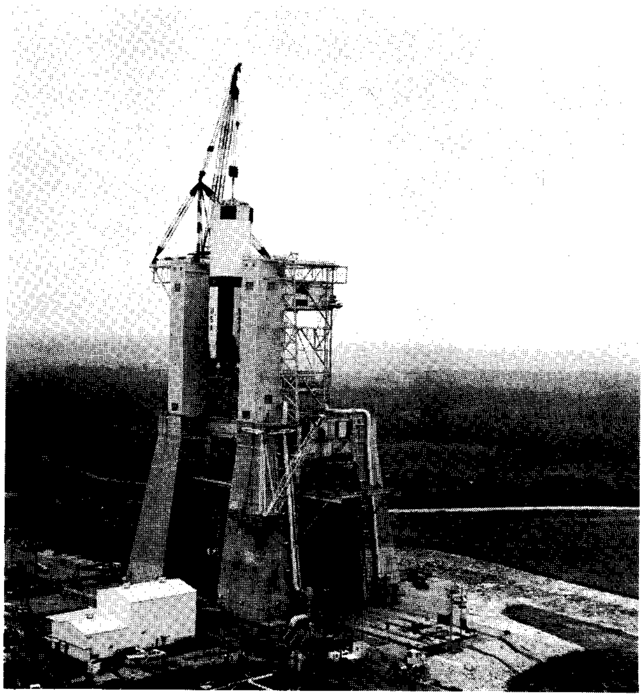
The 363-foot tall Saturn V launch vehicle was placed in another unique Marshall Center test facility—the Dynamic Test Stand. Testing of the complete three stage vehicle and its Apollo spacecraft here was done to determine its bending and vibration characteristics. Tallest of Marshall Center's tall towers, the dynamic test stand is 98 feet square.

Numerous tests of the liquid hydrogen-liquid oxygen powered engine have been conducted in Marshall's J-2 engine test facility. Tankage for the facility is a battleship version of the Saturn V third stage. The J-2 engine stand is 156 feet tall and has a base of 34 by 68 feet. It is located in the MSFC's East Test Area.



H-5-31246

Vibration Version—A ground test version of the Saturn V first stage moves through the West Test Area of the Marshall Space Flight Center. The large dynamic test stage was built to undergo vibration and bending tests. Test stand at right is a single F-1 engine facility.



H-40246

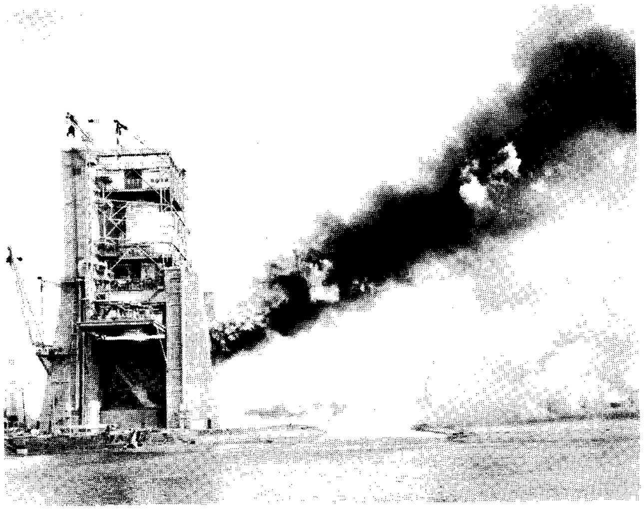
Positioning—A Saturn V first stage is placed into a test stand at Marshall Space Flight Center.

A portion of the Kennedy Space Center "spaceport" was created at the Marshall Center's ground support equipment test facility to check out giant mechanical swing arms which are used on Launch Complex 39 to connect the Apollo/Saturn V space vehicle to the launcher tower. The 18-acre facility has eight swing arm test positions and one position for testing access arms to be used by Apollo astronauts.

Also at Marshall are an F-1 engine turbopump position in the East Test Area and a load test facility in the Propulsion and Vehicle Engineering Laboratory.

A new Saturn V rocket "electrical simulator" or breadboard facility at the Marshall Center duplicates the electrical operation of the vehicle. Elements simulated include the first stage booster, second stage, third stage, and an instrument unit.

Other Saturn V facilities at the Marshall Center include a booster checkout area, two new assembly areas and a components acceptance building.



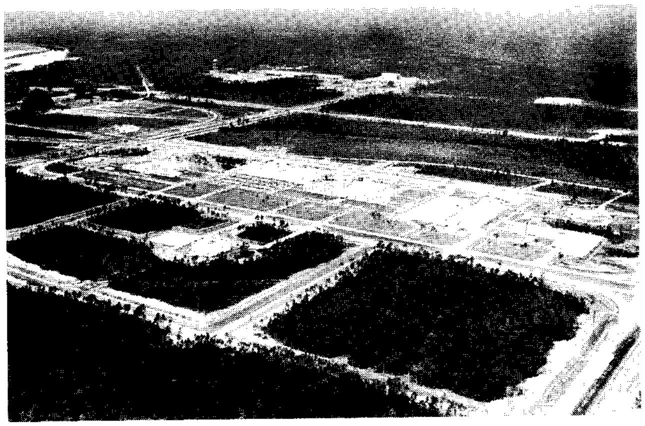
H-5-23531

Static Firing—The Marshall Space Flight Center captive fired all five F-1 engines of the Saturn V S-IC-T for 16½ seconds on May 6, 1965. Later they were fired for 41 seconds.

MISSISSIPPI TEST FACILITY

NASA has developed the Mississippi Test Facility, a field organization of the Marshall Space Flight Center, as a testing site for the Saturn V launch vehicle's two lower stages.

Acceptance testing of first and second stages are being conducted at the $300 million facility. In addition, limited repair and modification of J-2 engines will be performed at MTF on behalf of all NASA operations in the Southeast.



H-MTF-2077

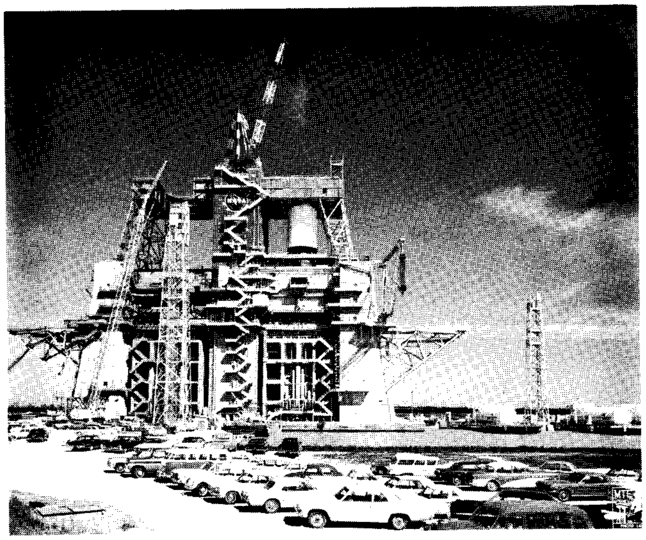
Aerial View of Mississippi Test Facility

The General Electric Co., under a prime contract with NASA, operates and maintains the facility, providing site services, technical systems, and test support to NASA and to stage contractors and other tenants.

North American Rockwell through its Space Division, is the prime contractor to NASA for developmental and acceptance testing of second stages. SD personnel conduct the tests within the second stage test complex.

The Boeing Company is the prime contractor to NASA for developmental and acceptance testing of first stages. Stages manufactured by Boeing are tested by the company in the first stage test complex.

The U. S. Army Corps of Engineers was NASA's agent for land acquisition, design engineering, and construction.



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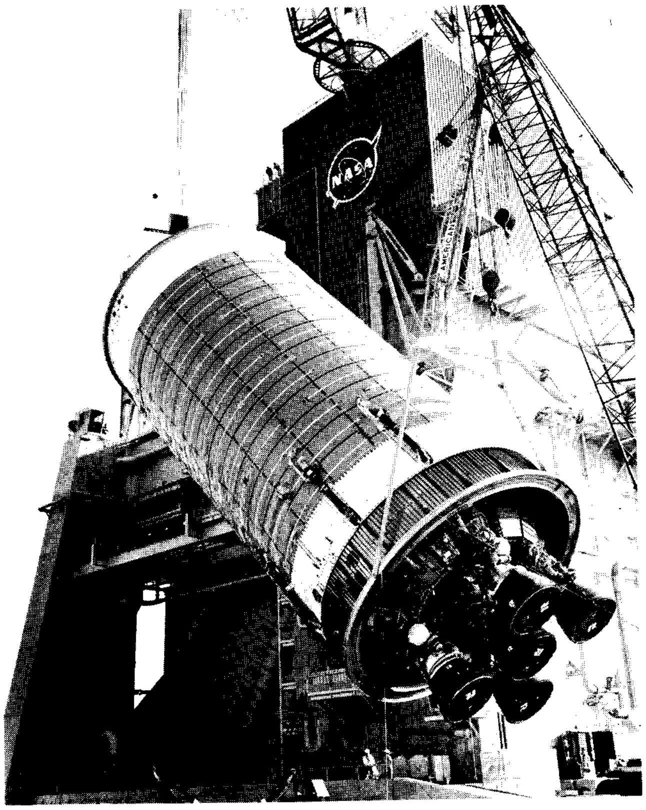
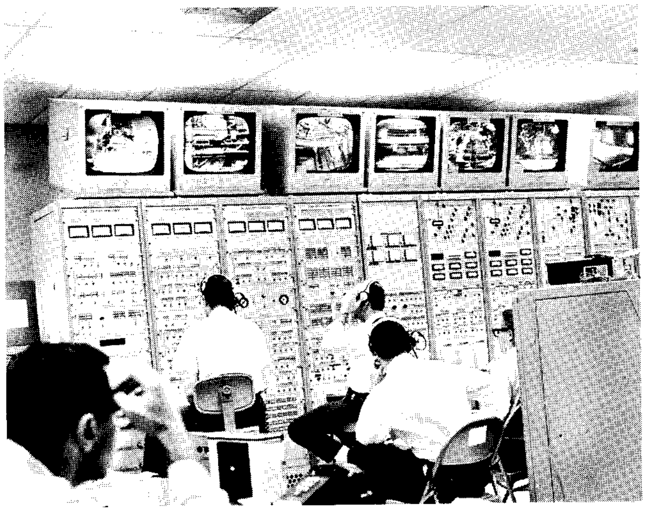
First Stage Test Stand at MTF



H-MTF-4215

Stage Hoisted—The all-systems test version of the second stage is lifted into its test stand at MTF.

Management, operational, and support personnel engaged at the Mississippi Test Facility number approximately 2,890.



H-MTF-1432-B7

Checkout—Engineers and technicians of North American Rockwell are shown in the second stage Test Control Center at the Mississippi Test Facility during final preparation for static firing of all-systems test model of the stage.

The MTF site was selected from 34 areas considered mainly because of its accessibility to water routes and its nearness (45 miles by water) to the Michoud Assembly Facility in New Orleans. The government-owned fee area comprises 13,424 acres and is surrounded by an acoustic buffer zone involving an additional 128,526 acres in Hancock and Pearl River counties and Saint Tammany Parish.

H-MTF-66-1823A

Static Firing—A giant plume of vapor billows skyward during the first static firing test at MTF. The Saturn second stage, built for NASA by North American Rockwell burned for 15 seconds April 23, 1966.

MTF is composed of three principal complexes including approximately 60 buildings and structures. Among predominant features are the three huge test stands in the Saturn V complex. There are two separate stands for testing second stages. The first stage test stand is a dual-position structure which, with overhead crane, towers over 400 feet. The Laboratory and Engineering Complex houses engineering, administrative, and technical personnel. The Industrial Complex has facilities for equipment and personnel necessary for site and test support maintenance.

The relatively small force of NASA personnel assigned to MTF has overall management and supervisory responsibilities in overseeing the work of the contractors. NASA personnel are also responsible for final evaluation of static firings and issuance of flightworthiness certificates to stage contractors.

KENNEDY SPACE CENTER

Launch Philosophy

Saturn V vehicles are assembled, checked out, and launched at Launch Complex 39 at Kennedy Space Center. Complex 39 embodies a new mobile concept of launch operations which includes superior reliability and time savings offered by assembly and checkout in a protected environment and reduction of actual pad time as much as 80 per cent with a consequent increase in launch rate capability. The ability to adapt economically to future program requirements is another advantage. For example, the service platforms used in the Saturn/Apollo program could be used for other vehicles of similar configuration, and the area can accommodate space boosters with thrusts up to 40 million pounds.

Facilities

The major components of Launch Complex 39 include: (1) the Vehicle Assembly Building, where the space vehicle is assembled and prepared; (2) the mobile launcher, upon which the vehicle is erected for checkout, and from which, later, it is launched; (3) the crawlerway, upon which the fully assembled vehicle is carried by transporter to the launch site; (4) the mobile service structure, which provides external access to the vehicle at the launch site; (5) the transporter which carries the launch vehicle, mobile launcher, and mobile service structure to various positions at the launch complex; and (6) the launch area from which the space vehicle is launched.

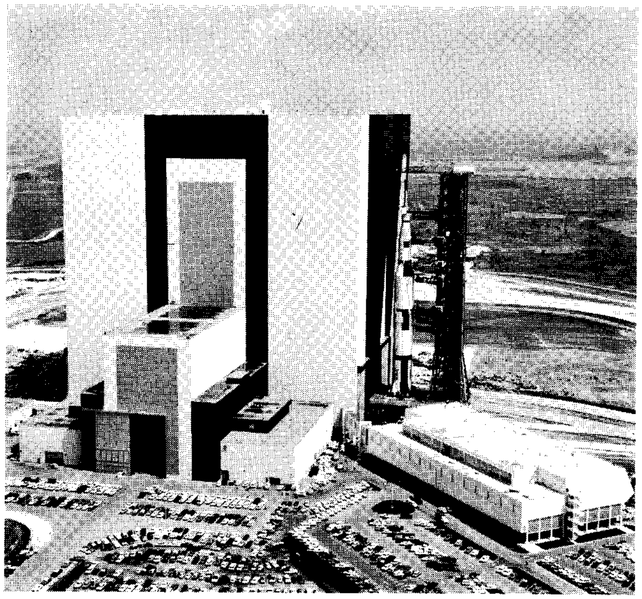
THE VEHICLE ASSEMBLY BUILDING

The Vehicle Assembly Building (VAB) consists of a high bay area 525 feet tall, a low bay area 210 feet tall, and a four-story launch control center (LCC) connected to the high bay by an enclosed bridge. The VAB, with a volume of 130 million cubic feet, covers eight acres of land. There are four assembly and checkout bays in the high bay area. The low bay area contains eight stage preparation and checkout cells equipped with systems to simulate stage interface. The launch control center houses display, monitoring, and control equipment for checkout and launch operations. There are four firing rooms in the LCC, one for each high bay and checkout area. Work platforms, mounted on opposite walls in the high bay area, are designed to enclose various work areas around the launch vehicle. Platforms extend or retract in less than 10 minutes. Twenty-ton hydraulic jacks are used to align platforms.

The Saturn V, after prelaunch checkout on its mobile launcher, is carried by the transporter from the VAB through a door shaped like an inverted "T". The door is 456 feet high. The base of the door is 149 feet wide and 113 feet high; the remainder is 76 feet wide. There are four such doors in the VAB, one for each of its four high bays. In keeping with the protective environment of the building, doors were designed to withstand winds of 125 miles per hour.

There are 141 lifting devices in the VAB, ranging from one-ton mechanical hoists to two 250-ton high-lift bridge cranes.

Each pair of high bays shares a bridge crane. The cranes have a lifting height of 456 feet and a travel distance of 431 feet.

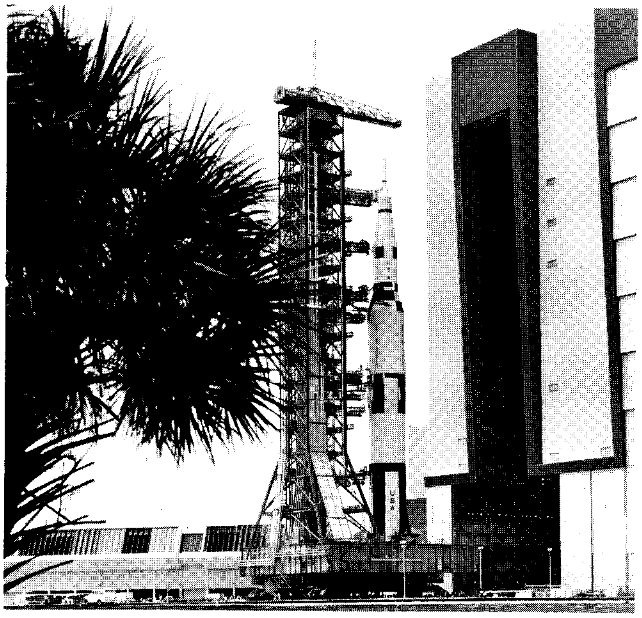


K-107-66P-237

Checkout Vehicle—The Saturn V facilities vehicle begins its journey from the Vehicle Assembly Building to the launch pad. Its purpose was to check out facilities, train launch crews, and verify procedures at KSC.

LAUNCH CONTROL CENTER

Located Southeast of the VAB is the launch control center (LCC). This four-story building is the elec-



K-107-66PC-75

Saturn V Facilities Vehicle Rollout at Kennedy Space Center tronic brain of Launch Complex 39. Here final countdown and launch of Saturn V's will be conducted. The LCC is also the facility from which a multitude of checkout and test operations will be conducted while space vehicle assembly is taking place inside the VAB.

Two separate, automated computer systems are used to check out and conduct the countdown for the Saturn V. The acceptance checkout equipment, or ACE, is used for the Apollo spacecraft. The Saturn ground computer system is used for the various stages of the vehicle.

Located in the launch control center is the heart of the Saturn ground computer system. Here checkout and preflight countdown are conducted.

This system has as its "brain" two RCA 110A computers. One is located in the launch control center and the other is in the mobile launcher upon which the Saturn V is erected.

K-100-66C-813

Moving Tower—Personnel watch a mobile launch tower moving along the crawlerway at Kennedy Space Center.

Through the process control system, all stages are checked, and data from the engines and from the guidance, flight control, propellants, measurement, and telemetry systems is provided.

The Saturn ground computer system also includes a DDP 224 display computer located in the LCC. It can drive up to 20 visual cathode ray display tubes.

The RCA 110A computer is capable of transmitting 2,016 discrete signals to the vehicle where it is possible for the computer in the mobile launcher to return 1,512 discrete signals.

A digital data acceptance system collects and makes available onboard analog data to the computers.

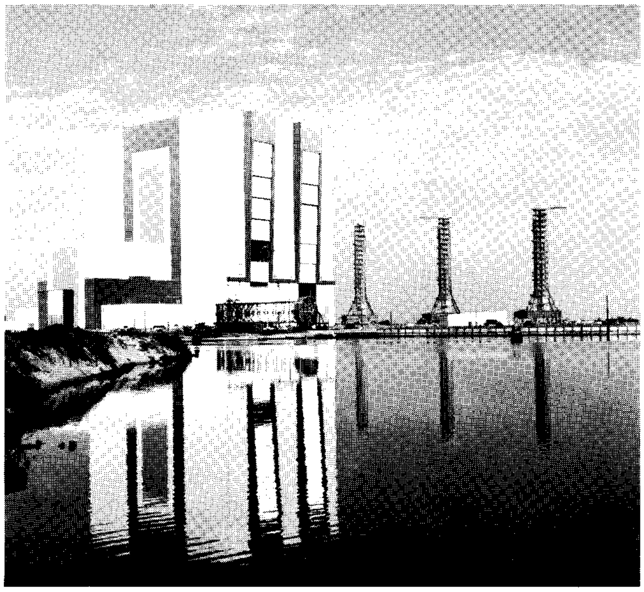
A triply redundant system for discrete output information allows more reliability. There are 1,512 signals going to the mobile launcher showing "on" and "off" commands. If one signal fails or reports a wrong command and the other two signals transmit another command, the majority command is indicated in the display and transmitted to the vehicle.

There are 15 display systems in each LCC firing room, with each system capable of giving digital information instantaneously.

Sixty television cameras are positioned around the Saturn V transmitting pictures on 10 channels.

Additionally, the LCC contains several hundred operational intercommunication channels which enable the launch team and the launch director to be in voice contact with the astronauts aboard the spacecraft.

Automatic checkout of the Apollo spacecraft is accomplished through acceptance checkout equipment (ACE). Through the use of computers, display consoles, and recording equipment, ACE provides an instantaneous, accurate method of spacecraft preflight testing. ACE also is used at the spacecraft contractor plants and in testing at the Manned Spacecraft Center in Houston.



K-100-66C-456

Vehicle Assembly Building at KSC Viewed From Across the Turning Basin of Launch Complex 39

Computerized checkout of the Saturn stages at the launch pad and the Apollo ACE system at the Manned Spacecraft Operations Building at the Kennedy Space Center are tied together by instrumentation.

The Saturn V employs completely automated computer controlled checkout systems for each of its stages. The system uses a carefully detailed computer program and associated electronic equipment to perform a complete countdown checkout of each stage and all its various systems, subsystems, and components.

With electronic speed, it moves through a thorough and reliable countdown, yet permits test engineers to monitor every step of the operation and to override the computer functions, if necessary.

To monitor fuel and oxidizer mass for the three stages of the Saturn V vehicle, a propellant tanking computer system (PTCS) is used. This system controls propellant tank fill and replenishment. Liquid oxygen and liquid hydrogen must be replenished constantly to compensate for boil-off.

PROPELLANT STORAGE AND TRANSFER

Propellant facilities at Launch Complex 39 include a LOX system, the RP-1 system, the liquid hydrogen system, the propellant tanking computer system, the spacecraft support system, and the data transmission system.

The propellant tanking computer system provides a means of monitoring amounts during the fueling operations. It also accurately controls fuel level during the final phase of tank fill and replenish.

The data transmission system provides an accurate method for the transmission of propellant and environmental control system electrical signals from the launch site to the LCC.

The liquid oxygen system provides oxidizer fill and drain for the three stages of the Saturn V. The system includes a storage tank, a vaporizer, two replenishing pumps, transfer lines, vent lines and drain basin, and electric circuitry for monitoring and actuating the pneumatic control system.

The round liquid oxygen storage tank holds 900,000 gallons and is situated 1,450 feet from the launch site. It has a stainless steel inner wall 62 feet 9 inches in diameter. The space between this inner sphere and the outer wall is filled with gaseous nitrogen and perlite for insulation.

To load liquid oxygen, a command originates in the LCC at the LOX control panel. The signal is transmitted to the mobile launcher by the data transmission system and then to the LOX storage area. The electrical signals are converted to pneumatic pressure to operate the valves, and the flow of LOX from the storage tank into the vaporizer begins. The vaporizer converts the liquid oxygen into gaseous oxygen, which then is fed back into the tank to pressurize it to the 10 psig needed to begin the flow. The pumps are started and the LOX is pumped through the transfer lines to the vehicle.

The RP-1 system provides fuel fill, drain, and filtering capabilities for the first stage. The system includes three storage tanks each with a capacity of 86,000 gallons, transfer lines, a launch site facility, and electric circuitry.

The liquid hydrogen system provides fueling and draining for the second and third stages. It includes a storage tank with a capacity of 850,000 gallons, a vaporizer, transfer lines, and a burn pond in which excess propellant is burned.

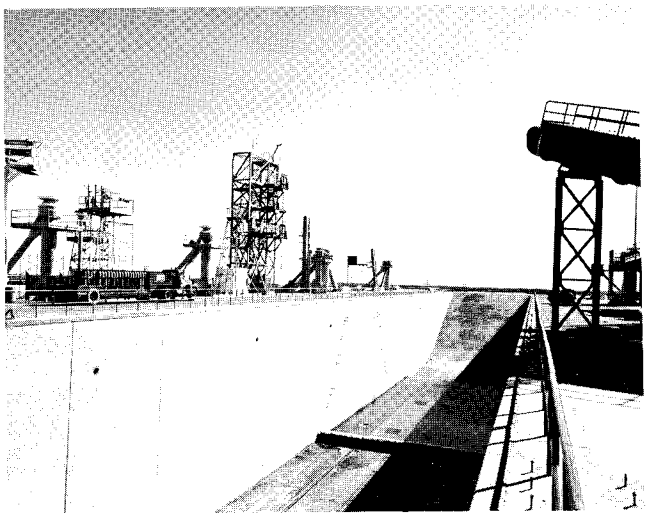
The double walled storage tank, 1,450 feet from the launch site, has a stainless steel inner wall with a diameter of 61 feet 6 inches. The space between the inner and outer walls is filled with perlite.

FLAME DEFLECTOR

To dissipate the rocket exhaust from the F-1 engines, a flame deflector, a flame trench, and a water deluge system are used in the launch area.

The inverted V-shaped steel flame deflector features a replaceable ceramic-coated leading edge. Exhaust from the outer engines strikes the point of the inverted V. At the same time, the deflector is exposed to water deluge during and after liftoff.

The center engine exhaust impinges on the ceramic



K-100-66C-825

View of Pad 39A East Side at KSC and Flame Trench from North End leading edge. The heat resistant ceramic surfaces erode slowly in the blast. As they do, the great thermal energy generated is carried away in superheated particles. All exhaust and particles are deflected through a flame trench where their energy is dissipated harmlessly into the atmosphere.

The mobile deflector weighs 1.2 million pounds and is moved to its position beneath the launch pedestal along a rail system. Two deflectors are available for each launch area, although only one is required per launch.

THE LAUNCH AREA

Final preparation of the space vehicle for launching, including propellant and ordnance loading, final checkout, and countdown takes place in the launch area.

K-100-66C-5629

Aerial of Pad 39A with VAB in Background

There are presently two launch areas on Complex 39. Each area is polygon-shaped with the linear distance from side to side at approximately 3,000 feet. The launch sites are 8,730 feet apart to allow operations on the pads to be handled independently for safety reasons.

Liquid oxygen, RP-1, and liquid hydrogen are stored near the perimeter of the launch sites. Helium and nitrogen gases are stored at 10,000 psi near the center.

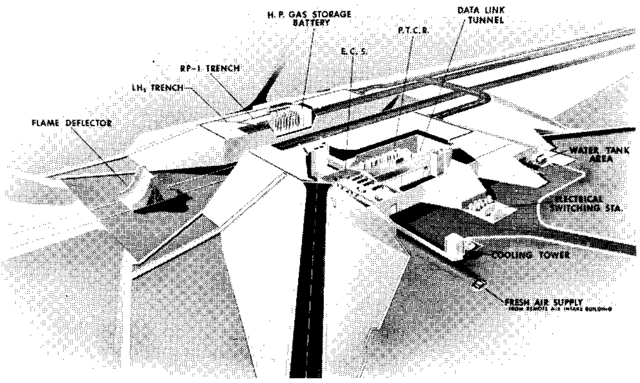
An elevated steel and concrete hardstand is located in the center of each area. Steel support fittings for the mobile launcher and mobile service structure are anchored to the hardstand. The exhaust flame trench runs through the center of the hardstand. Prior to the launch, the wedge-shaped flame deflector is moved along rails into the trench.

The liquid oxygen system consists of a 900,000-gallon LOX storage facility and transfer system.

The RP-1 system consists of a storage area containing three 86,000-gallon tanks and a transfer system. The tanks have a carbon steel shell and a bonded stainless steel lining.

Gaseous nitrogen and helium are stored underground in vessels near the launch pad at pressures of 6,000 psi.

Automation of vehicle prelaunch checkout is expected to uprate mission confidence and to increase launch rate capability. The heart of this automatic checkout system is the computer complex.



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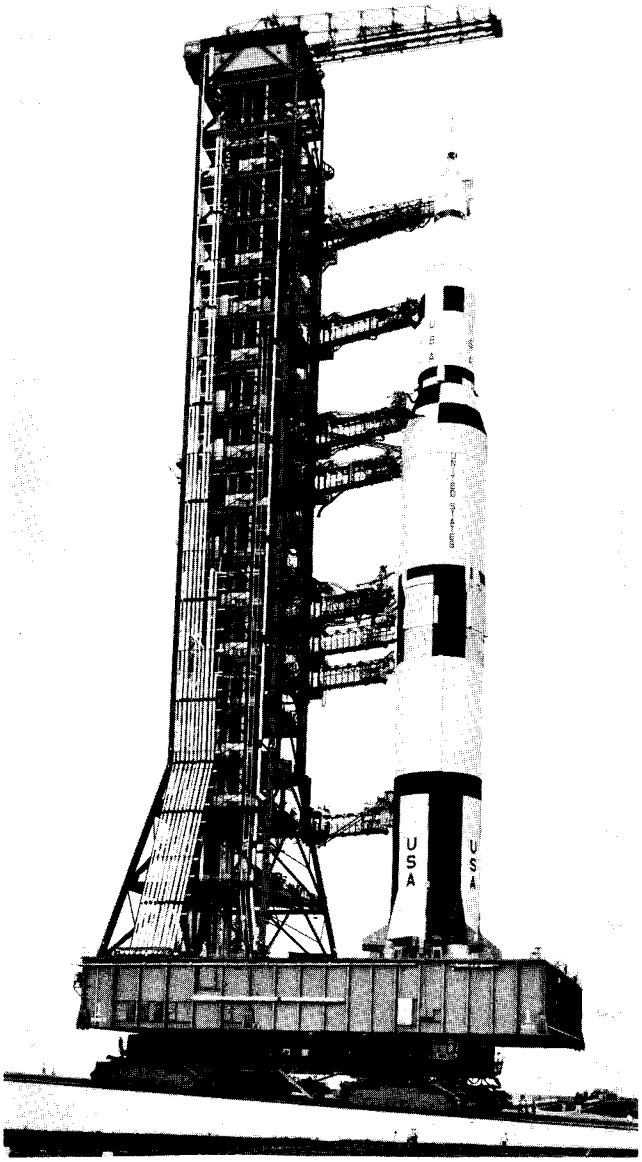
Cutaway Illustration of Pad 39A

Transporter

The capacity to transport the massive mobile launcher with a fully erected Saturn V in launch-ready condition is a key to the mobile concept of Launch Complex 39. This is accomplished by a huge transporter which moves the mobile launcher and vehicle from the VAB to the launch site, approximately 3.5 miles away. The transporter, weighing 6-million pounds, has top speeds of 2 miles per hour unloaded, 1 mile per hour carrying the mobile launcher and space vehicle (dry) or .8 miles per hour with the mobile service structure. The vehicle—131 feet long and 114 feet wide—moves on four double-tracked units, each 10 feet high and 40 feet long. Each unit is driven by an electric motor.

Tractive power is provided by 16 direct current motors served by two diesel-driven direct current generators. The generators are rated at 1,000 kilowatts each and are driven by 2,750 horsepower diesel engines. Speed of the vehicle is controlled by varying the generator fields. Power for the fields is provided by two 750-kilowatt power units which also provide power for pumps, lights, instrumentation, and communications.

The transporter is one of the largest land vehicles ever constructed. Yet, in transit, it must maintain a level platform within 10 minutes of arc and be capable of locating itself at its launch site and VAB positions within a 2-inch tolerance.



K-107-66PC-87

AS-500F, mobile launcher on transporter climbs ramp of Pad A, Launch Complex 39.

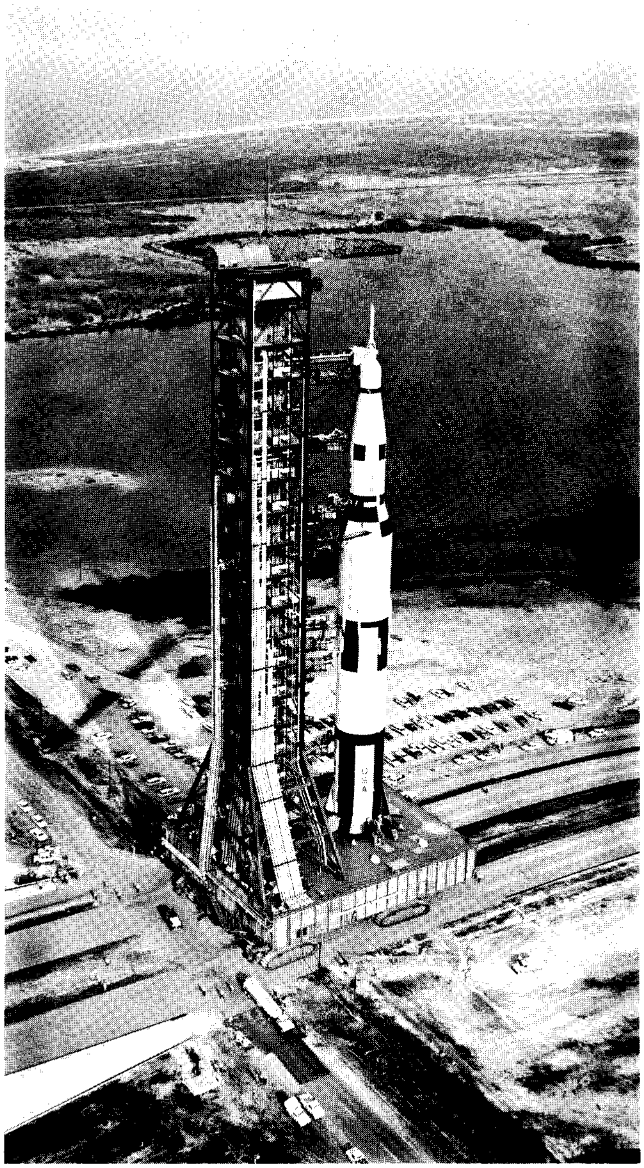
MOBILE SERVICE STRUCTURE

External access to the Saturn V space vehicle at the launch site is provided by the mobile service structure. The steel-truss structure rises 410 feet above ground level and 363 feet above the deck of the mobile launcher. It has five platforms which close around the vehicle. Two platforms are powered to move up and down. The remaining three are relocatable, but not self-powered. A mechanical equipment room, operations support room, communications and television equipment room, and various other equipment compartments are located in the base.

The service structure is moved to and from the pad by the transporter. Once in position, either at the launch pad or in a parking area, the structure is anchored to support pedestals. The service structure remains in position at the pad until about T-7 hours when it is removed to its parking area 7,000 feet from the pad.

MOBILE LAUNCHER

The mobile launcher is a movable launch platform with an integral umbilical tower. The launcher base



K-107-66PC-91

Arrival to Launch Pad—The facilities vehicle arrives at Launch Complex 39A.

is a two-story steel structure covering more than half an acre. The 398-foot tower, which supports the electrical servicing and fluid lines for the vehicle, is a steel structure mounted on the base. The base and tower weigh 10.5 million pounds and stand 445 feet above ground level.

Among major considerations in design of the mobile launcher were crew safety and escape provisions and protection of the platform and its equipment from blast and sonic damage.

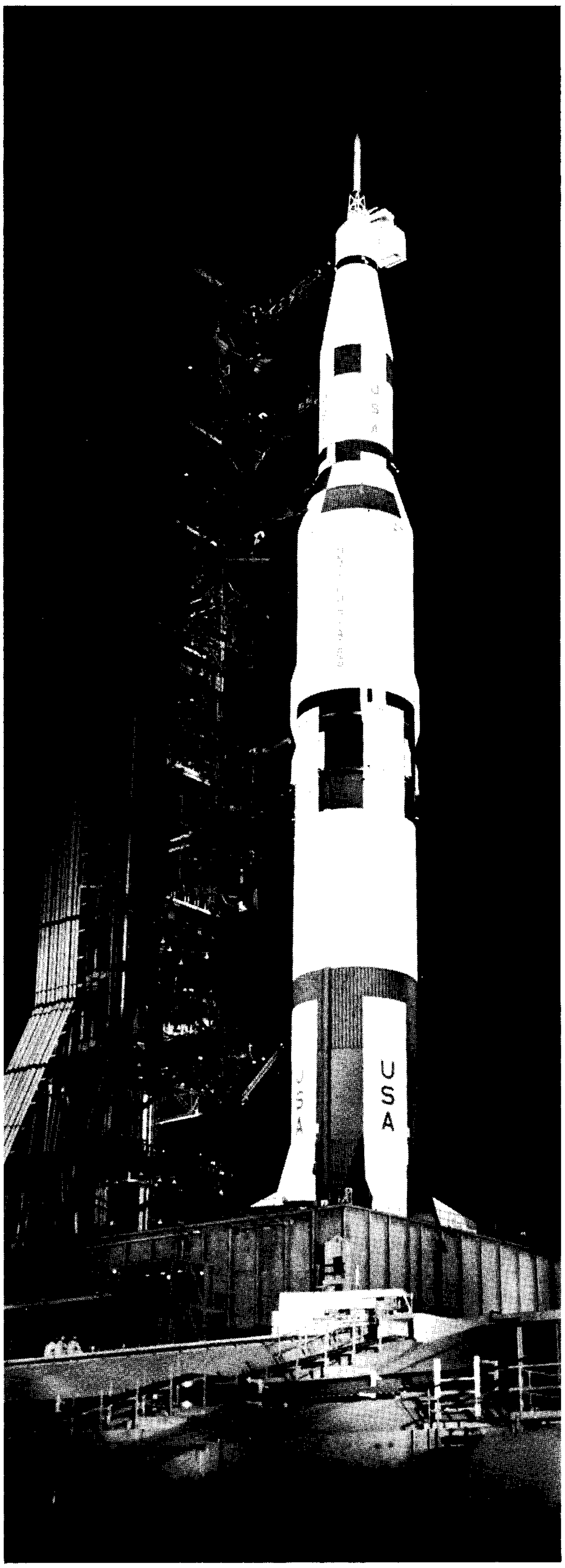
Personnel may be evacuated from upper work levels of the umbilical tower by two high speed elevators, descending at 600 feet per minute. After leaving the elevator, they can drop through a metal chute into a blast and heatproof room inside the base of the pad hardstand.

The mobile launcher provides physical support and is a major facility for checkout of the space vehicle from assembly at the VAB until liftoff at the launch site.

The top level of the launcher base houses digital acquisition units, computer systems, controls for actuation of service arms, communications equipment, water deluge panels, and other control units. Included in the lower level are hydraulic charging units, environmental control systems, electrical measuring equipment, and a terminal room for instrumentation and communications interface. Mounted on the top deck of the base are four vehicle holddown and support arms and three tail service masts.

The umbilical tower is an open steel structure providing support for nine umbilical service arms, 18 work and access platforms, and, for propellant, pneumatic, electrical, water, communications, and other service lines required to sustain the vehicle. A 250-ton capacity hammerhead crane is mounted atop the umbilical tower.

The launcher restrains the vehicle for approximately 8.9 seconds after ignition to allow thrust buildup and verification of full thrust from all engines. The design "up-load" during the holddown period is 3 million pounds. If one or more of the engines fail to develop full thrust, the vehicle is not released, and all engines automatically are shut down.



Night Shot—A 363-foot-tall Saturn V facilities vehicle is shown in place at Launch Pad 39A.

K-107-66PC-63

TESTING

INTRODUCTION

The expense of the Saturn V makes it imperative that no effort be spared to assure that it will perform as expected in flight. The magnitude of the Saturn V ground test program, therefore, is unprecedented. To qualify for flight, all components and systems must meet standards deliberately set much higher than actually required. This margin of safety is built into all manrated space hardware.

Compared with earlier rocket programs the ground testing on Saturn V is more extensive and the flight testing is shorter. The ground test programs conducted on the F-1 and J-2 engines, which power the three stages, offer an example of the thoroughness of this testing effort. The J-2 has been fired some 3,500 times on the ground, for a total running time of more than 97 hours. During flight of five Saturn IB and two Saturn V launch vehicles, 17 J-2 engines operated a total time of almost two hours. The F-1 has been fired more than 2,772 times for a running time of more than 64 hours. On two Saturn V launches ten F-1 engines have run for a total of about 25 minutes.

Further, in earlier rocket programs such as Redstone, Thor, and Jupiter, 30 to 40 R&D flight tests were standard. In the Saturn I program,where more emphasis was placed on ground testing prior to the flight phase, 10 R&D flight tests were planned. The vehicle was declared operational after the first six firings met with success.

The Saturn IB—an improvement on the basic Saturn I—was manrated after three flights. On the Saturn V, only two flights are planned prior to the attainment of a "manned configuration."

The inspection to which flight hardware is subjected is thorough. Following are examples of many steps which are taken to inspect the Saturn V vehicle:

1. X-rays are used to scan fusion welds, 100 castings, and 5,000 transistors and diodes.
2. A quarter mile of welding and 5 miles of tubing are inspected with the use of a sound technique (ultrasonics). The same type of inspection is given to adhesive bonds, which are equivalent in area to an acre.
3. An electrical current inspection method is used on 6 miles of tubing, and dye penetrant tests are run on 2.5 miles of welding.

Each contractor has his own test program patterned to a rather basic conservative approach. It begins with research to verify specific principles to be applied and materials to be used. After production starts each contractor puts flight hardware through qualification testing, reliability testing, development testing, acceptance testing, and flight testing.

QUALIFICATION TESTING

Qualification testing of parts, subassemblies, and assemblies is performed to assure that they are capable of meeting flight requirements. Tests under the conditions of vibration, high-intensity sound, heat, and cold are included.

RELIABILITY TESTING

Reliability analysis is conducted on rocket parts and assemblies to determine the range of failures or margins of error in each component. Reliability information is gathered and shared by the rocket industry.

DEVELOPMENT TESTING

A battleship test stage constructed more solidly than a flight stage is often used to prove major design parameters within a stage. Such a vehicle verifies propellant loading, tank and feed operation, and engine firing techniques.

Battleship testing is followed by all-systems testing. For example, one of four ground test stages of the first stage completed 15 firings at Marshall Space Flight Center in Huntsville. The firings proved that the design and fabrication of the complete booster and of its subsystems were adequate.

The entire Apollo/Saturn V vehicle, consisting of the three Saturn V propulsive stages, the instrument unit, and an Apollo spacecraft, was assembled in the Dynamic Test Stand at the Marshall Center. This is the only place, aside from the launch site, where the entire Saturn V vehicle has been assembled. The purpose of dynamic testing was to determine the bending and vibration characteristics of the vehicle to verify the control system design. The 363-foot assembly was placed on a hydraulic bearing or "floating platform". Electromechanical shakers caused the vehicle to vibrate, simulating the response expected from flight forces.

ACCEPTANCE TESTING

Finished work undergoes functional checkout to insure it meets operational requirements. Tests range from continuity and compatibility of wiring to all-systems ground testing. Fluid-carrying components are subjected to pressures beyond normal operating requirements, and structural components receive visual and X-ray inspections. Instruments simulate flight conditions to evaluate total performance of electrical and mechanical equipment.

Rocket engines are static-fired before delivery to the stage contractor. Such tests demonstrate performance under conditions simulating flight temperatures, pressures, vibrations, etc.

Each flight stage completes a series of systems tests which lead to a full-power, captive acceptance firing. Afterwards it is refurbished and given a postfiring checkout before going to Kennedy Space Center.

AUTOMATIC CHECKOUT

A fully automated, computer-controlled vehicle checkout has been designed into all major segments of the Saturn V for extensive stage test operations.

Automatic checkout is used first in the final factory checkout and then throughout prefiring preparations for static tests and during the actual countdown for these firings. It is employed again throughout the postfiring checkout and finally for prelaunch checkout and launch operations at Kennedy Space Center.

The sytem uses a carefully detailed computer program and associated electronic equipment to perform the complete countdown of each Saturn stage.

With electronic speed, the automatic checkout moves through a precisely controlled and repeatable checkout test program. The system performs a point-by-point test of each function, indicates responses to tests, and pin-points any malfunction that occurs. The automatic checkout can also indicate ways to double check a questionable response in order to define any difficulty. It virtually eliminates the possibility of human error during a countdown.

FLIGHT TESTING

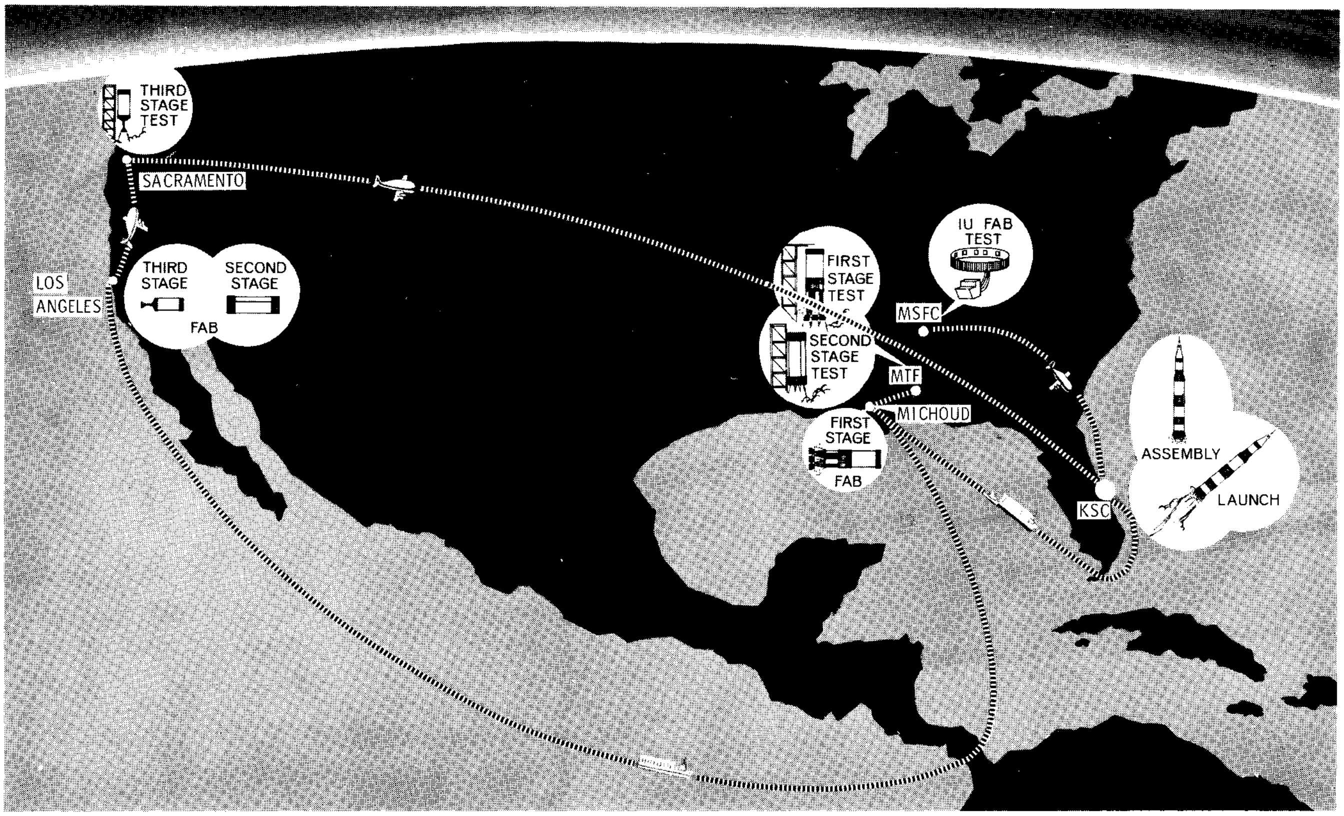
Every flight program is designed to provide a mass of vehicle performance information which is needed in planning future launches. Each stage carries a complete network of instrumentation to measure and record the performance of every system, subsystem, and vital component.

TEST DOCUMENTATION

In all Saturn V test operations, from ground development through flight, documentation of results is as important as the acquisition of data. The performance history of every part, component assembly, subsystem, and system must be accurately detailed and permanently recorded.

These records give engineers a basis for making evaluations of the performance of parts and subsystems. These evaluations provide maximum confidence in every vehicle.

The formidable task of record-keeping has necessitated the establishment of a test data bank for Saturn V program engineers. It can be an invaluable source of reference in the event of minor or major malfunctions in a test or flight.



Traveling Saturn– This depicts the Saturn V assembly and test sequence and the transportation routes of rocket-carrying craft.

VEHICLE ASSEMBLY AND LAUNCH

ASSEMBLY AND CHECKOUT

Saturn V stages are shipped to the Kennedy Space Center by ocean-going vessels or by specially designed aircraft. Apollo spacecraft modules are transported by air and delivered to the Manned Spacecraft Operations Building at Kennedy Space Center for servicing and checkout before mating with the Saturn V.

Saturn V stages go into the Vehicle Assembly Building low bay area where preparation and checkout begins. Receiving inspection and the low bay checkout operations are first performed before stages are erected within a high bay.

After being towed into the high bay area and positioned under the 250-ton overhead bridge crane, slings are attached to the first stage and hooked to the crane. The stage is positioned above the launch platform of the mobile launcher and lowered into place. Then it is secured to four holddown/support arms. These support the entire space vehicle during launch preparation and provide holddown during thrust buildup prior to launch.

Next, engine fairings are installed on the stage and fins are moved into position and installed in line with the four outboard engines.

Mobile launcher electrical ground support equipment is connected to the launch control center (LCC) via the high speed data link, and the test program is started with the actual launch control equipment.

Prior to and during this time, all low bay testing is completed and the upper stages are prepared for mating. The mating operation consists of stacking the stages. Umbilical connection begins immediately and continues during the mating operation on a noninterference basis. The vertical alignment of the vehicle is performed after each stage is mated.

When the launch vehicle is ready, the Apollo spacecraft is brought to the VAB and mated.

Checkout of all systems is performed concurrently in the high bay. The first tests provide power and cooling capability to the vehicle, validate the connections, and establish instrumentation. When this is completed, systems testing begins. The systems tests are controlled and monitored from the LCC wherever practical and "break-in" tests are held to a minimum. Following the validation of each stage, a data review is held and the vehicle is prepared for combined systems tests.

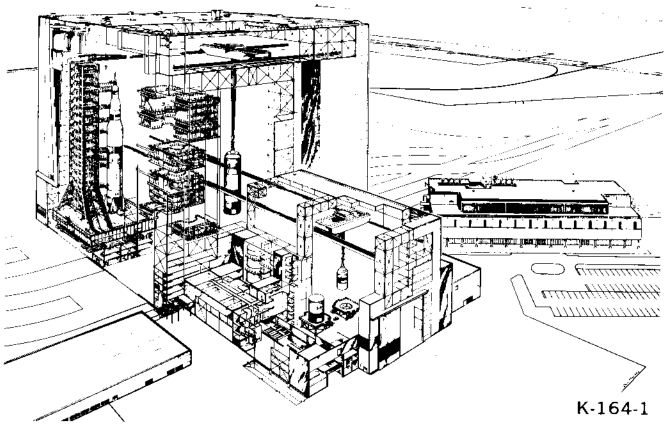


Illustration of Vehicle Assembly Building Interior at Kennedy Space Center

The combined systems tests verify the flight-readiness of the overall vehicle. These tests include a malfunction sequence test, an overall test of the launch vehicle, an overall test of the spacecraft, and a simulated flight test. Prior to the simulated flight test, final ordnance installation is completed. After the test, vertical alignment is checked, a data review is held, and the vehicle is prepared for transfer to the pad. These preparations include disconnecting pneumatic, hydraulic, and electrical lines from the mobile launcher to the VAB.

After the lines are disconnected, the transporter is moved into position beneath the mobile launcher. Hydraulic jacks engage the fittings on the mobile launcher and raise it approximately 3 feet so that it clears its mount mechanisms. Then the transporter moves out of the VAB, over the crawlerway, to the launch pad.

TESTING AT LAUNCH SITE

At the launch pad, the transporter moves the mobile launcher into position, lowers and locks it onto another set of mount mechanisms. The transporter then moves to the mobile service structure parking area, picks up the service tower, and positions it beside the Saturn V to provide vehicle access for pad operations.

The digital data link, communications circuitry, pneumatic supply lines, propellant lines, environmental controls, and electrical power supply lines are connected.

Power again is applied to the vehicle and the control and monitor links are verified. Pad testing is held to a minimum. The high bay from which the vehicle was moved remains empty during pad operations.

A spacecraft systems verification test is performed, followed by a space vehicle cutoff and malfunction test. Radio frequency compatibility is established and preparations are made for a final flight readiness test, which involves sequence tests paralleling the actual countdown and inflight operations. Compatibility with the stations of the Eastern Test Range and the Integrated Mission Control Center in Houston, Tex., are verified at this time.

Following an evaluation of the flight readiness test, all systems are reconfigured for launch, and all plugs reverified. A countdown-demonstration test is then performed as the final test prior to launch. The countdown-demonstration test consists of an actual launch countdown, complete with propellant loading, astronaut embarkation, etc., with the exception of actual ignition. This test exercises all systems, the launch crew, and the astronauts, and prepares the "team" for the actual operation to follow. This "dress rehearsal" is used to divulge any last minute problems and affords the mission a better chance of success.

Upon completion of the countdown demonstration test, the space vehicle is recycled to pre-count status, and preparations are made for the final countdown phase of launch operations. Normal recycle time between completion of the countdown demonstration test and beginning of launch countdown is 48 to 72 hours.

Propellant loading of the Apollo spacecraft is performed prior to launch day. Aerozine 50 is the fuel and nitrogen tetroxide, the oxidizer. Also prior to launch day, hypergolics for the third stage reaction control system are loaded and ordnance connected. Loading of the cryogenic propellants for the launch vehicle begins on launch day at approximately T-7 hours. (The kerosene is loaded one day before launch.)

Liquid oxygen loading is begun first. The tanks are precooled before filling. Precool of one tank can be accomplished concurrently with the fill of another. Loading is started with the second stage to 40 per cent, followed by the third stage to 100 per cent. The second stage is then brought to a full 100 per cent followed by loading the first stage to 100 per cent. This procedure allows time for the liquid oxygen leak checks to be performed prior to full loading of the second stage. Liquid oxygen is pumped at a flowrate of 1,000 gallons per minute for the third stage. For the second stage, the tank rate is 5,000 gallons per minute, and the first stage tank flowrate is 10,000 gallons per minute.

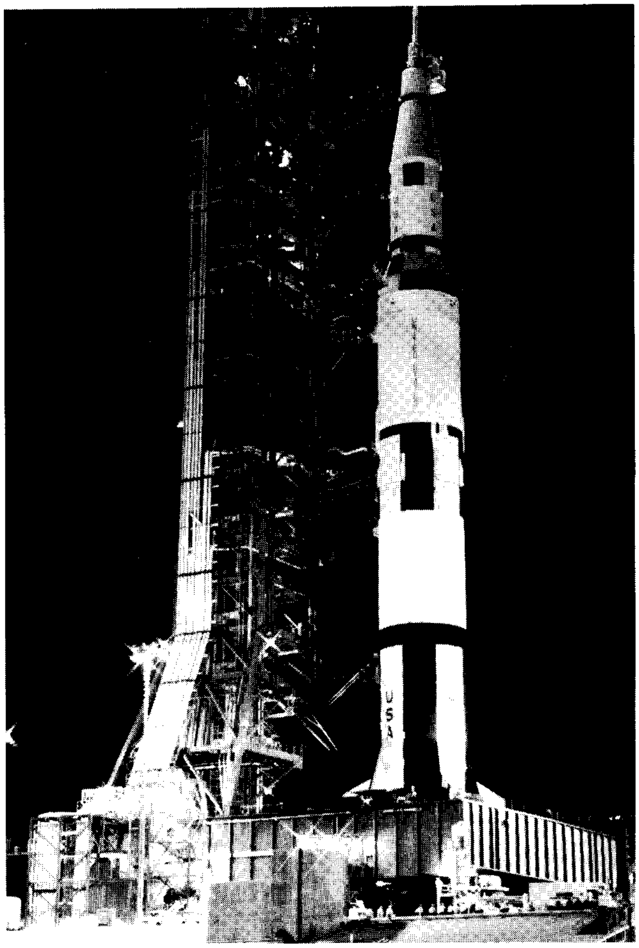
Liquid hydrogen loading is initiated next, beginning with the second stage to 100 per cent. Loading of the third stage liquid hydrogen is last. Liquid hydrogen is pumped to the second stage at a rate of 10,000 gallons per minute, and to the third stage at a rate of 3,000 gallons per minute. Topping of cryogenic tanks of the launch vehicle continues until launch. Total cryogenic loading time from start to finish is 4 hours and 30 minutes.

At approximately T-90 minutes, after propellants are loaded, the astronauts enter the spacecraft from the mobile launcher over the swing arm walkway.

LAUNCH

During the remainder of the countdown, the final systems checks are conducted.

Launch vehicle propellant tanks are then pressurized, and the first stage engines ignited. During the thrust buildup of the F-1 engines, the operation of each of these engines will be automatically checked. Upon confirmation of thrust OK condition, the launch commit signal is given to the holddown arms and liftoff occurs.



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Assembled Vehicle—The Saturn V facilities vehicle, the 500F, arrives at Launch Complex 39A.

PROGRAM MANAGEMENT

NASA ORGANIZATION

The Saturn V development program comes under the direction of the NASA Office of Manned Space Flight, Washington, D.C. That office assigned development responsibility to the Marshall Space Flight Center, one of the three Manned Space Flight field centers. Another of those field centers, the Kennedy Space Center, has been delegated the responsibility of launching the Saturn V. (Development of the Apollo spacecraft, the first "payload" for the Saturn V, was assigned to the Manned Spacecraft Center, the other MSF field organization.)

Marshall Center Project Management Organization

Tens of thousands of prime and subcontractor employees and civil service employees are working on the Saturn program. At one time the manpower level was more than 125,000. Saturn industrial activities are scattered nationwide but there are three major areas of concentration:

1. the Northeast, with its grouping of electronic industries.
2. the Southeast, for production, test, and launch operations.
3. the West Coast, with its concentration of aerospace industries for design, production, and test work.

In addition, various research projects by scientific institutions and subcontractor production efforts contributing to the Saturn program are spread throughout the nation.

The wide dispersion makes necessary very comprehensive and reliable management systems and control techniques to manage the program effectively. The geographic dispersion of the Saturn effort requires excellent communications. The Marshall Center must be aware of related programs carried out by other NASA centers—especially the Manned Spacecraft Center, managing the Apollo spacecraft program, and Kennedy Space Center, responsible for Saturn launches.

The Marshall Center has found that one of the more effective tools for total program visibility is especially constructed and outfitted rooms called Program Control Centers. The Saturn V launch vehicle program office and other major groups have such centers.

The budget for the current fiscal year at the MSFC is about $850 million. The center must have a well staffed organization responsive to the many changes which can take place in a program of this magnitude.

One of the Marshall Center's two major divisions—Industrial Operations—is responsible for the management of the Saturn launch vehicle development programs for NASA manned space flight. Lee B. James, the Saturn V program manager in Industrial Operations, controls the project effort, plans, and budgets. For technical solutions to vehicle problems, the manager gets assistance from the laboratories of the Research and Development Operations—the second major and largest MSFC division reporting directly to the center director. Because of the many interfaces between the stages and with ground support equipment, program management responsibility in Industrial Operations includes establishing specifications and procedures which assure physical and functional compatibility. Formerly, the Marshall Center did the overall design of stages and major systems inhouse, but more recently, particularly with subsystems and components, the program managers have concentrated on performance specifications and left the details to the contractors. This management function keeps the program engineers very much in the mainstream of technical design activity. Thus, Industrial Operations program managers are quite active in the areas of: test requirements, qualification testing, product control, systems engineering, program control, and flight operations.

Marshall Center's Research and Development Operations laboratories are oriented functionally in such primary disciplines as mechanical engineering, electronics, and flight mechanics. Collectively, the laboratories provide the deep-rooted technological foundation on which the success of all Marshall projects depends. In the project offices, technical decisions are made which affect many areas. These decisions are formulated by drawing upon the full technical resources of the laboratories, which maintain a high level of professional competence.

Laboratory personnel work on selected projects to keep their technical knowledge updated and their technical competence at a high pitch. This is the Marshall work bench philosophy—the "dirty hands" approach.

The Saturn V program office is headed by a program manager. There is a stage manager or project director for each of five major vehicle systems. A stage manager primarily deals with only one major contractor. In the case of the instrument unit and the ground support equipment project, there are several major contractors. The principle of a single project management focal point is the objective of each project team.

Program management is vested in the program manager. Technical project management, so far as NASA is concerned, occurs at the stage or project level. The program and stage managers are fully responsible for technical adequacy, reliable performance, and for management of all related contractor activity.

These program and project managers must be backed up and supported by technical competence in depth. This in-depth support is provided, to a degree, by a staff of competent technical and business management people in the program manager and stage manager office, and to a much larger degree, by Research and Development Operations.

There is a resident manager at each of the contractor plants to act as the "official" voice for the Marshall Center. All MSFC instructions to the contractor are transmitted through the resident manager. Through the resident manager, MSFC maintains a direct contact with contractor operations and is kept informed of the status of all significant program events.

Marshall Center laboratory technical personnel are assigned to the resident managers' staffs. These technical people are assigned to each resident manager's office to provide him with assistance in resolving technical problems, and to keep the MSFC technical laboratories directly informed of field technical effort. Laboratory participation is dictated by need as determined by project management.

Many people are involved in attaining the final goal. Project management, technical, and contractor personnel are tied in a close knit group capable of managing this country's large launch vehicle program.

MANAGEMENT PERSONNEL

NASA

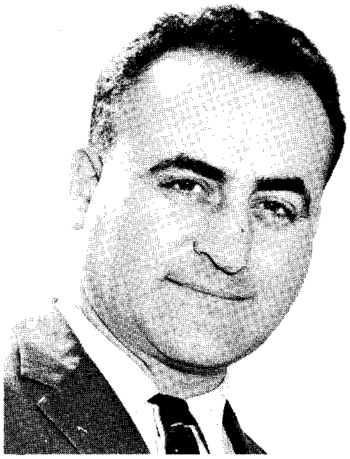
Dr. George E. Mueller, Associate Administrator for Manned Space Flight, NASA Headquarters.

Lt. Gen. Samuel C. Phillips, Director, Apollo Program, NASA Headquarters.

Dr. Wernher von Braun, Director, Marshall Space Flight Center.

Dr. Kurt H. Debus, Director of John F. Kennedy Space Center.

Lee B. James, Manager, Saturn V Program Office, Marshall Space Flight Center.



Rocco A. Petrone, Director of Launch Operations, Kennedy Space Center.

MANAGEMENT PERSONNEL

BOEING

G. H. Stoner, Group Vice President, Aerospace.

R. H. Nelson, General Manager, Launch Systems Branch.

L. D. Alford, Manager, Huntsville.

H. D. Gunning, Manager, Michoud.

F. L. 'Bud' Coenen, Director, Boeing Atlantic Test Center.

MANAGEMENT PERSONNEL

MCDONNELL DOUGLAS

Charles R. Able, Chairman and Chief Executive Officer, McDonnell Douglas Astronautics Company.

Jack L. Bromberg, Vice President, Deputy General Manager, McDonnell Douglas Astronautics Company—Western Division.

Theodore D. Smith, Director, Huntington Beach Development Engineering, McDonnell Douglas Astronautics Company—Western Division.

Steven D. Truhan, Director, Florida Test Center for McDonnell Douglas Astronautics Company—Western Division. Directs and coordinates all Company activities at Kennedy Space Center.

Harold E. Bauer, Director, Saturn/Apollo Programs, McDonnell Douglas Astronautics Company—Western Division. Responsibility for all aspects of the present development program on the S-IVB upper stages of the Saturn IB and Saturn V launch vehicles.

A. P. O'Neal, Director, Saturn Development Engineering. McDonnell Douglas Astronautics Company—Western Division.

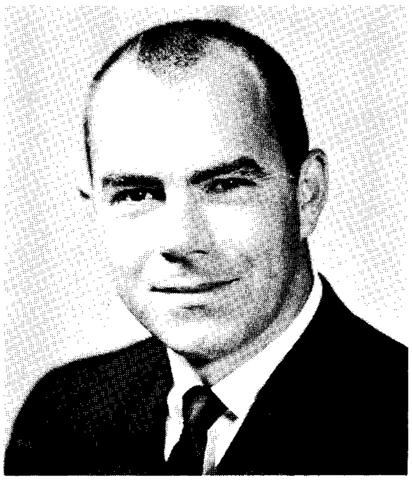
MANAGEMENT PERSONNEL

IBM

Bob O. Evans, President of the Federal Systems Division.

Arthur E. Cooper, Federal Systems Division Vice President and General Manager, Space Systems Center, Bethesda, Md.

Clinton H. Grace, Facility Manager, Space Systems Center, Huntsville, Ala.



Ammon G. Belleman, Facility Manager, Space Systems Center, Kennedy Space Center.

MANAGEMENT PERSONNEL

SPACE DIVISION —NORTH AMERICAN

William B. Bergen, Vice President, North American Aviation, Inc.; President, Space Division. Downey, Calif.

Robert E. Greer, Vice President, Space Division; Program Manager, Saturn Second Stage.

William F. Parker, Deputy Program Manager, Saturn Second Stage.

Bastian 'Buz' Hello, Vice President and General Manager, Launch Operations, Space Division, Florida.

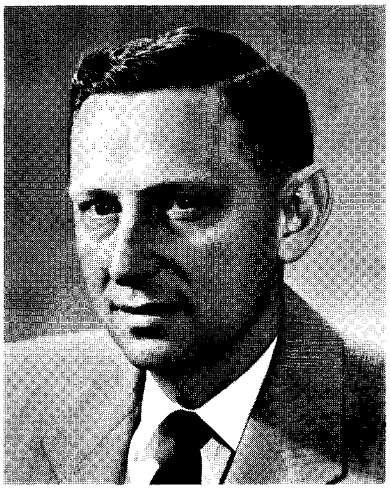
MANAGEMENT PERSONNEL

ROCKETDYNE—NORTH AMERICAN

Samuel K. Hoffman, President of Rocketdyne and Vice President of North American Aviation, Inc.

William J. Brennan, Vice President and General Manager, Liquid Rocket Division.

Paul D. Castenholz, Program Manager, J-2 Engine.



Norman C. Reuel, Assistant General Manager, Liquid Rocket Division.

David E. Aldrich, Program Manager, F-1 Engine.

FLIGHT HISTORY

AS-501 (APOLLO 4)

The first Apollo/Saturn V launch vehicle, the AS-501, performed all of its vehicle mission objectives. The vehicle was launched at 7 a.m. EST on November 9, 1967 from Launch Complex 39 at the NASA-Kennedy Space Center, Fla. The countdown proceeded smoothly and the launch came exactly on time. All vehicle systems and subsystems performed "nominally" and ground support equipment performance was satisfactory.

Prime mission objectives, with respect to the rocket, included an all-up test of the vehicle with its three stages and instrument unit, the first in-orbit restart of the third (S-IVB) stage, and the first use of Launch Complex 39 and ground support equipment.

Flight of the three stages was near nominal. The trajectory was near the expected and all three propulsion systems performed with no apparent anomalies. The instrument unit systems were all stable during the flight.

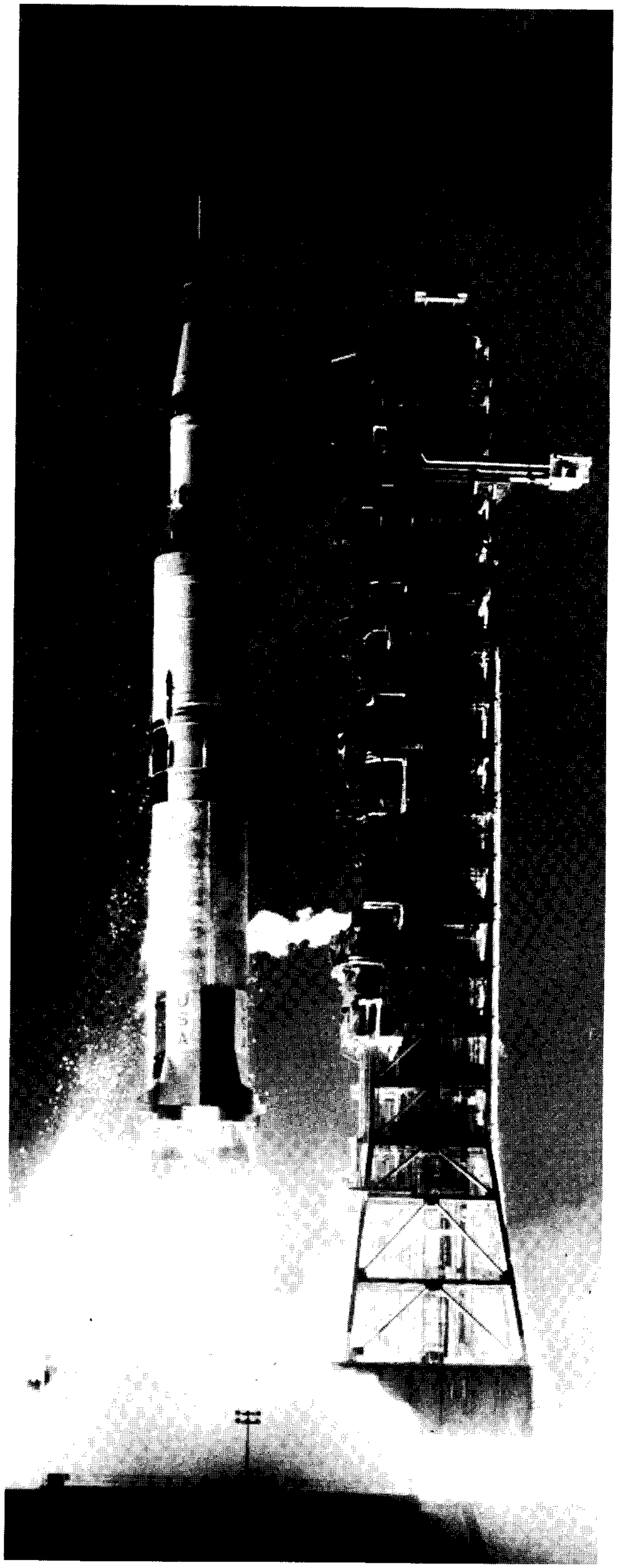
First (S-IC) stage flight was near the expected. S-IC center F-1 engine cutoff was given by a timer at 135.5 seconds. S-IC outboard engine cutoff came by liquid oxygen depletion at 150.8 seconds with the vehicle at 38.3 miles altitude traveling at 6,024.6 miles per hour. Booster and second (S-II) stage first and second plane separations each occurred within 1.2 seconds of the predicted times. Cameras on the S-II photographed a smooth separation.

Propulsion and other systems, including propellant utilization, the pressurization, and the pneumatic control pressure system, operated within expected tolerances.

The S-II engines, stage propellant utilization system, pressurization system, pneumatic control pressure system, camera ejection system, and the helium injection system operated properly and within expected tolerances.

All five J-2 engines operated properly during engine start and burn. Ground controllers noted that the thrust chamber jacket temperature heat-up rate was slightly higher than predicted, and the engine start bottle pressures were slightly higher than predicted but both were within required limits. S-II stage cutoff came at 519.8 seconds, 3.5 seconds later than predicted. The S-II stage's liquid hydrogen tank insulation performed satisfactorily with no defects noted during countdown or in flight.

The third (S-IVB) stage first and second burns were 6.2 seconds longer and 15.2 seconds shorter,



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Apollo 4 (AS-501)Launch, November 9, 1967 respectively, than predicted. The first burn began at 520.7 seconds. The J-2 engine was cut off by the guidance system at 665.6 seconds. This was 9.6 seconds later than expected. The vehicle was traveling 17,428.2 miles per hour and was at an altitude of 118.6 miles.

The S-IVB was reignited over the eastern United States after two revolutions in earth orbit. The second burn operation was cut off by guidance and was 15.2 seconds shorter than predicted, which was attributed primarily to 37 seconds of burn time at the high thrust level operation of the J-2 engine during second burn. A low liquid hydrogen ullage pressure reading was recorded at the Kennedy Space Center immediately before J-2 engine restart. The reading was 28 pounds and the expected minimum pressure was 31 pounds. This had no effect on engine operation.

The pressure in the helium repressurization spheres was apparently lower than expected during S-IVB restart preparations but reignition was achieved without difficulty.

Hydraulic systems on all three stages performed without evidence of out-of-tolerance conditions. Maximum engine deflection was 0.6 degree on the S-IC and 0.8 degree on the S-11.

Structurally, the Saturn V vehicle performed with no problems. Maximum bending occurred between 70 and 80 seconds. Longitudinal loads were near nominal throughout flight, and longitudinal acceleration at S-IC center engine cutoff was 4.15 G, which was very near the expected value.

The instrument unit on this flight was the first to be flight tested since an external structural stiffener was added to reduce vibration effects on the inertial platform. Vibrations in the area were lower than those on previous flights of the Saturn IB. The instrument units on the Saturn IB and Saturn V are essentially the same.

Telemetry taken during the first 560 seconds of powered flight showed guidance and control to be nominal.

The emergency detection system was flown "open loop" on this flight. All indications were that the system operated satisfactorily. The EDS was developed for the manned Apollo flights so that astronauts and ground controllers could know of impending troubles in the rocket in time to take corrective action.

Apollo 4 experienced only a few measurement failures. Two known measurement failures and 40 questionable measurements were identified out of the approximately 2,862 taken on the flight. This is a loss of less than two per cent.

Both onboard cameras viewing first and second stage separation recorded excellent quality pictures. The cameras were recovered shortly after being ejected into the Atlantic Ocean.

AS-502 (APOLLO 6)

The second Saturn V launch vehicle, AS-502, was not totally successful although it achieved most of its objectives and placed more than 264,000 pounds into earth orbit. The vehicle was launched from Complex 39 at the NASA-Kennedy Space Center on April 4, 1968. The launch occurred on schedule at 7 a.m. EST after a smooth countdown.

The first (S-IC) stage performed as planned and hydraulic system performance was satisfactory. Stage thrust was essentially the same as predicted during the first portion of the flight. However, a longitudinal oscillation ("Pogo" effect), measured at five cycles per second, was experienced during the latter portion of first stage burn. The phenomenon was also noted on the first Saturn V flight, AS-501, but on AS-502 it was much greater.

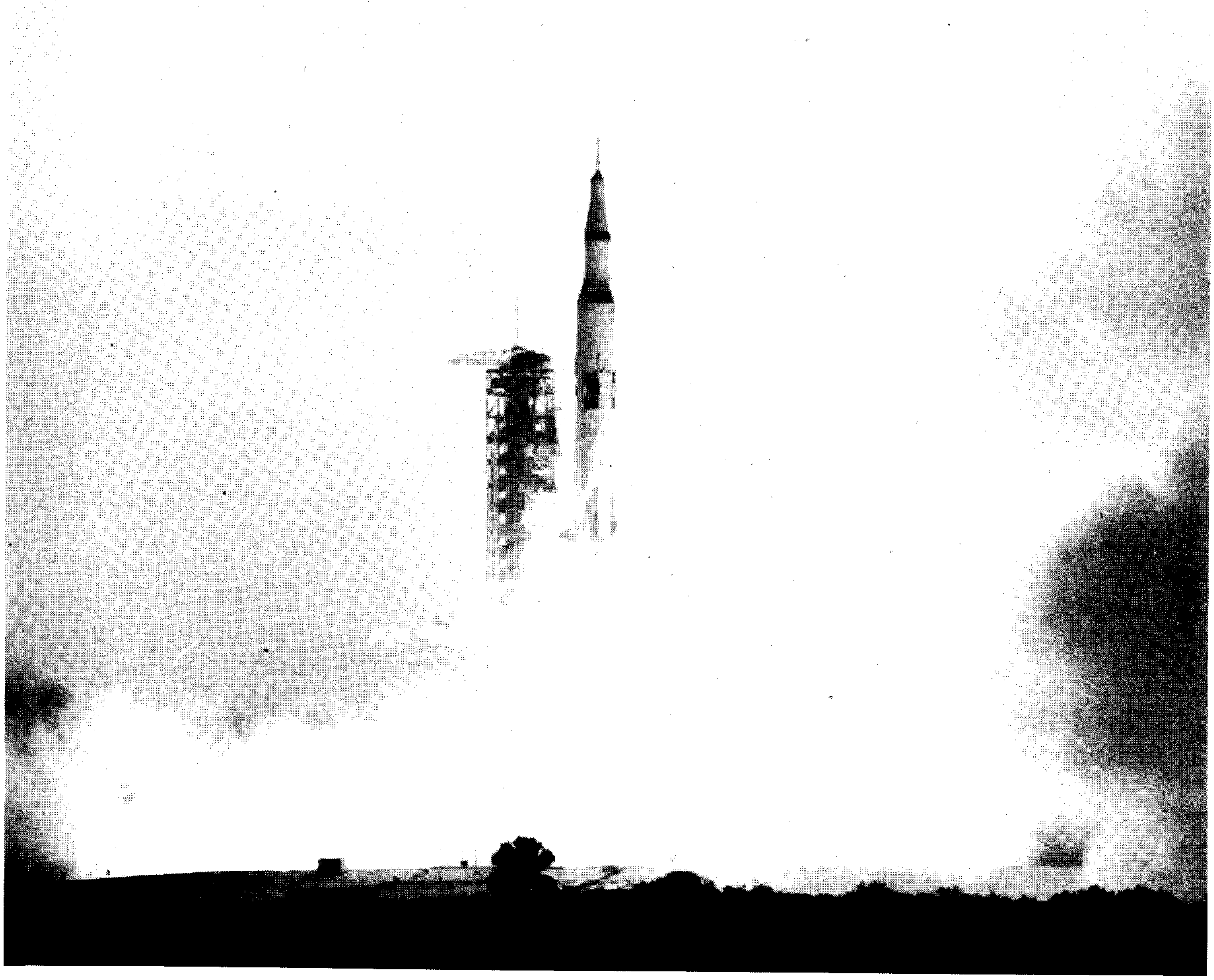
Second stage engines numbered 2 and 3 cut off prematurely at 408.7 and 410 seconds after liftoff, respectively, causing a 58-second longer than normal second stage burn and larger than expected deviations from second stage flight end conditions.

S-II performance was satisfactory through first stage boost, S-II ignition and the early portion of S-II powered flight. The earliest observed deviations were decreasing temperatures on the main oxidizer valve and its control line on engine number 5 and a steady increase in engine number 2 yaw actuator pressure, occurring at 278.4 seconds.

A sudden 5,000 pound thrust decrease and other deviations at 318 seconds preceded a cutoff signal to engine number 2. That cutoff signal also caused engine number 3 to shut down, because the wires carrying cutoff commands to engines numbered 2 and 3 were interchanged.

Hydraulic system performance was satisfactory on the second stage until about 140 seconds before premature shutdown of the two engines. At this time the increase in the yaw and pitch actuator differential pressures occurred.

First burn of the third (S-IVB) stage was 29.2 seconds longer than planned to compensate for the early cutoff of the two second stage engines. The result was a high cutoff velocity and an elliptical parking orbit. The attainment of this orbit was a demonstration of the unusual flexibility designed into the Saturn V.



Apollo 6 (AS-502)Launch, April 4, 1968

All engine and stage restart conditions appeared normal but the S-IVB's J-2 engine did not restart in orbit. The restart was to have propelled the S-IVB and Apollo spacecraft into a simulated translunar trajectory.

The third stage performed satisfactorily through first burn and orbital coast. Shortly after orbit insertion a cold helium supply leak was observed but bottle pressure was sufficient to meet second burn requirements. Even though normal engine and stage prestart conditions were observed, the engine received the start signal and the engine valves opened properly, the engine did not reignite.

Study of data relating to the S-IVB reignition problem indicated a leak in one of the two propellant lines leading to the J-2 engine's augmented spark igniter (ASI). In such a case, propellants reaching the spark plugs were insufficient, or inadequate in mixture, to achieve the proper start conditions.

Third stage hydraulic system performance was normal through first burn. Shortly before spacecraft separation, a programmed command to initiate the auxiliary hydraulic pump was given but the pump failed to operate. Ground commands after spacecraft separation also failed to start the system. Pump operation was not a requirement for engine restart.

Guidance and other instrument unit functions were satisfactory. Flight profile was nominal up to the loss of engine number 2 on the second stage. At second stage cutoff the altitude was high and velocity low. This led to a longer burn of the third stage and a velocity slightly higher than normal, causing the third stage and spacecraft to go into an elliptical orbit.

Prior to launch 29 measurements were waived. During flight there were nine known failures and 19 questionable measurements of the approximately

2,800 measurements planned originally. Telemetry performance was good on all links.

Onboard television cameras gave good data. Only two of the six on-board film cameras were recovered. Both these cameras viewed the separation of the first and second stages.

A study of data relating to the failure of the number 2 J-2 engine on the second stage and the single J-2 on the third stage indicated that in each case a propellant line leading to the engine's augmented spark igniter (ASI) ruptured. Those lines have been redesigned to remove the flexible sections where the breaks occurred. The new lines have been tested and proven adequate with a sufficient safety margin. All J-2's in the future will use the new lines.

The oscillations in the first stage also prompted an extensive investigation which led to the decision to create "shock absorbers" in the large liquid oxygen (LOX) lines leading to four of the five F-1 engines. This was done by injecting helium into cavities in the existing LOX prevalves to damp out LOX surges.

APPENDIX—GLOSSARY

The following list defines acronyms, abbreviations, nomenclature, and other terminology used in the Saturn V News Reference.

TERM	DESCRIPTION
APS	Auxiliary propulsion system
Bulkhead	A dome-shaped segment which encloses the end of a propellant tank.
Burnout	Point at which engines shut down due to lack of fuel or oxidant.
Burst Diaphragm	A disc designed to rupture at a predetermined pressure differential.
Bus	A main circuit for transfer of electrical current.
Cavitation	The formation of bubbles in a liquid, occurring whenever the static pressure at any point in the fluid flow becomes less than the fluid vapor pressure.
Convection	Mass motions within a fluid
Cryogenic	Ultra-low temperature
DDAS	Digital data acquisition system
Exhaust Nozzle	The lower section of the thrust chamber of a liquid rocket engine.
Expansion Area Ratio	The ratio of the measurements of an engine nozzle exit section to that of the nozzle throat area.
Exploding Bridgewire	Wire which explodes when subjected to a high voltage, high energy pulse.
Fusion Weld	To join two pieces of metal together by bringing the surfaces to a molten state by electric arc or gas flame controlled to produce a localized union through fusion or recrystallization across the interface.
Gimbal	A device on which a reaction engine may be mounted and which allows for angular movement in two directions.
GOX	Gaseous oxygen
GSE	Ground support equipment
Hydrostatic Test	Use of water for pressure test of propellant containers.
Hypergolic Liquids	Liquids that ignite spontaneously when mixed with each other.
Impeller	A device that imparts motion to a fluid or air.
Inducer	A pump which increases the pressure and motion of a fluid.
KSC	Kennedy Space Center
LH_2	Liquid hydrogen
LOX	Liquid oxygen
LVDA	Launch vehicle data adapter
LVDC	Launch vehicle digital computer
Monocoque	A structure in which all or most of the stresses are carried by the skin.
MSC	Manned Spacecraft Center
MSFC	Marshall Space Flight Center
Multiplexer	A mechanical or electrical device for time sharing of a circuit.
NASA	National Aeronautics and Space Administration
ODOP	Offset Doppler System
Pitch	Movement of the vehicle from its lateral axis.
PSI	Pounds per square inch
PSIA	Pounds per square inch absolute
PSIG	Pounds per square inch gage
Purge	To remove residual fluid or gas.

TERM	DESCRIPTION
Retrorocket	A rocket fitted to a stage to produce thrust opposed to the stages forward motion.
RF	Radio frequency
RJ-I	A grade of kerosene which is used in the hydraulic system prior to lift-off.
Roll	The rotation of a vehicle about its axis.
RP-I	A rocket fuel consisting essentially of kerosene.
Squib	An explosive device used in the ignition of a rocket engine. Usually called an igniter.
Stator	A mechanical part that remains stationary with respect to a rotating or moving part or assembly.
Thermocouple	A device which converts thermal energy directly into electrical energy.
Thrust	The force developed by a rocket engine.
Thrust Vectoring	An attitude control for rockets wherein one or more engines are gimbal-mounted so that the direction of the thrust force may be changed in relation to the center of gravity of the vehicle to produce a turning movement.
Torus	A circular duct (manifold) used to collect fluid or gases.
Ullage	The amount that a container, such as a fuel tank, lacks of being full.
Umbilical	Any of the servicing lines between the ground or tower and a launch vehicle.
Volute	A flow passage that collects and redirects fluids.
Yaw	Movement of a vehicle from its longitudinal axis.

APPENDIX A—SATURN V SUBCONTRACTORS

The following are lists of subcontractors who have played a major role in the development and production of the Saturn V launch vehicle. It should be recognized that many more subcontractors contributed to the total vehicle and program; however, it is not practical to list all in this document.

BOEING MAJOR SUBCONTRACTORS

SUBCONTRACTOR	LOCATION	PRODUCT
Aeroquip Corp.	Jackson, Mich.	Couplings, pneumatic, and hydraulic hoses
Aircraft Products	Dallas, Tex.	Machined parts
AiResearch Manufacturing Co.	Phoenix, Ariz.	Valves
Applied Dynamics, Inc.	Ann Arbor, Mich.	Analog computers
Arrowhead Products, Div. of Federal-Mogul Corp.	Los Alamitos, Calif.	Ducts
The Bendix Corp., Pioneer-Central Div.	Davenport, Iowa	Loading systems and cutoff sensors
Bourns, Inc. Instrument Div.	Riverside, Calif.	Pressure transducers
Brown Engineering Co., Inc.	Lewisburg, Tenn.	Multiplexer equipment
The J. C. Carter Co.	Costa Mesa, Calif.	Solenoid valves
Consolidated Controls Corp.	Bethel, Conn. Los Angeles, Calif.	Pressure switches, transducers, and valves
The Eagle-Picher Co., Chemical and Metals Div.	Joplin, Mo.	Batteries
Electro Development Corp.	Seattle, Wash.	AC and DC amplifiers
Flexible Tubing Corp.	Anaheim, Calif.	Ducts
Flexonics, Div. of Calumet and Hecla, Inc.	Bartlett, Ill.	Ducts
General Precision, Inc. Link Ordnance Div.	Sunnyvale, Calif.	Propellant dispersion systems
Gulton Industries, C. G. Electronics Div.	Albuquerque, N. M.	Wiring boards
Hayes International Corp.	Birmingham, Ala.	Auxiliary nitrogen supply units
Hydraulic Research and Manufacturing Co.	Burbank, Calif.	Servoactuators and filter manifolds
Johns-Manville Sales Corp.	Manville, N. J.	Insulation
Kinetics Corporation of California	Solano Beach, Calif.	Power transfer switches
Ling-Temco-Vought, Inc.	Dallas, Tex.	Skins, emergency drains, and heat shield curtains
Marotta Valve Corp.	Boonton, N. J.	Valves
Martin Marietta Corp.	Baltimore, Md.	Helium bottles
Moog, Inc.	East Aurora, N. Y.	Servoactuators
Navan Products, Inc.	El Segundo, Calif.	Seals
Parker Aircraft Co.	Los Angeles, Calif.	Valves
Parker Seal Co.	Culver City, Calif.	Seals
Parsons Corp.	Traverse City, Mich.	Tunnel assemblies

BOEING MAJOR SUBCONTRACTORS (continued)

SUBCONTRACTOR	LOCATION	PRODUCT
Precision Sheet Metal, Inc.,	Los Angeles, Calif.	Filter screens, anti-vortex, and adapter assemblies
Purolator Products, Inc., Western Div.	Newbury Park, Calif.	Umbilical couplings
Kandall Engineering Co.	Los Angeles, Calif.	Valves
Raytheon Co.	Waltham, Mass.	Cathode ray tube display system
Rohr Corp.	Chula Vista, Calif.	Heat shields
Servonic Instruments, Inc.	Costa Mesa, Calif.	Pressure transducers
Solar, Division of International Harvester	San Diego, Calif.	Ducts
Southwestern Industries, Inc.	Los Angeles, Calif.	Calips pressure switches
Space Craft, Inc.	Huntsville, Ala.	Converters
Stainless Steel Products, Inc.	Burbank, Calif.	Ducts
Standard Controls, Inc.	Seattle, Wash.	Pressure transducers
Statham Instruments, Inc.	Los Angeles, Calif.	Pressure transducers
Sterer Engineering and Manufacturing Co.	Los Angeles, Calif.	Valves
Stresskin Products Co.	Costa Mesa, Calif.	Insulation
Systron-Donner Corp.	Concord, Calif.	Servo accelerometers
Thiokol Chemical Corp., Elkton Div.	Elkton, Md.	Retrorockets
Trans-Sonics, Inc.	Burlington, Mass.	Measuring systems and thermometers
Unidynamics/St. Louis, A Division of UMC Industries, Inc.	St. Louis, Mo.	Spools, harnesses and ducts
United Control Corp.	Redmond, Wash.	Ordnance devices and control assemblies
Vacco Industries	South El Monte, Calif.	Filters, relief valves, and regulators
Whittaker Corp.	Chatsworth, Calif.	Valves and gyros
Fred D. Wright Co., Inc.	Nashville, Tenn.	Support assemblies and measuring racks

DOUGLAS MAJOR SUBCONTRACTORS

SUBCONTRACTOR	LOCATION	PRODUCT
Accessory Products Co., Div. of Textron, Inc.	Whittier, Calif.	Valves/heaters
Aeroquip Corp., Marman Div.	Los Angeles, Calif.	Clamps
Airdrome Parts Co.	Inglewood, Calif.	Fittings
AiResearch Mfg. of Ariz.	Phoenix, Ariz.	Valves
Airtex Dynamics, Inc.	Compton, Calif.	Tank assemblies
Amco Engineering Co.	Chicago, Ill.	Cabinets
American Electronics, Inc.	Fullerton, Calif.	Batteries
Amp, Inc.	Harrisburg, Pa.	Electrical panels
Ampex, Corp.	Los Angeles, Calif.	Tape recorders
Amphenol Borg Electronics Corp.	Chicago, Ill.	Connectors
Anaconda Metal Hose Div.	Waterbury, Conn.	Metal hose

DOUGLAS MAJOR SUBCONTRACTORS (continued)

SUBCONTRACTOR	LOCATION	PRODUCT
Astrodata, Inc.	Anaheim, Calif.	Telemetry
Avnet Corp.	Westbury, L. I., N. Y.	Electrical connectors
Barry Controls	Watertown, Mass.	Electronic controls
Bertea Products	Pasadena, Calif.	Transmitter Fabricated assemblies
Brown Engineering Co., Inc.	Huntsville, Ala.	Telemetry equipment
Calmec Mfg. Co.	Los Angeles, Calif.	Valves
Capital Westward, Inc.	Paramount, Calif.	Filters
Christie Electric Corp.	Los Angeles, Calif.	Meters
Consolidated Electrodynamics	New York, N. Y.	Electronic equipment
Data Sensors	Gardena, Calif.	Transducers
Deutsch	Los Angeles, Calif.	Fittings
Dynatronics, Inc.	Orlando, Fla.	Telemetry equipment
Eagle-Picher Co.	Joplin, Mo.	Batteries
Electra Scientific Corp.	Fullerton, Calif.	Transducers
Electrada Corp.	Culver City, Calif.	Electrical components
Fairchild Camera Inst. Corp.	Plainview, L. I., N. Y.	Cameras, oscilloscopes
Fairchild Controls Div.	Montebello, Calif.	Valves
Fairchild Hiller Corp.	Bayshore, L. I., N. Y.	Valves
Fairchild Semiconductor	Hollywood, Calif.	Semiconductors
Fairchild Stratos	Bayshore, L. I., N. Y.	Valves
Flexible Metal Hose Mfg. Co.	Costa Mesa, Calif. Northridge, Calif.	Metal hose
Flomatics, Inc.	Natoma, Calif.	Valves
Frebank Co.	Glendale, Calif.	Switches
Control Data Corp.	Minneapolis, Minn.	Computers
General Electric Co.	Waterford, N. Y.	Electrical components
Giannini Controls Corp.	Durate/Pasadena, Calif.	Transducers
Grove Valve Regulator Co.	El Segundo/Oakland, Calif.	Valves
Hadley, B. H. Co.	Pomona, Calif.	Relays
Hewlett Packard Co.	Pasadena, Calif.	Oscilloscopes/recorder
Hexcel Products, Inc.	Berkeley, Calif.	Honeycomb panels
Honeywell, Inc.	Minneapolis, Minn.	Temperature controls, gyros, sensors
ITT Wire and Cable	Clinton, Mass.	Wire and cable
K-Tronics	Los Angeles, Calif.	Semiconductors
Kaiser Aluminum Chemical Sales, Inc.	Spokane, Wash. Halelhrope, Md.	Raw material, cable
Kinetics, Corp.	Solano Beach, Calif.	Electronic equipment
Ladewig Valve	Los Angeles, Calif.	Valves
Lanagan, W. M. Co., Inc.	Costa Mesa, Calif.	Valves
Leonard, Wallace O., Inc.	Pasadena, Calif.	Valves
Linde Co., Div. Union Carbide	El Monte, Calif.	Batteries
Litton Industries of Calif.	Beverly Hills, Calif.	Transducers
Magnesium Alloy Prods. Co.	Compton, Calif.	Castings
Magnetika, Inc.	Venice, Calif.	Batteries

DOUGLAS MAJOR SUBCONTRACTORS (continued)

SUBCONTRACTOR	LOCATION	PRODUCT
Marotta Valve Corp.	Santa Ana, Calif.	Valves
Marshall, G. S. Co.	San Marino, Calif.	Electronic hardware
Mason Electric Div. Ansul.	Los Angeles, Calif.	Switches
Master Specialties Co.	Los Angeles, Calif.	Meters
Menasco Mfg. Co.	Burbank, Calif.	Fabricated assemblies
Military Products Div., Clary	San Gabriel, Calif.	Elec. fittings
Moog Servo Controls, Inc.	Aurora, N.Y.	Valves
Motorola Semiconductor Products	Hollywood, Calif.	Semiconductors
Pacific Valve, Inc.	Long Beach, Calif.	Valves
Parker Aircraft Co.	Los Angeles, Calif.	Fittings/valves
Pesco Products Div., Borg Warner	Bedford, Ohio	Pumps
Philco Corp.	Philadelphia, Pa.	Radio equipment
Planautics Corp.	Solano Beach, Calif.	Switches
Potter Brumfield	Princeton, Ind.	Switches, relays
Purolator Products, Inc., Western Div.	Van Nuys, Calif.	Filters
Reynolds Metals Co.	Birmingham, Ala.	Raw material (alum.)
Rosemount Eng. Co.	Minneapolis, Minn.	Temperature controls
Sandorn Co.	Waltham, Mass.	Recorders
Scintilla Div. Bendix	Sidney, N.Y.	Electrical connectors
Sealol Corp.	Providence, R.I.	Valves
Servonic Instruments, Inc.	Costa Mesa, Calif.	Transducers
Signet Scientific	Burbank, Calif.	Ovens
Snap Tite	Union City, Pa.	Connectors
Sperry Gyro Co. Div., Sperry Rand	Great Neck, L.I., N.Y.	Instruments
Stainless Steel Products	Burbank, Calif.	Flexible ducts
Statham Instruments, Inc.	Los Angeles, Calif.	Transducers
TRW, Inc.	Cleveland, Ohio	Attitude control rocket engines
Trans-Sonics, Inc.	Lexington, Mass.	Temperature controls
Technology Instruments Corp.	Newbury Park, Calif.	Potentiometers
Telemetrics, Inc., Subsidiary of Arnoux Corp.	Santa Ana, Calif.	Ground support electronics
Texas Instruments, Inc.	Dallas, Tex.	Resistors/transistors
U.S. Steel Corp. U.S. Steel, Supply Div.	Seattle, Wash.	Raw material
Vacco Valve Company	El Monte, Calif.	Valves
Vickers	Detroit, Mich.	Pumps
Vinson Mfg. Co., Inc.	Van Nuys, Calif.	Valves
W & S Industries	El Monte, Calif.	Connectors
Winsco Instruments Control	Santa Monica, Calif.	Temp. control units
Wyman Gordon Co.	Worcester, Mass.	Forgings

IBM MAJOR SUBCONTRACTORS

SUBCONTRACTOR	LOCATION	PRODUCT
Aerodyne Controls Corp.	Farmingdale, N. Y.	First stage regulator Gas bearing pressure regulator

IBM MAJOR SUBCONTRACTORS (continued)

SUBCONTRACTOR	LOCATION	PRODUCT
Airtek Div., Fansteel Metallurgical Corp.	Compton, Calif.	2-cubic-foot bottle 165-cubic-inch sphere
Applied Microwave Laboratory, Inc.	Andover, Mass.	T/M power divider
Astro Space Laboratories, Inc.	Huntsville, Ala.	Bleeder assembly Union orifice assembly
Automatic Metal Products Corp.	Brooklyn, N. Y.	Coaxial switch
AVCO Corp.	Nashville, Tenn. Huntsville, Ala.	Cable trays Gas bearing mounting panel Measuring range cards Measuring range cards software Thermal conditioning panels
Avion Electronics, Inc.	Paramus, N. J.	Telemetry directional coupler VSWR measuring assembly
Bourns, Inc.	Riverside, Calif.	Transducer pressure gauges
Brown Engineering Co., Inc.	Huntsville, Ala.	Acquisition system Alternating current amplifier assembly Measuring rack selector Measuring rack assembly Mod 145 multiplexers Mod 245 multiplexers Mod 270 multiplexers Pulse code modulated digital data Radio frequency analog Radio frequency pulse code modulated Telemetry assembly A3 Telemetry assembly B1 Telemetry assembly SSB Telemetry calibrators Telemetry pulse code
Chrysler Corp.	Huntsville, Ala.	Q-ball Vehicle plate assembly
Conic Corp.	San Diego, Calif.	UHF transmitter
Cox Instruments Div., Lynch Corp.	Detroit, Mich.	Flowmeters
Eagle-Picher Industries, Inc.	Joplin, Mo.	Primary battery
Electro Development Corp.	Seattle, Wash.	Amplifier direct current Channel selector A Channel selector B
Electronic Communications, Inc.	St. Petersburg, Fla.	Control computer
EIMAC Div., Varian	San Carlos, Calif.	UHF transmitter
The Foxboro Co.	Foxboro, Mass.	Gas temperature probe
Fenwall Electronics, Inc.	Framingham, Mass.	Temperature gauges
Flodyne Controls, Inc.	Linden, N. J.	Shut-off ball valve
Gulton Industries, Inc.	Hawthorne, Calif.	Electrical cables Vibration accelerometers 5-volt power supply
Hamilton-Standard Div., United Aircraft Corp.	Windsor Locks, Conn.	Preflight heat exchanger
Hayes International Corp.	Huntsville, Ala.	Network cables
Hydro-Aire Div., Crane	Burbank, Calif.	Coolant pump, R&D
ITT Wire and Cable Div., International Telephone and Telegraph Corp.	Clinton, Mass.	Wire Cables Coaxial cables

IBM MAJOR SUBCONTRACTORS (continued)

SUBCONTRACTOR	LOCATION	PRODUCT
Marotta Valve Corp.	Boonton, N. J.	Shut-off valve Solenoid valve
Martin Co.	Orlando, Fla.	Control signal processor
Melpar, Inc.	Falls Church, Va.	C-band antenna Command antenna Command directional coupler Command power divider Telemetry antenna Telemetry power divider
Motorola, Inc.	Scottsdale, Ariz.	C-band transponder Command receiver
North American Aviation, Inc.	Tulsa, Okla.	Structure segments
Nortronics Div., Northrop Corp,	Norwood, Mass.	Rate gyro package
Ralph M. Parsons Electronics Corp.	Pasadena, Calif.	Tape recorder
Perkin-Elmer Corp.	Norwalk, Conn.	Retroreflector
Potter Aeronautical Corp.	Union, N. J.	2 flowmeter
Purolator Products, Inc.	Los Angeles, Calif.	Gas bearing pressure regulator Quick disconnect coupling
Rantec Corp.	Calabasas, Calif.	Telemetry RF coupler
Raytheon Co.	Bristol, Tenn.	AZUSA antenna
Resistoflex Corp.	Roseland, N. J.	Flex hose assembly
Rosemount Engineering Co.	Minneapolis, Minn.	2 temperature gauge
Servonic Instruments, Inc.	Costa Mesa, Calif.	2 transducer pressure
Sierra Electronics Div., Philco-Ford Corp.	Menlo Park, Calif.	Coaxial terminal
Solar Div., International Harvester Co.	San Diego, Calif.	Gas bearing heat exchanger Manifold assembly Water methanol accumulator
Space Craft, Inc.	Huntsville, Ala.	CIU 501 Command decode Frequency DC converter 2 multiplexers 410 Servoaccelerometer unit
Spaco, Inc.	Huntsville, Ala.	2 Servoaccelerometer unit
Statham Instruments, Inc.	Los Angeles, Calif.	Control accelerometer
Systron-Donner Corp.	Concord, Calif.	2 Force balance accelerometers
Tavco, Inc.	Santa Monica, Calif.	Pressure switch
Teledyne Precision, Inc.	Hawthorne, Calif.	Thermal probes
Transco Products, Inc.	Venice, Calif.	CCS coaxial switch
TRW, Inc.	Cleveland, O.	Coolant pump
United Control Corp.	Redmond, Wash.	Temperature control assembly
Vacco Industries	South El Monte, Calif.	Filter Quality test filter
Watkins-Johnson Co.	Palo Alto, Calif.	Power amplifier
Wyle Laboratories	Huntsville, Ala.	Component testing Structure testing

NORTH AMERICAN SPACE DIVISION MAJOR SUBCONTRACTORS

SUBCONTRACTOR	LOCATION	PRODUCT
Acoustica Associates	Los Angeles, Calif.	Transducers
American Brake Shoe Co.	Oxnard, Calif.	Hydraulic pumps
Amp, Inc.	Hawthorne, Calif.	Patch panels
Amphenol Space & Missile Systems Div., Amphenol-Borg Electronics Corp.	Chatsworth, Calif.	Patch panels
Astrodata, Inc.	Anaheim, Calif.	Time code generator Digital multimeter
Babcock Relay, Division of Babcock Electronics	Costa Mesa, Calif.	Relays Generator
Barry Controls	Glendale, Calif.	Shock mounts
Boonshaft and Fuchs, Division of Weston	Monterey Park, Calif.	Servoanalyzer
Computer Measurements	San Fernando, Calif.	Counters
Consolidated Electrodynamics Corp.	Pasadena, Calif.	Tape recorder
Deutsch Co., Electronic Components Division	Banning, Calif.	Connectors
Electrada Corp.	Culver City, Calif.	Test conductor console
Electronic Specialty Co.	Los Angeles, Calif.	Hybrid junction, band pass, filter, low pass filter
Electroplex, Subsidiary Borg-Warner Corp.	Santa Ana, Calif.	Logic modules Power supplies
Fairchild Precision Metal Products, Division of Fairchild Camera and Instrument	El Cajon, Calif.	Cryogenic lines
B. H. Hadley (Royal Industries)	Pomona, Calif.	Disconnects
Hallicrafters Pacific Division	Santa Ana, Calif.	Power supply
W. O. Leonard, Inc.	Pasadena, Calif.	Vent valves
Micro-Radionics, Inc.	Van Nuys, Calif.	RF couplers
Non-Linear Systems, Inc.	So. Pasadena, Calif.	Scanners
Parker Aircraft Co.	Los Angeles, Calif.	Hydraulic systems
Rantec Corp.	Calabasas, Calif.	Mistram antenna Mistram coupler Multiplexer
Rocketdyne Div., North American Aviation	McGregor, Tex.	Ullage rocket motors
Servonic Instruments, Inc.	Costa Mesa, Calif.	Transducers
Solar Division, International Harvester Corp.	San Diego, Calif.	Cryogenic lines
Stainless Steel Products	Burbank, Calif.	Cryogenic lines
Transco Products, Inc.	Venice, Calif.	Power dividers
United Electrodynamics, Inc.	Pomona, Calif.	PCM/RF assemblies

NORTH AMERICAN ROCKETDYNE MAJOR SUBCONTRACTORS

SUBCONTRACTOR	LOCATION	PRODUCT
A & M Castings Inc.	South Gate, Calif.	Aluminum castings

NORTH AMERICAN ROCKETDYNE MAJOR SUBCONTRACTORS (continued)

SUBCONTRACTOR	LOCATION	PRODUCT
Ace Industries	Santa Fe Springs, Calif.	Machined assemblies, nozzles, rotors, stators, and brg. supports
Adept Mfg. Co.	Los Angeles, Calif.	Inducers
Amphenol Corp.	Broadview, Ill.	Connectors
Anaconda Metal Hose	Los Angeles, Calif.	Flex. hoses
Anaconda Amer. Brass Co.	Detroit, Mich.	OFHC copper
Arcee Foundry	Norwalk, Calif.	Castings
Arcturus Mfg. Co.	Oxnard, Calif.	Forgings
Bendix Corp., Scintilla Div.	Sidney, N. Y.	Connectors
Beuhler Corp., Indiana Gear Wks.	Indianapolis, Ind.	Gears and shafts
Borg Warner Corp., Borg Warner Mech. Seals Div.	Vernon, Calif.	Seals
Calif. Doran Heat Treating Co.	Los Angeles, Calif.	Thermal processing of various major engine components
Cam Car Company	Rockford, Ill.	RD bolts
Chicago Rawhide Co.	Chicago, Ill.	Seals
Cleveland Graphite Bronze Div, Clevite Corp.	Cleveland, O.	Seals
Consolidated Electrodynamics Corp.	Pasadena, Calif.	Connectors and transducers
Aerospace Div., DK Mfg. Co.	Batavia, Ill.	Flex lines, bellows and gimbals
Fairchild Metal Products Div. Fairchild Camera & Instru.	El Cajon, Calif.	Bellows, ducts, gimbals, and line assemblies
General Labs, Inc.	Norwich, N. Y.	Exciters and igniters
Globe Aerospace	North Hollywood, Calif.	Machined metal parts, fittings, and elbows
Herlo Engineering Corp.	Hawthorne, Calif.	Major machined components
Hollywood Plastics, Inc.	Los Angeles, Calif.	ABS Royalite closures, covers, and other protective devices
Howmet Corp., Austenal Div.	Dover, N. J.	Castings
Huntington Alloys, International Nickel Co.	Huntington, W. Va.	Inco sheet and plate
Industrial Tectonics, Inc. General Controls, Inc.	Burbank, Calif.	Actuators, brgs.
Kentucky Metals	Louisville, Ky.	Honeycomb
L. A. Gauge Co., Inc.	Sun Valley, Calif.	Valves
Langley Corp.	San Diego, Calif.	Seals, gimbals, machined assemblies
LeFiell Mfg. Co.	Santa Fe Springs, Calif.	Tubing--thrust chamber
McWilliams Forge Co.	Rockaway, N. J.	Forgings
Orbit Machine Corp.	Gardena, Calif.	Seals
Parker Seal Company	Culver City, Calif.	O-Rings, seals, and orifice plates
Paragon Die Tool & Engr.	Pacoima, Calif.	Stators, brg. supports
Precision Sheet Metal, Inc.	Los Angeles, Calif.	Major sheet metal subassemblies, jackets

NORTH AMERICAN ROCKETDYNE MAJOR SUBCONTRACTORS (continued)

SUBCONTRACTOR	LOCATION	PRODUCT
Quadrant Engr. Company	Gardena, Calif.	Valves and components
Reisner Metals, Inc.	South Gate, Calif.	Forgings
Rohr Corp.	Chula Vista, Calif.	Major sheet metal subassemblies
Rosemount Engr. Co.	Minneapolis, Minn.	Pressure and temp. transducers
Scientific Data Systems	Pomona, Calif.	Circuit boards
Southwestern Industries	Los Angeles, Calif.	Switches
Solar, Div. of International Harvester Co.	San Diego, Calif.	Valves
Statham Instruments	Los Angeles, Calif.	Transducers, connectors, electronic assemblies
Standard Pressed Steel	Santa Ana, Calif. & Jenkinstown, Penna.	RD bolts
Texas Instruments	Dallas, Tex.	Transistors
Turbo Cast Inc.	Los Angeles, Calif.	Castings
Union Carbide Corp., Haynes Stellite Div.	Kokomo, Ind.	Castings, Hastelloy C, Rene 41
Viking Forge & Steel Co.	Albany, Calif.	Forgings
Western Arc Welding Co.	Los Angeles, Calif.	Welded assemblies
Western Way Inc.	Van Nuys, Calif.	Ducts, line assemblies, heat exchangers
Winsco Instruments & Controls	Santa Monica, Calif.	Transducer and receptacles, temp. and pressure transducers
Wyman-Gordon Company	N. Grafton, Mass.	Forgings

APPENDIX B—INDEX

ADDENDUM