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Saturn I/IB Launch Vehicle Operational Status and Experience

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THROUGHOUT THE SATURN I/IB PROGRAM considerable progress has been made in the advancement of checkout systems, launch facilities, and flight monitoring for real-time decisions, as well as the continued uprating of vehicle capabilities to meet changing mission requirements.

Each Saturn I/IB flight has provided information useful in the evolution of space rockets. This evolutionary trend is apparent when the achievements of the individual flights are reviewed.

This paper is intended to describe the vehicle operations, systems, and functions, and to review the progress and experiences which have been associated with the Saturn I/IB Program.

LAUNCH VEHICLE DESCRIPTION

The mission of the Saturn I/IB is to qualify the S-IVB stage and Instrument Unit for Saturn V applications, verify the Apollo spacecraft integrity, and train astronauts in actual manned Apollo flights. To achieve these objectives, the Saturn I first had to prove the cluster design concept of the engines and tanks, the carrier capability of two live stages, the guidance and control system, vehicle structures, and the vehicle/launch facility interface.

Development of the Saturn IB was based on a blending of existing designs for the Saturn I and the Saturn V. It uses a redesigned Saturn I booster (designated the S-IB stage), together with the S-IVB upper stage and the Instrument Unit from the Saturn V.

The concept permitted comparatively rapid development of a new vehicle. Maximum use of existing designs and facilities developed for the early Saturn configurations saved both time and costs.

Saturn IB thus became a second generation of the Saturn family -- the first United States rocket booster developed from the start as a large payload, manned space launch vehicle.

CONFIGURATION HISTORY - The Saturn I launch ve-

ABSTRACT

NASA's Saturn I/IB Launch Vehicle Program has been a remarkable success story in terms of flight accomplishments. Fourteen launch vehicles have flown without a single significant launch vehicle failure; that is, a failure that affected the mission objective.

The present Saturn IB launch vehicle comprises an S-IB stage, an S-IVB stage, and an Instrument Unit stage. The S-IB stage and the S-IVB stage are propulsion or booster stages. The Instrument Unit houses the navigation, guidance, and control equipment. A variety of payloads has been flown on the launch vehicle and different types of payloads are planned for future flights; however, the primary payloads for which the launch vehicle was designed are the Command Service Module and the Lunar Module.

This paper discusses the operational characteristics of the fourteen successful flights in terms of the launch vehicle description; launch vehicle and payload integration and interface considerations; checkout, launch, and flight operations, including equipment, facilities, procedures, time, and organization; and post-flight activities.

The paper concludes with a discussion on problem resolution, feedback, and flight accomplishments.

hicles were of two basic configurations: Block I (SA-1 through SA-4) consisted of a live booster stage (S-I) and dummy upper stages (S-IVD and S-VD); and Block II (SA-5 through SA-10) consisted of an S-I stage, an S-IV stage, and an Instrument Unit. Each Saturn IB launch vehicle consists of an S-IB stage, an S-IVB stage, and an Instrument Unit. When the launch vehicle is combined with the Apollo payload, the configuration is designated Apollo-Saturn (AS). (See Fig. 1.) The Saturn program is directed by the Marshall Space Flight Center (MSFC), which produced much of the early hardware.

Saturn I, Block I - This vehicle had only one live stage, the S-I stage. It consisted of eight tanks, each 70 in. in diameter, clustered around a central tank 105 in. in diameter. Four of the external tanks were fuel tanks for the RP-1 (kerosene) fuel; the other four, which were spaced alternately with the fuel tanks, were liquid oxygen (LOX) tanks, as was the larger center tank. The fuel tanks were interconnected, as were the liquid oxygen tanks; thus, any engine could obtain propellant from any tank.

The vehicle was powered by the S-I stage's eight H-1 engines, each producing a thrust of 165,000 lb, for a total thrust of more than 1,300,000 lb. The engines were arranged in a double pattern: the four inboard engines were fixed in a square pattern around the stage axis, canted outward slightly, while the four outboard engines were located in a larger square pattern offset 45 deg from the inner pattern. Unlike the inner engines, each outer engine could be gimbaled in a ± 7 deg square gimbal pattern.

The upper stages of the Block I rocket were dummies, simulating the three-stage configuration of the Saturn I vehicle. The ballast of the dummy stages was primarily water. The use of water, which could not survive reentry, gave an added safety factor to the tests. Typical height of a Block I vehicle was approximately 163 ft.

Saturn I, Block II - This vehicle was basically the two-stage configuration of the Saturn I vehicle; that is, it included a live second stage and a simulated Apollo payload. The first stage was an improved version of the Block I S-I stage. While the tank arrangement and the engine patterns were the same, there were marked changes between the Block I and II versions. The Block II S-I stage had eight fins added for greater aerodynamic stability in the lower atmosphere. The eight fins were arranged in two sets, four large and four stub fins mounted alternately about the base of the stage. Block II H-1 engines had a thrust of 188,000 lb each, for a combined thrust of more than 1,500,000 lb.

The Block II second stage (S-IV) used liquid hydrogen

	SATURN I BLOCK I	SATURN I BLOCK II	SATURN IB
DRY WEIGHT	117,737 LBS	152,000 LBS	159,000 LBS
LIFTOFF WEIGHT	1,090,360 LBS	1,138,000 LBS	1,280,000 LBS

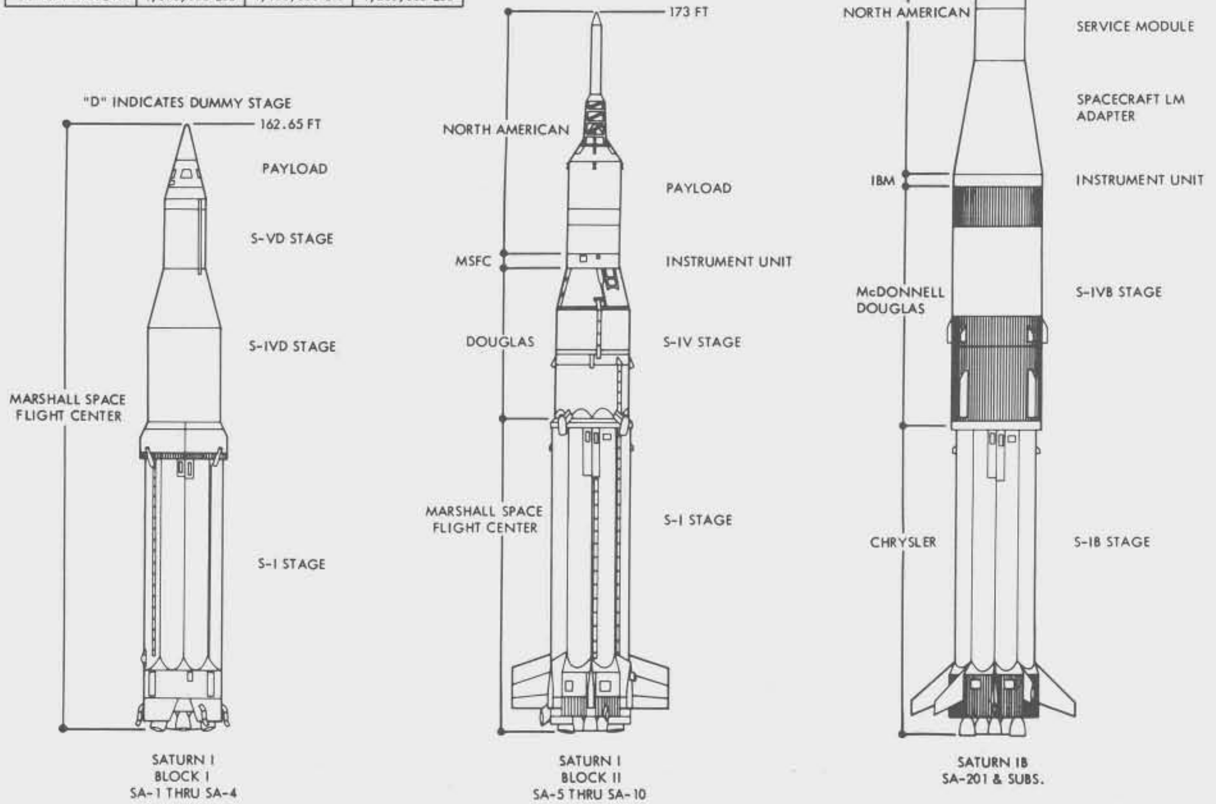


Fig. 1 - Saturn configurations

and liquid oxygen as its propellants. The liquid hydrogen and liquid oxygen tanks shared the same tank structure, separated by a common bulkhead made of two sheets of aluminum bonded to a fiberglass honeycomb core. The stage had six RL-10 engines arranged in a circle that could be gimbaled in a ±4 deg square pattern. Each of these engines produced a thrust of 15,000 lb, for a total combined thrust of 90,000 lb.

As a two-stage vehicle, the Saturn I had full orbital capability for a variety of payloads. A Jupiter nose cone, a boilerplate (dummy) Apollo spacecraft, and meteoroid detection satellites were orbited as payloads by this two-stage configuration.

The Block II vehicles also differed from the earlier vehicles by having an Instrument Unit (IU), the device that acted as the brain of the vehicle. The later Instrument Units for the last three Block II vehicles were lighter and smaller than the earlier version, consequently the later Block II vehicles were shorter and somewhat lighter than the earlier Block II vehicles.

The height of the vehicle varied with the mission; a typical height for a later Block II vehicle was approximately 173 ft.

Saturn IB - The Saturn IB Launch Vehicle, including the spacecraft and launch escape tower, stands approximately 225 ft tall and is about 21.7 ft in diameter. (See Figs. 1 and 2.) Total weight empty is about 85 tons, and lift-off weight, fully fueled, is approximately 650 tons. (See Table 1.)

The S-IB (first-stage) flight is powered by eight H-1 engines generating 200,000 lb of thrust each, for a total of 1.6 million lb. In approximately 2.5 minutes of operation, the stage consumes 41,000 gal of RP-1 fuel and 66,000 gal of liquid oxygen, to reach an altitude of approximately 42 miles at burnout. H-1 engines for later S-IB vehicles will be uprated to 205,000 lb of thrust each.

The S-IVB stage, with a single 200,000 lb thrust J-2 engine, consumes 64,000 gal of liquid hydrogen and 20,000 gal of liquid oxygen in about 7.5 minutes of operation to achieve orbital velocity and altitude. Thrust of the J-2 will be uprated by 5000 lb in later Saturn IB vehicles.

The Instrument Unit, developed for the Saturn IB and Saturn V launch vehicles, is the Saturn IB "brain" responsible for originating electronic commands for stage steering, engine ignition and cutoff, staging operations, and all primary timing signals.

Primary payload for the Saturn IB is the Apollo spacecraft, which is being developed by NASA's Manned Spacecraft Center (MSC) for manned flights to the moon. It is carried atop the Instrument Unit to complete the vehicle's launch configuration.

The S-IB stage (see Fig. 3 and Table 2), manufactured by Chrysler Corp. Space Div. (CCSD), is similar to the R&D S-I stage, but has a lightened structure, a redesigned fin system, uprated engines, a simplified propulsion system, and reduced instrumentation.

The main stage body is a cluster of nine propellant tanks.

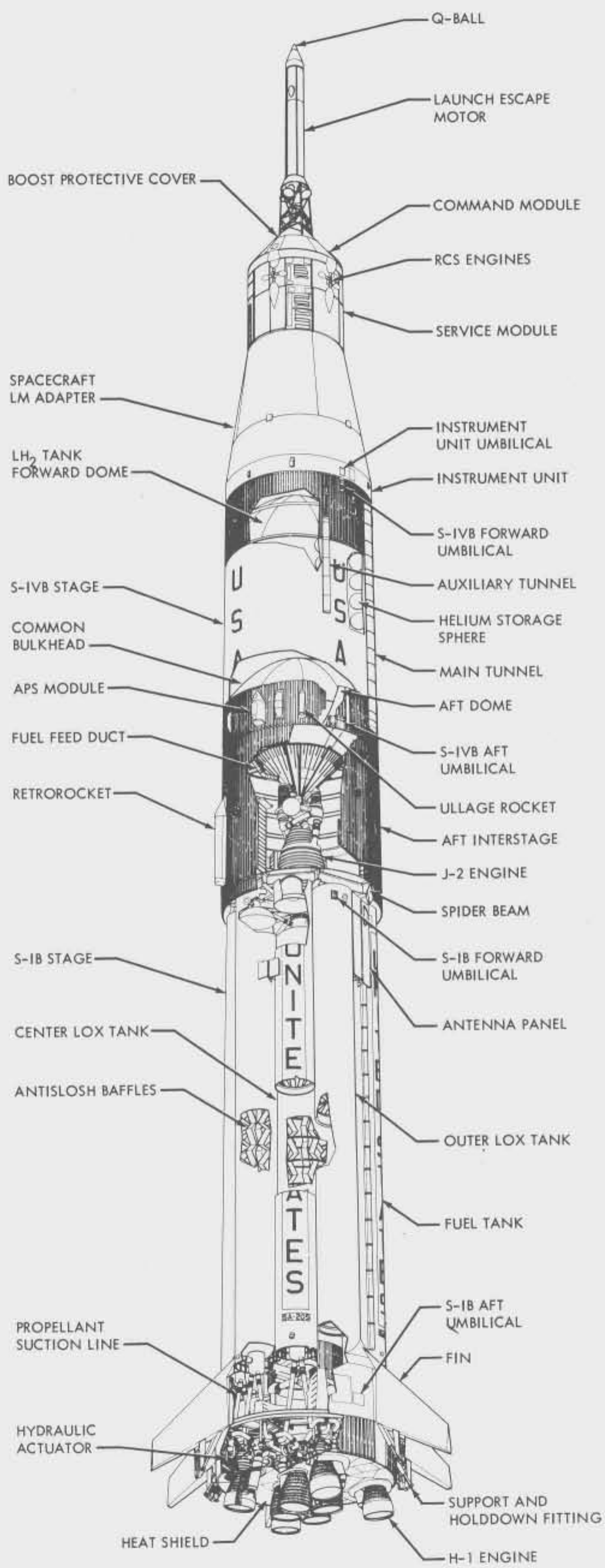


Fig. 2 - Saturn IB cutaway

The cluster consists of four fuel tanks and four LOX tanks arranged alternately around a larger center LOX tank. Each tank has antislosh baffles to minimize propellant turbulence in flight. Stage electrical and instrumentation equipment is located in the forward and aft skirts of the fuel tanks.

A tail unit assembly supports the aft tank cluster and provides a mounting surface for the engines. Eight fin assemblies support the vehicle on the launcher and improve the aerodynamic characteristics of the vehicle. A stainless steel honeycomb heat shield encloses the aft tail unit to protect it from the engine exhausts. A firewall above the engines separates the propellant tanks from the engine compartment. Eight H-1 Rocketdyne engines boost the vehicle during the first phase of powered flight. The four inboard engines are stationary and the four outboard engines gimbal for flight control. Two hydraulic actuators position each outboard engine on signal from the inertial guidance system.

A spider beam unit secures the forward tank cluster and attaches the S-IB stage to the S-IVB aft interstage. Seal plates cover the spider beam to provide an aft closure for the S-IVB stage engine compartment.

The S-IVB stage (see Fig. 4 and Table 3), manufactured by McDonnell Douglas Astronautics Co., uses a single propellant tank with common-bulkhead design and is powered by one J-2 engine. The aft interstage connects the S-IVB skirt to the S-IB spider beam unit. The aft skirt aft interstage junction is the separation plane.

A closed loop hydraulic system gimbals the J-2 engine for pitch and yaw control during flight. An auxiliary propulsion system (APS), using two APS modules on the exterior

Table 1 - Vehicle Weight Breakdown
(101 by 107 NM Orbit)

SPACECRAFT	38,100	
INSTRUMENT UNIT	4,355	
S-IVB STAGE INERT	23,948	
USEABLE RESERVE PROPELLANT (INCLUDES FPR)	1,690	
INSERTION WEIGHT		68,093
J-2 THRUST DECAY PROPELLANT AND LOX VENTING	186	
S-IVB CUTOFF WEIGHT		68,279
S-IVB PROPELLANT CONSUMED	226,785	
S-IVB APS PROPELLANT CONSUMED	6	
LAUNCH ESCAPE SYSTEM	8,650	
ULLAGE CASES	214	
S-IVB "90% THRUST" WEIGHT		303,934
S-IVB START AND BUILDUP PRPT. CONSUMED	371	
ULLAGE PROPELLANT CONSUMED	164	
S-IVB STAGE WEIGHT AT SEPARATION		304,469
S-IVB SEPARATION COMPONENT:		
ULLAGE COMPONENT AND PROPELLANT	17	
S-IVB AFT FRAME HARDWARE	31	
S-IB/S-IVB INTERSTAGE	6,456	
S-IB DRY WEIGHT	84,396	
S-IB RESIDUALS AND RESERVES	10,589	
S-IVB FROST CONSUMED	100	
S-IB FROST CONSUMED	1,000	
S-IB SEAL PURGE CONSUMED	6	
S-IB FUEL ADDITIVE CONSUMED	27	
S-IB GEARBOX LUBRICANT CONSUMED	719	
INBOARD ENGINE THRUST DECAY PRPT. CONSUMED	2,156	
OUTBOARD ENGINE THRUST DECAY PRPT. CONSUMED TO SEPARATION	1,957	
S-IB MAINSTAGE PROPELLANT CONSUMED	878,022	
VEHICLE LIFTOFF WEIGHT		1,289,945

NOTE: WEIGHTS IN POUNDS

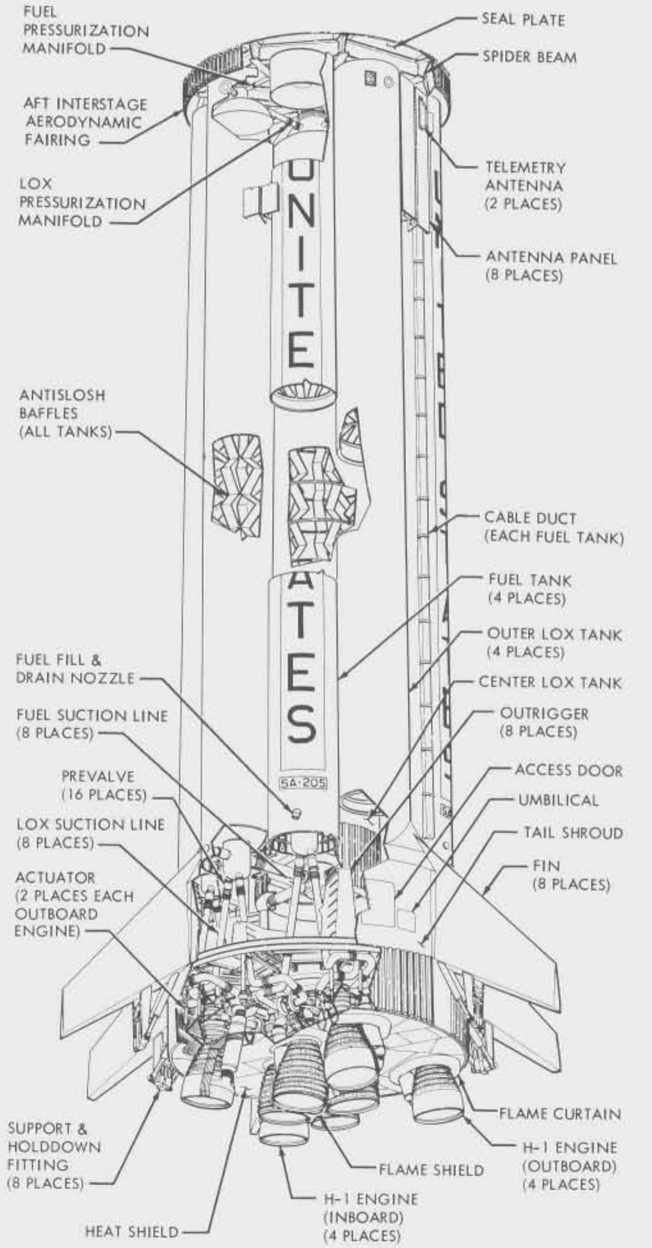


Fig. 3 - S-IB stage

aft skirt, provides vehicle roll control during powered flight and three-axis control during coast. The exact propellant mass load needed for orbital insertion with minimum residuals at cutoff is determined before launch. A propellant utilization (PU) system provides this control during flight. Instrumentation is mounted in the forward and aft skirts.

The Instrument Unit (IU) (see Fig. 5 and Table 4), manufactured by International Business Machines (IBM) Corp., is a three-segment ring structure. It is sandwiched between the S-IVB stage and the spacecraft Lunar Module (LM) adapter (SLA), with the S-IVB tank dome extending into the IU area. The unit is an unpressurized compartment with honeycomb-panel cold plates mounted around the inside periphery to accommodate the stage equipment. These panels are thermally conditioned by a stage-oriented system, which also conditions equipment in the S-IVB stage forward skirt. The IU houses equipment that guides, controls, and monitors vehicle performance from prelaunch operation to the end of active lifetime in orbit.

PRELAUNCH CHECKOUT

PHILOSOPHY - Prelaunch checkout involves all testing and launch operations required to check out completely and verify the interface and operational readiness of ground and vehicle hardware for launch, beginning with stage arrival at the NASA Kennedy Space Center (KSC) and terminating at launch countdown. Arriving units are prepared for erection at the launch complex as early as possible, with only stage-peculiar tasks taking precedence over the basic philosophy of overall testing of the space vehicle.

The primary purpose is to ensure that the flight vehicle and supporting ground systems are capable of performing mission objectives. Checkout verifies the design intent of the systems and pinpoints malfunctions.

Table 2 - S-IB Stage Data Summary

DIMENSIONS	
LENGTH	80.2 FT
DIAMETER	
AT PROPELLANT TANKS	21.4 FT
AT TAIL UNIT ASSEMBLY	22.8 FT
AT FINS	40.7 FT
FIN AREA	53.3 FT^2 EACH OF 8 FINS
MASS	
DRY STAGE	84,400 LB_m
LOADED STAGE	993,000 LB_m
AT SEPARATION	95,000 LB_m
ENGINES, DRY, LESS INSTRUMENTATION	
INBOARD, PLUS TURNBUCKLES	2,003 LB_m EACH
OUTBOARD, LESS HYDRAULICS	1980 LB_m EACH
PROPELLANT LOAD	907,400 LB_m (408,000 KG)
ENGINES	
BURN TIME	146 SEC (APPROXIMATE)
TOTAL THRUST (SEA LEVEL)	1.6 M LB_f
PROPELLANTS	LOX AND RP-1
MIXTURE RATIO	2.23:1 $\pm$ 2%
EXPANSION RATIO	8:1
CHAMBER PRESSURE	689 psia
OXIDIZER NPSH (MINIMUM)	35 FT OF LOX OR 65 psia
FUEL NPSH (MINIMUM)	35 FT OF RP-1 OR 57 psia
GAS TURBINE PROPELLANTS	LOX AND RP-1
TURBOPUMP SPEED	6717 RPM
ENGINE MOUNTING	
INBOARD	32 IN. RADIUS, 3 DEG CANT ANGLE
OUTBOARD	95 IN. RADIUS 6 DEG CANT ANGLE
HYDRAULIC SYSTEM	
ACTUATORS (OUTBOARD ONLY)	2 PER ENGINE
GIMBAL ANGLE	$\pm$ 8 DEG SQUARE PATTERN
GIMBAL RATE	15 DEG/SEC IN EACH PLANE
GIMBAL ACCELERATION	1776 DEG/SEC^2
PRESSURIZATION SYSTEM	
OXIDIZER CONTAINER	INITIAL HELIUM FROM GROUND SOURCE; S-IB BURN, GOX
FUEL CONTAINER	HELIUM
OXIDIZER PRESSURE	
PREFLIGHT	60 psia
INFLIGHT	50 psia
FUEL PRESSURE	
PREFLIGHT	17 psig
INFLIGHT	15 TO 17 psig
ULLAGE	
OXIDIZER	1.5%
FUEL	2.0%
ENVIRONMENTAL CONTROL SYSTEM	
PREFLIGHT AIR CONDITIONING	AFT COMPARTMENT & INSTRUMENT CANISTERS F1 AND F2
PREFLIGHT GN_2 PURGE	AFT COMPARTMENT & INSTRUMENT CANISTERS F1 AND F2
ASTRIONICS SYSTEMS	
GUIDANCE	PITCH, ROLL, AND YAW PROGRAM THRU THE IU DURING S-IB BURN
TELEMETRY LINKS	FM/FM, 240.2 MHz; PCM/FM, 256.2 MHz
TRACKING	ODOP
ELECTRICAL	BATTERIES, 28 Vdc (2 ZINC-SILVER OXIDE); MASTER MEASURING VOLTAGE SUPPLY, 28 Vdc TO 5 Vdc.
RANGE SAFETY SYSTEM	PARALLEL ELECTRONICS, REDUNDANT ORDNANCE CONNECTIONS.

Prelaunch checkout begins at a system level consistent with verifying and establishing a high degree of confidence in the operation of a particular component, subsystem, system, stage, or launch vehicle prior to progressing to the subsequent higher level of testing, through integrated space vehicle testing prior to launch countdown.

Results of each test are displayed on a real-time basis and will be immediately evaluated before proceeding to the subsequent test phases. Recorded test results are examined in greater detail as the testing sequence progresses. When these detailed examinations reveal unsatisfactory results, an investigation is conducted and corrective action taken; then the test is repeated to ascertain the validity of results.

Table 3 - S-IVB Stage Data Summary

DIMENSIONS	
LENGTH	59.1 FT
DIAMETER	21.7 FT
MASS	
DRY STAGE	22,500 LB_m
LOADED STAGE	225,000 LB_m
AT ORBITAL INJECTION	24,000 LB_m
S-IB/S-IVB INTERSTAGE	7,000 LB_m
PROPELLANT LOAD	230,000 LB_m(MAX)
ROCKET ENGINES	
J-2 ENGINES	
BURNTIME	440 SEC
THRUST	200,000 LB_f @ 200,000 FT
PROPELLANT	LOX AND LH_2
MIXTURE RATIO	5.5:1 (MAX), 4.5:1 (MIN)
EXPANSION RATIO	27:1
OXIDIZER NPSH	35 PSIA MIN
FUEL NPSH	27 PSIA MIN
GAS TURBINE PROPELLANT	LOX AND LH_2
FUEL TURBINE SPEED	27,000 RPM
OXIDIZER TURBINE SPEED	3,600 RPM
HYDRAULIC SYSTEM	
ACTUATORS	HYDRAULIC (TWO)
GIMBAL ANGLE	$\pm$7 DEG SQUARE PATTERN
GIMBAL RATE	8 DEG/SEC EACH PLANE
GIMBAL ACCELERATION	171.5 DEG/SEC^2
APS	
NUMBER OF ENGINES	6 (3 PER MODULE)
THRUST (PER ENGINE)	150 LB_f VACUUM
PROPELLANTS	HYPERGOLIC (MMH AND N_2O_4)
PROPELLANT LOAD	62 LB_m PER MODULE
LOCATION	AFT SKIRT AT FIN POSITIONS I AND III
RETROMOTORS	
NUMBER OF ENGINES	4
THRUST (PER ENGINE)	36,720 LB_f @ 200,000 FT
BURNTIME	1.52 SEC
PROPELLANT	SOLID, TP-E-8035 (THIOKOL)
LOCATION	90 DEG INTERVALS
ULLAGE ROCKETS	
NUMBER OF ENGINES	3
THRUST (PER ENGINE)	3,460 LB_f @ 1,000,000 FT
BURNTIME	3.9 SEC
PROPELLANT	SOLID
LOCATION	120 DEG INTERVALS AROUND S-IVB AFT SKIRT. ENGINES CANTED OUTWARD 35 DEG. JETTISONED 15 SEC AFTER STAGE SEPARATION.
PRESSURIZATION SYSTEM	
OXIDIZER CONTAINER	HELIUM
FUEL CONTAINER	INITIAL: HELIUM FROM GROUND SOURCE, GH_2 FROM J-2 ENGINE DURING S-IVB BURN
OXIDIZER PRESSURE	
PREFLIGHT	37 TO 40 psia
INFLIGHT	37 TO 40 psia
FUEL PRESSURE	
PREFLIGHT	28 TO 31 psia
INFLIGHT	28 TO 31 psia
ENVIRONMENTAL CONTROL SYSTEM	
PREFLIGHT AIR CONDITIONING	AFT COMPARTMENT AND FORWARD SKIRT
PREFLIGHT GN_2 PURGE	AFT COMPARTMENT AND FORWARD SKIRT
FLIGHT	UNIT CONDITIONING SYSTEM
ASTRIONICS SYSTEMS	
GUIDANCE	PATH ADAPTIVE GUIDANCE MODE THRU THE IU DURING S-IVB BURN
TELEMETRY LINK	258.5 MHz
ELECTRICAL	BATTERIES: 28 Vdc (3 ZINC SILVER-OXIDE) 56 Vdc (1 ZINC SILVER-OXIDE) STATIC INVERTER-CONVERTER: 28 Vdc TO 115 Vac, 400 Hz SINGLE PHASE, 2.5 Vpp, 400 Hz SQUARE WAVE AND 117, 21, AND 49 Vdc CHILLDOWN INVERTER (2): 56 Vdc TO 56 Vac, 3 PHASE, 400 Hz QUASI-SQUARE WAVE 5-VOLT EXCITATION MODULE: 28 Vdc TO 5 Vdc, -20 Vdc AND 10 Vpp, 2000 Hz SQUARE WAVE 20-VOLT EXCITATION MODULE: 28 Vdc TO 20 Vdc
RANGE SAFETY SYSTEM	PARALLEL ELECTRONICS, REDUNDANT ORDNANCE

Note

ALL MASSES ARE APPROXIMATE

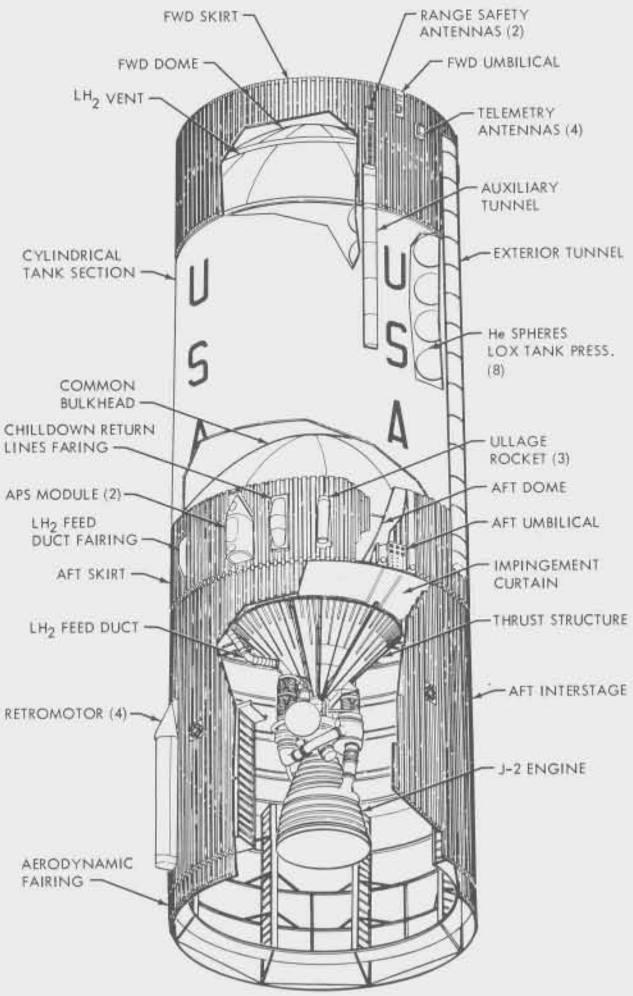


Fig. 4 - S-IVB stage

Table 4 - IU Data Summary

DIMENSIONS	
HEIGHT	3.0 FT (0.9 m)
DIAMETER	21.7 FT (6.6 m)
MASS	4,150 LB_m (1,880 kg)
PRESSURIZATION SYSTEM	ENVIRONMENT CONTROL SYSTEM GN_2 SUPPLY
ENVIRONMENTAL CONTROL SYSTEM	
PREFLIGHT COMPARTMENT CONDITIONING	IU AND S-IVB FORWARD SKIRT INTER-STAGE COMPARTMENT TEMPERATURE IS CONTROLLED BY CONDITIONED AIR FROM A GROUND SUPPLY UNTIL 200 MINUTES PRIOR LH_2 LOADING OF THE S-IVB AT WHICH TIME CONDITIONED GN_2 IS SUPPLIED
FLIGHT EQUIPMENT CONDITIONING	
PREFLIGHT	METHANOL/WATER (60/40 MIXTURE) FROM A GROUND SUPPLY THROUGH A HEAT EXCHANGER CONDITIONS ELECTRONIC EQUIPMENT
INFLIGHT	METHANOL/WATER IN A CLOSED LOOP SYSTEM CONDITIONS IU ELECTRONIC EQUIPMENT AND SUPPLIES COOLANT TO THE S-IVB EQUIPMENT LOCATED IN THE FORWARD SKIRT FOR TEMPERATURE CONTROL
ASTRIONICS SYSTEMS	
GUIDANCE	S-IB STAGE, PITCH, YAW, ROLL PROGRAMS, S-IVB STAGE PATH ADAPTIVE GUIDANCE SYSTEMS, ST-124M GUIDANCE PLATFORM AND DIGITAL COMPUTER.
TELEMETRY LINKS	FM/FM 250.7 MHz; PCM/FM 255.1 MHz
TRACKING	C-BAND RADAR; AZUSA C
ELECTRICAL	BATTERIES, 28 Vdc (3 ZINC SILVER-OXIDE). 56 V POWER SUPPLY. MASTER MEASURING VOLTAGE SUPPLY: 28 TO 5 Vdc
IU COMMAND SYSTEM	UP-DATA LINK TO THE IU (450 MHz)

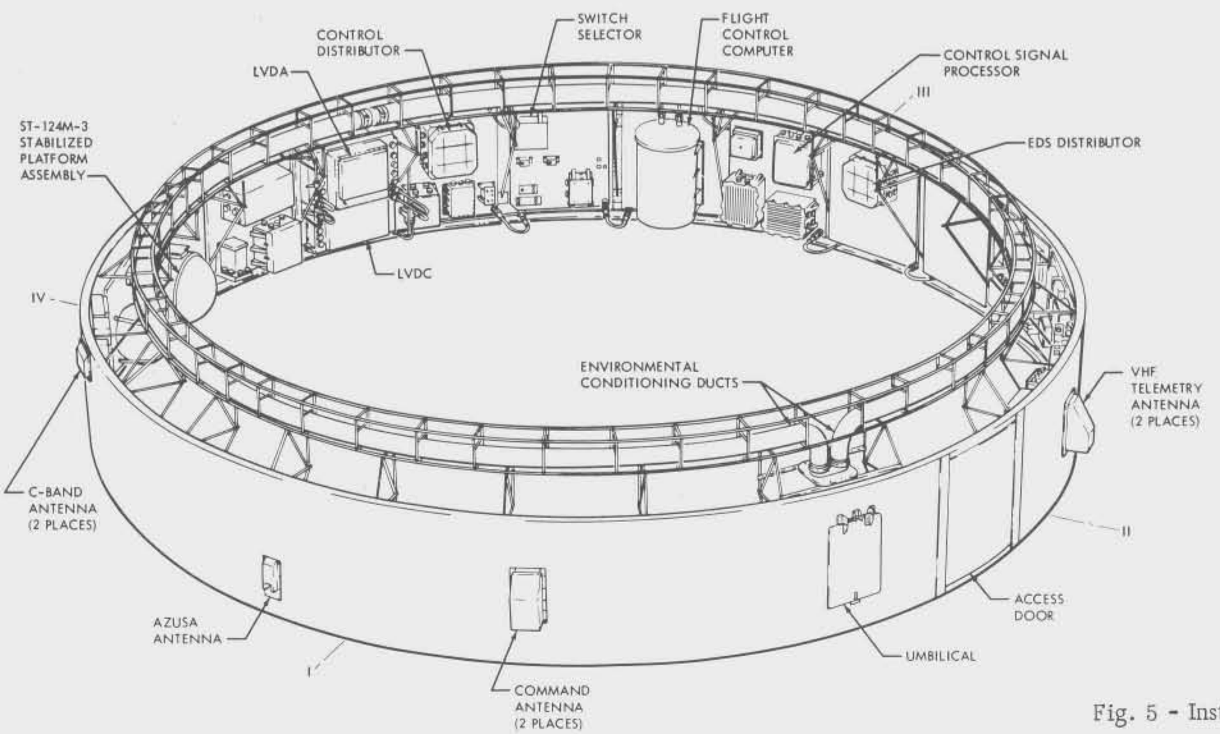


Fig. 5 - Instrument unit

To the maximum extent possible, each component, subsystem, and system of each stage of the vehicle is flight-qualified before arrival of the stage at KSC through qualification and reliability tests, in process and manufacturing inspections, and static and post-static testing and analysis.

PRELAUNCH CHECKOUT PLAN - To develop an adequate level of confidence and integrity in the spacecraft and launch vehicle with a minimum of testing, a prelaunch checkout and test sequence similar to that illustrated in Fig. 6 is utilized.

Upon arrival at KSC, each stage is given a thorough receiving inspection prior to being readied for erection at the launch complex. After receiving inspection and preparation for erection is complete, the S-IB, S-IVB, and the Instrument Unit, respectively, are transferred to the launch complex and erected. The launch vehicle is then covered to provide environmental protection. As each stage is erected, mechanical interfaces are checked and each stage is aligned with the mating stage. In conformance with the philosophy of integrated testing, only stage-peculiar testing takes place prior to integrated testing of the launch vehicle. After the launch vehicle and supporting ground and launch facility systems have been verified and prepared for space vehicle integrated testing and the spacecraft has been erected and aligned, spacecraft testing begins in preparation for overall space vehicle integrated testing. Integrated testing includes Emergency Detection System (EDS) tests, ordnance installation, countdown demonstration tests, flight readiness test, S-IB fuel loading, hypergol loading, countdown, and launch.

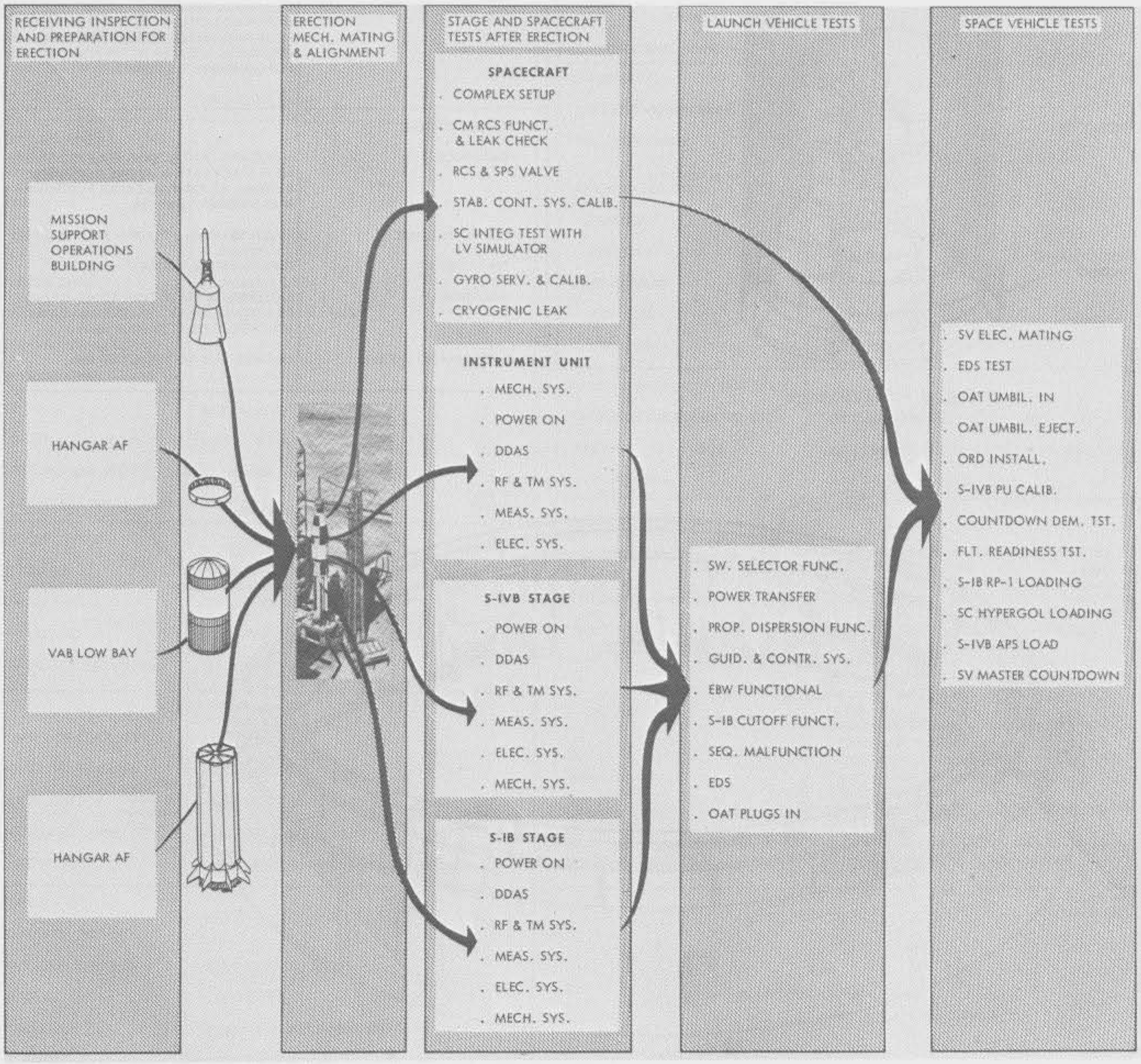


Fig. 6 - Prelaunch test and checkout

These tests ensure that the stages and ground support equipment are compatible, that the launch vehicle is flight-ready, and that static checkout, transportation, handling, age, or humidity have not deteriorated the functional and performance characteristics of the vehicle. Particular emphasis is given to verifying the continued integrity of launch- and flight-critical items.

PRELAUNCH CHECKOUT AND LAUNCH FACILITIES - Saturn IB vehicles are checked out and launched from either of two launch complexes at KSC, depending on whether the mission requires a CSM or LM spacecraft. Launch Complexes (LC) 34 and 37B, although somewhat dissimilar in configuration, are functionally identical at the launch vehicle mechanical interface. Fig. 7 shows the layout of LC-34 and identifies the major installations.

LAUNCH CONTROL CENTER (BLOCKHOUSE) - Master control of prelaunch and launch operations is accomplished from the Launch Control Center (LCC). The LCC houses and protects personnel, instrumentation, and control equipment and integrates all areas of the launch complex. The two-story igloo-type building, located approximately 1000 ft from the pad, encloses 9500 sq ft of protected floor space and 2150 sq ft of unprotected equipment space. The circular, domed, reinforced-concrete building can withstand blast pressures of 2188 psig. The first floor houses tracking, telemetry, closed-circuit television, and communications equipment. Vehicle launch-control and recording equipment and a glass-enclosed observation room are on the second floor.

AUTOMATIC GROUND CONTROL STATION - The Automatic Ground Control Station (AGCS) is a concrete and steel structure located beneath the umbilical tower and launch pad. It encloses 8170 sq ft of usable floor space for housing vehicle prelaunch test equipment. The AGCS also provides a distribution point for cables, space for vehicle power test equipment, and a distribution point for all high pressure gases. An electronic equipment room, battery room, distributor pit, and generator room are located within the AGCS. A covered cableway, which contains interconnecting control cables, links the AGCS and Launch Control Center(LCC) for integrated control of the prelaunch and launch operations.

OPERATIONS SUPPORT BUILDING - The Operations Support Building houses laboratories and fabrication work space and is used for troubleshooting, repair, and checkout of Saturn IB vehicle and launch complex components used at LC-34.

LAUNCH PAD - The 430 ft diameter, 16 ft high concrete and steel launch pad provides the load-bearing base that supports the launcher, vehicle, umbilical tower, service structure, and AGCS. The areas adjacent to the northwest and southeast ends of the launcher have refractory brick

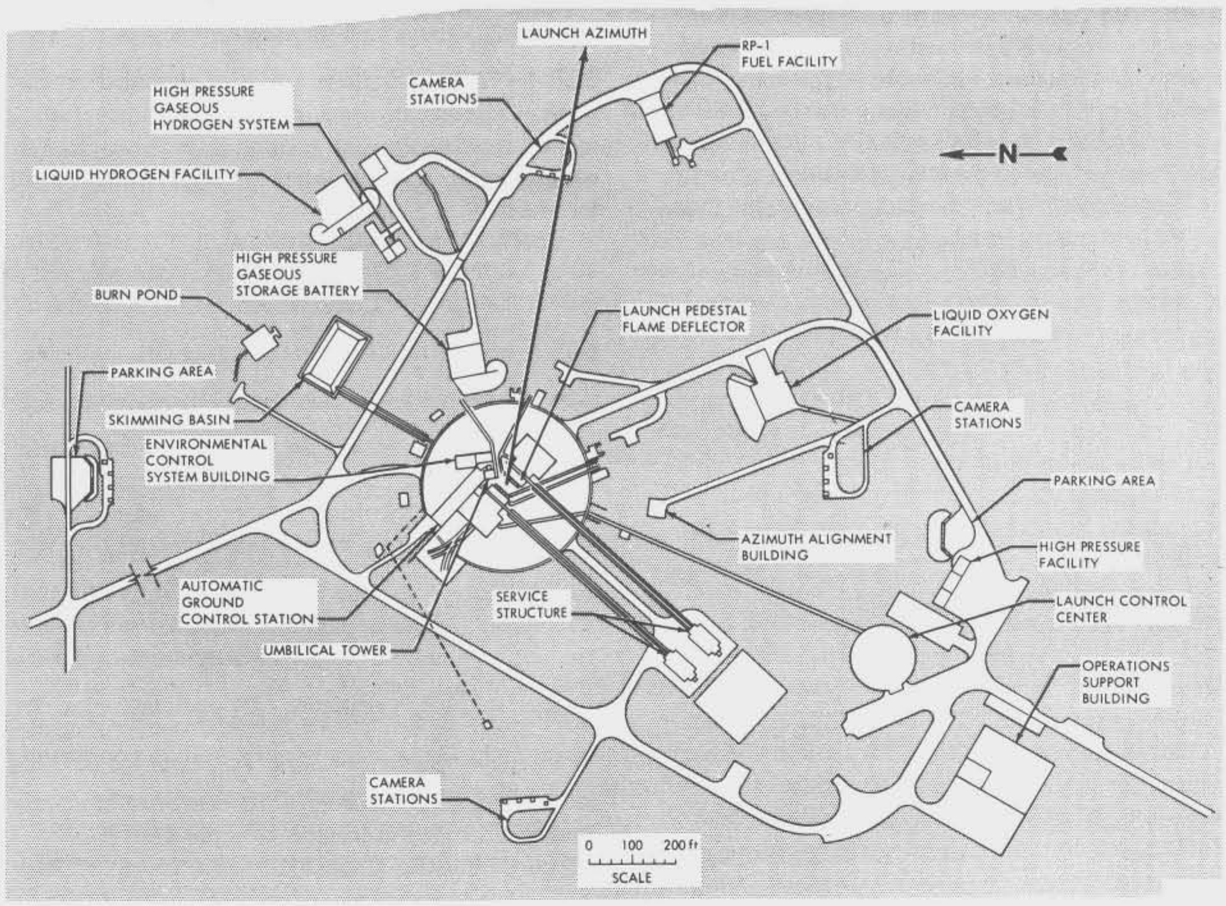


Fig. 7 - Launch complex 34

surfaces that minimize heat, flame, and blast damage caused by H-1 engine exhausts. Liquid spillage on the pad drains off through a perimeter trench, which surrounds the pad.

LAUNCH PEDESTAL - The four-legged, steel-reinforced, table-top-type launch pedestal is 42 ft square and 27 ft high. A 25 ft diameter opening in the center of the top surface allows passage of H-1 engine exhausts to the deflector below the launch pedestal and, during prelaunch operations, permits access to the vehicle boattail section. Steel plating protects all surfaces exposed to the vehicle exhaust stream. Eight holddown arms, mounted on the launch pedestal top surface, retain the vehicle until liftoff. Pneumatic, electrical, water, and environmental control systems are integral parts of the launch pedestal.

FLAME DEFLECTOR - The steel rail-mounted flame deflector positioned beneath the launcher protects the launch pedestal and the vehicle boattail from heat, blast, and flame damage during ignition, mainstage, and liftoff by diverting the 5, 000 FH-1 engine exhaust streams in two directions. Four retractable wheels transport the deflector from the parking area to its position beneath the launch pedestal. Once the deflector reaches its position beneath the launch pedestal, the wheels retract and shear plates drop into slots to secure the deflector in place. Two deflectors remain parked on rails for use under the launcher: one for regular use and the other as a backup.

SERVICE STRUCTURE - The rail-mounted service structure facilitiates vehicle erection, assembly, and checkout. Two box-truss legs measuring 70 by 37 ft support the inverted-U-shaped, rigid box-truss-frame structure, which is 310 ft high. A vertically adjustable bridge crane, with 80,000 and 120,000 lb hoists, provides a maximum lift height of 245 ft. Seven fixed and five vertically adjustable platforms provide access to the vehicle. Elevators within the structure legs carry personnel and equipment to and from the seven fixed-structure levels.

UMBILICAL TOWER SYSTEM - The umbilical tower supports four swing arms, the Apollo access arm, electrical cable, cryogenic propellant lines, environmental ducts, and pneumatic lines leading to the vehicle. The 240 ft high, steel-trussed tower mounts on the top northeast corner of the AGCS roof. The tower tapers from a 26 ft square base to a 10 ft, 10 in. square platform at its top. A 5000 lb capacity hoist mounted on the platform utilizes a trolley, which extends the hook 27 ft from the boom pivot point at maximum travel.

SWING ARM SYSTEM - The swing arm system at LC-34 consists of four swing arms, the Apollo access arm, control switch, umbilical connections, and the swing arm control panel, which connect the ground support systems on the umbilical tower to the vehicle. The swing arms are designed to disconnect the service lines from the vehicle and swing 90-135 deg horizontally to one side for clearance at liftoff. Swing arms 1, 2, 3, and 4 connect umbilically to the S-IB stage, S-IVB stage, Instrument Unit, and payload, respectively. Elevators carry personnel and equipment to and from the levels of the structure.

FIRING ACCESSORIES - The firing accessories consist of two short cable masts, which support pneumatic and electrical lines for the S-IB stage; the fuel mast and the LOX mast, which transport RP-1 and LOX to the S-IB stage; the LOX replenish line, which tops the S-IB stage containers during final countdown; two liftoff and two swing arm control switches; the boattail conditioning and water quench system; and eight holddown arms. These accessories are attached to the launcher and provide the connections between the vehicle and launch complex for propellant transfer, boattail conditioning, and pneumatic and electrical service and control prior to liftoff.

ENVIRONMENTAL CONTROL SYSTEM - The Environmental Control System (ECS) supplies air or gaseous nitrogen (GN_2) at the required humidity and temperature to the vehicle and launch pedestal during prelaunch operations. It provides all necessary air and GN_2 for inert gas purging and air conditioning of the various compartments. The ECS equipment, which initiates, controls, monitors, corrects, and terminates ECS operation, is located in six areas: the converter-compressor facility, remote fresh air intake facility, cooling tower, AGCS roof, umbilical tower, and the LCC blockhouse. Master control of the ECS comes from the LCC blockhouse and the AGCS roof. GN_2 is supplied to the system from the converter-compressor facility, and the air supply comes from the remote fresh air intake facility. Modular equipment on the AGCS roof supplies conditioned air or GN_2 through umbilical tower ducts to the instrument unit, S-IVB forward interstage, engine compartments, S-IB instrument compartments, and the launch pedestal. Environmental conditioning unit cooling water is supplied from the cooling tower, which receives hot water and fan-cools it for reuse.

CONVERTER-COMPRESSOR FACILITY AND HIGH PRESSURE BATTERY SYSTEM - All launch complex and vehicle system 6000 psig GN_2 and helium comes from the GN_2 and helium storage facility. One group of GN_2 cylinders supplies GN_2 to the pneumatic control distributor, pneumatic LOX control console, and the operations support building. The helium storage battery supplies helium to the pneumatic control distributor. A vaporizer, which receives liquid nitrogen (LN_2) from a storage tank, supplies GN_2 to the GN_2 storage battery. Helium compressors supply high pressure (6000 psig) gaseous helium to the helium storage battery. The GN_2 and helium storage facility consists of the LN_2 storage tank, LN_2 vaporizers, LN_2 pumping units, GN_2 desiccant tank, GN_2 storage battery, helium compressor unit, and helium storage battery. The converter-compressor facility is located at Launch Comple 37.

HIGH PRESSURE GASEOUS HYDROGEN SYSTEM - The GH_2 system provides storage for 6000 psig and burn-off disposal for any vented GH_2. Within the storage area are vessels, gages, hand valves, check valves, relief valves, a burst disc, orifices (used to reduce flow), a pressure transducer, regulators, a filter, and flexible hose. The storage area also receives 3500 psig GN_2, which is reduced to 750 psig for purge. The burn-off area contains a 1000 gal propane tank, a compressed air bottle, hand valves, and a flame-front generator and vent stack used to ignite vapors and superfluous gases.

PNEUMATIC DISTRIBUTION SYSTEM - The pneumatic distribution system at LC-34 supplies GN_2, ambient and chilled gaseous helium (GHe), chilled GH_2, conditioned gaseous oxygen (GOX), and water glycol at specified pressures, temperatures, and flow rates to the Saturn IB vehicle. The pneumatic distribution system equipment is located at the AGCS, the umbilical tower, and the launcher.

LIQUID HYDROGEN STORAGE AND TRANSFER SYSTEM - The LH_2 system is a remotely controlled automatic or manual system designed to receive, store, and transfer liquid hydrogen between the S-IVB stage and ground support equipment (GSE) during launch countdown. The LH_2 system consists of a storage tank, storage pressurization subassemblies, transfer lines, vent piping, burn pond, and the necessary valves and control and monitor equipment. System operation is initiated by the transfer of LH_2 from a mobile tanker to the storage tank. Prior to fueling the S-IVB stage, all components of the LH_2 system are purged with dry gaseous nitrogen from the high pressure gas facility to reduce the oxygen content. To remove the nitrogen, the system is then purged with helium from the helium storage area adjacent to the LH_2 storage area.

Fuel transfer to the S-IVB stage is accomplished automatically by control from the LCC Pneumatically operated valves control all operations. One of the two vaporizers pressurizes the storage vessel to 65 psig. The liquid hydrogen flows through a chilldown valve at a reduced rate by storage tank pressure until the transfer line is full. The S-IVB stage is then filled at the same flowrate until it is 5% full. At this point, the fast-fill valve opens and the S-IVB stage fuel tank is filled to 98% flight mass. Fast-fill is then terminated and slow-fill continues to 99%, when replenish is initiated. During replenish, fill continues to 100% and is held at that level. Fueling is controlled automatically by the propellant tanking computer system.

In addition to supplying the S-IVB stage fuel tank, the LH_2 system also supplies liquid hydrogen to a helium heat exchanger on the umbilical tower. The heat exchanger cools gaseous helium before it is loaded on board the S-IVB stage. The redundancy built into this system permits alternate fluid routing and control in the event of partial system failures which could prevent successful completion of the LH_2 vehicle loading, replenishing, and draining functions. The control panels are located in the LH_2 storage area, LCC, AGCS, and on the umbilical tower.

LIQUID OXYGEN SYSTEM - The liquid oxygen system receives LOX from trailer trucks, stores it, and transfers it to the LOX mast and umbilical tower interfaces for supplying the S-IB and S-IVB stages of the Saturn IB vehicle. The system also has the capability of draining the vehicle LOX tanks back into the storage tank. The LOX system consists of a ground storage facility, propellant transfer equipment, and lanuch area equipment. The LOX loading is normally an automatic computer-controlled operation, but can be accomplished by manual control from the LOX components panel in the LCC If conditions for automatic progression of fill are not met, the interlocks will automatically revert the system to a safe condition. The control panels are located in the LOX storage area, LCC, AGCS, and on the umbilical tower.

RP-1 SYSTEM - The RP-1 system stores RP-1 fuel, conditions the fuel for use, supplies it to the S-IB stage, and, when necessary, drains RP-1 from the stage back into the storage tanks. The RP-1 skimming pond is used to drain off spilled RP-1 and water. The fill operation entails three distinct sequences; fast fill, slow fill, and line drain. These operations can be controlled automatically by computer or manually. Fuel is transferred at a rate of 1000 gpm by a single pump until the vehicle tanks are 15% full. Then, the transfer system and vehicle are checked for leaks. Filling is then resumed at 2000 gpm, using two pumps, until the vehicle tanks are 95% full. In the slow fill sequence, fueling continues to 100% full, as set into the propellant tanking computer system The control panels are located in the RP-1 storage area, LCC, AGCS, and the launch pedestal.

RP-1 system fueling requirements have essentially remained the same from SA-1 to the current Saturn IB vehicle. Methods and components used throughout the RP-1 storage and transfer system have attained a high reliability confidence factor through previous usage.

MECHANICAL GROUND SUPPORT EQUIPMENT - Mechanical Ground Support Equipment (MGSE) is used in the launch support of the Saturn IB Launch Vehicle. Launch support activities include tests, monitoring functions, checkout, and other services provided for the vehicle on the launch pad prior to and during countdown operations. MGSE used in this manner is nonflight hardware and its support of the launch vehicle is terminated prior to or at liftoff.

ELECTRICAL SUPPORT EQUIPMENT - Electrical Support Equipment (ESE) provides displays, interfacing logic, and distribution to perform checkout of the launch vehicle systems and to conduct launch operations, using the computer and communications systems. The ESE system provides the following:

1. The capability to verify compatibility between the vehicle and the launch facility.

2. Flight readiness evaluation of the individual launch vehicle stages following their erection upon the launch pad.

3. Flight readiness evaluation of the integrated launch vehicle.

4. Perform launch vehicle safety check at any point during checkout and launch operations.

5. The capability to isolate faults at the vehicle interface without exposing personnel to hazardous conditions.

6. Power and the switching and distribution equipment necessary for the above activities, with emergency backup capacity.

The computer system (Fig. 8) consists of two RCA-110A general purpose digital computers, plus associated input/output devices and electrical support equipment. The two computers, one located in the LCC and the other in the AGCS, operate in tandem, connected by a data link. The LCC computer operates as the master station and instructs the AGCS computer. The LCC computer system processes ESE panel switch position changes, services various requests from the six display control consoles, controls data link transmissions, and monitors system parameters. The AGCS computer system services discretes from the vehicle, issues discretes to the vehicle, and services the input/output requests. In addition, the RCA-110A computers can be interfaced with the ACE-S/C CDC-160G computer to request and receive various spacecraft-oriented EDS measurements and discretes and to display them on LCC monitors. The following hardware is used in association with the two RCA-110A computers:

1. Input/Output Data Channel (IODC) is used by both computers to communicate with peripheral equipment.

2. Input/Output Sense Lines (IOS) are used by both computers to sense peripheral equipment status.

3. Input/Output Address Lines (IOA) are used by both computers to activate equipment into a send or receive condition, set sequence control relays, drive digital-to-analog converters, initiate warning alarms, and select analog or digital signal lines.

4. Input/Output Buffer Registers (IOR) are used by both computers to communicate with special peripheral equipment such as guidance computers, payloads, transmission lines, telemetry equipment, and other pulse sources.

5. Other hardware line connections are the countdown clock (CDC), Greenwich Mean Time Clock (GMTC), and interval timers.

LAUNCH OPERATIONS

Master control of launch operations for the Saturn IB vehicle is provided by the LCC at KSC.

The LCC, supported by the Central Instrumentation Facility (CIF) at KSC and by the Huntsville Operations Support Center (HOSC) at MSFC is responsible to the Mission Director at the Mission Control Center (MCC) in Houston.

The master countdown consists of a series of tests and operations, performed in a logical order to ensure the space vehicle's readiness for launch. Performance of the countdown requires approximately two days and culminates in the launch of the Apollo/Saturn vehicle. The countdown commences after the spacecraft and launch vehicle have been mechanically and electrically mated and all space vehicle tests have been satisfactorily completed.

LAUNCH CONTROL CENTER (LCC) - The LCC houses and protects launch team personnel, instrumentation, and control equipment. It also is a central communications point for all areas of the launch complex including the launch pad, electrical power supply and distribution, electrical support equipment, radio frequency and telemetry ground and environmental measurements, propellants, gas facility, and water supply and distribution.

The firing room on the second floor of the LCC houses the Ground Support Equipment (GSE) required for space vehicle control, monitoring, and checkout, and for recording prelaunch operations. The GSE can be grouped into the con-

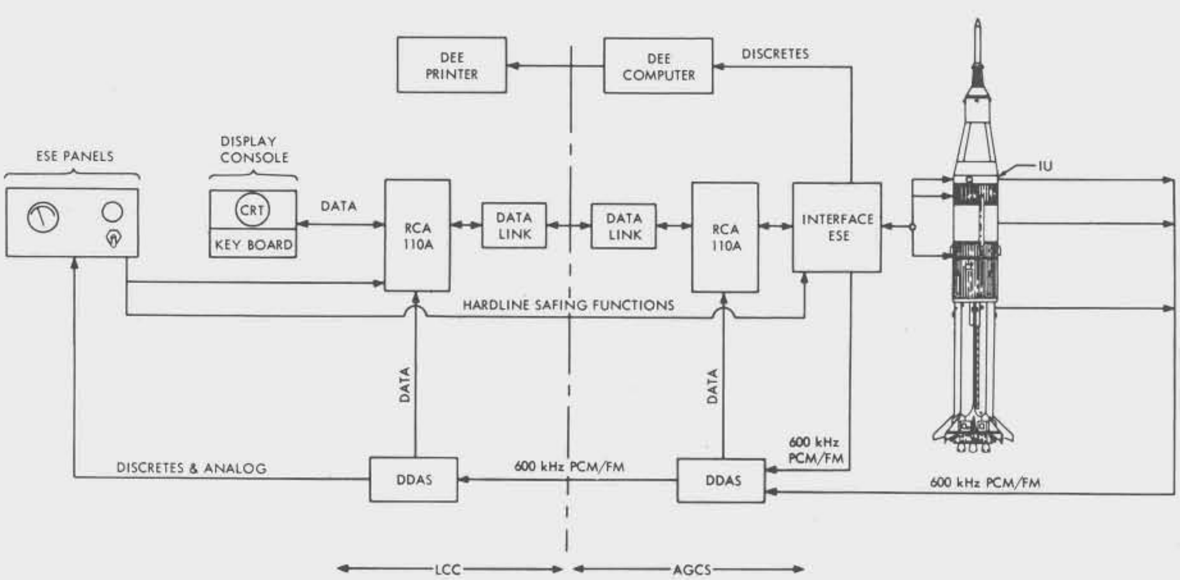


Fig. 8 - Computer system schematic

trol and monitoring stations. The equipment room on the first floor houses the necessary equipment to support the stations located in the firing room. The equipment includes telemetry, tracking, closed-circuit television, communication, instrumentation, the RCA-110A computer system, air conditioning, and power equipment.

CENTRAL INSTRUMENTATION FACILITY (CIF) - The CIF, located in the KSC industrial area, is the focal point for major prelaunch and launch telemetry, and ODOP support. The CIF houses timing equipment, countdown distribution equipment, data core, and a computer system. The computer system is used for Apollo Launch Data System (ALDS) telemetry processing, telemetry ground station, ODOP ground station, recording equipment for lightning warnings, Launch Information Exchange Facility (LIEF) equipment, and the ALDS/CASTS (Countdown and Status Transmission System). In Addition, the CIF houses the equipment for preparation of calibration data and postflight data reduction. An array of antennas located on the roof of the CIF receives telemetry and flight television signals.

Telemetry data from the Digital Data Acquisition System (DDAS), located at the LCC, are transmitted to the CIF telemetry station by hardlines during prelaunch checkout and during launch countdown. Data from the CIF antennas are provided to the LCC telemetry station. The LCC telemetry station supplies data after the umbilicals disconnect.

APOLLO LAUNCH DATA SYSTEM - The ALDS is the information link that interconnects KSC with MSC, MSFC, and other NASA centers. It comprises the following subsystems:

1. Telemetry
2. Countdown and status transmitting
3. Television
4. Trajectory
5. Command

The ALDS telemetry subsystem permits the real-time transmission of selected PCM telemetry data at the transmission rate of 40.8 kilobits per sec (kbps) from KSC to the Mission Control Center (MCC) for flight control display. The ALDS telemetry subsystem contains redundant buffering processors, parallel serial converters, and transmission links. Two identical bit streams are transmitted to the MCC over geographically separated paths. Inputs to the ALDS telemetry subsystem are obtained from the CIF. The subsystem performs parameter selection and synchronously formats the data into a serial PCM output that is transmitted to the MCC for distribution.

The CASTS transmits up to 120 selected discrete functions and 3 independent countdown time words from KSC to the MCC with an output bit rate of 2.4 kbps. These data are transmitted to the MCC independently of all other operational equipment and data circuits.

The ALDS television subsystem consists of equipment added to the Gemini launch data system television system, which transmits video from the complex areas to the video engineer's console in the Mission Support Operations (MSO) building. The video engineer's console controls selection of preprogrammed cameras, which supply inputs to wideband video lines. Transmission of the video signals to selected points within KSC and/or to the MCC is accomplished from the video console, with proper signal distribution provided at the Communication Distribution and Switching Center (CDSC).

The Apollo Launch Trajectory Data System (ALTDS) collects launch trajectory data from Eastern Test Range (ETR) and the Atlantic area ship data sources, and transmits it to the Real-Time Computer Complex (RTCC) at the MCC. The system supplies real-time, smoothed and raw, radar trajectory data to the RTCC at Houston. It also provides the liftoff event in the computer identification (ID) word from the impact prediction (IP) computers at the ETR.

CIF DISPLAYS - The recording and display equipment located in the CIF data presentation room includes the flight controllers' consoles, information systems consoles, recordings and display racks, rear screen TV projector, viewing screens, timing and countdown displays, and data evaluation tables. The three flight controllers' consoles contain Operations Intercommunication System (OIS) and TV displays. The consoles are assigned to Launch Vehicle Operations (LVO) Navigation, Networks, and Propulsion personnel. The flight controllers monitor system parameters by selecting the operational television system or telemetry displays and reference data in the form of slides. A commercial TV display is also provided for the controllers' use, when desired.

The networks flight controller has the capability to call up the telemetry displays to be presented. Telemetry data received are routed through the data core to a GE-625 computer system and the data display station. Either an alphameric or graphic presentation of a particular measurement may be selected. This display can then be presented on all flight controllers' consoles, if so desired. Discrete-event and limit-sensed lights are also located at this console. The flight controllers are also able to communicate directly with the MCC to provide them with information necessary for the launch operation.

The data display rack contains 20 meters (used to display telemetry data), a brush recorder, and an OIS communications panel. Typical TM data displayed on the meters are combustion chamber pressures, thrust chamber pressure, and first-stage engine cutoff signals. The brush recorder records both the pitch angular velocity and the attitude commands from the launch vehicle guidance system. These items are displayed in the LCC also.

Three plotting boards are used to display the vehicle trajectory information. The data for these boards are supplied by the ETR impact prediction system.

LAUNCH CONTROL CENTER PERSONNEL - During the countdown, the Launch Director is responsible to the Mission Director for assurance that the space vehicle, ground crews, launch operations organizations, and launch support equipment are ready for launch. He is responsible for all actions in the event of launch site emergencies, including an abort request. The Launch Director exercises control of activities at the launch site and is responsible for the implementation and coordination required for vehicle launch. The Flight

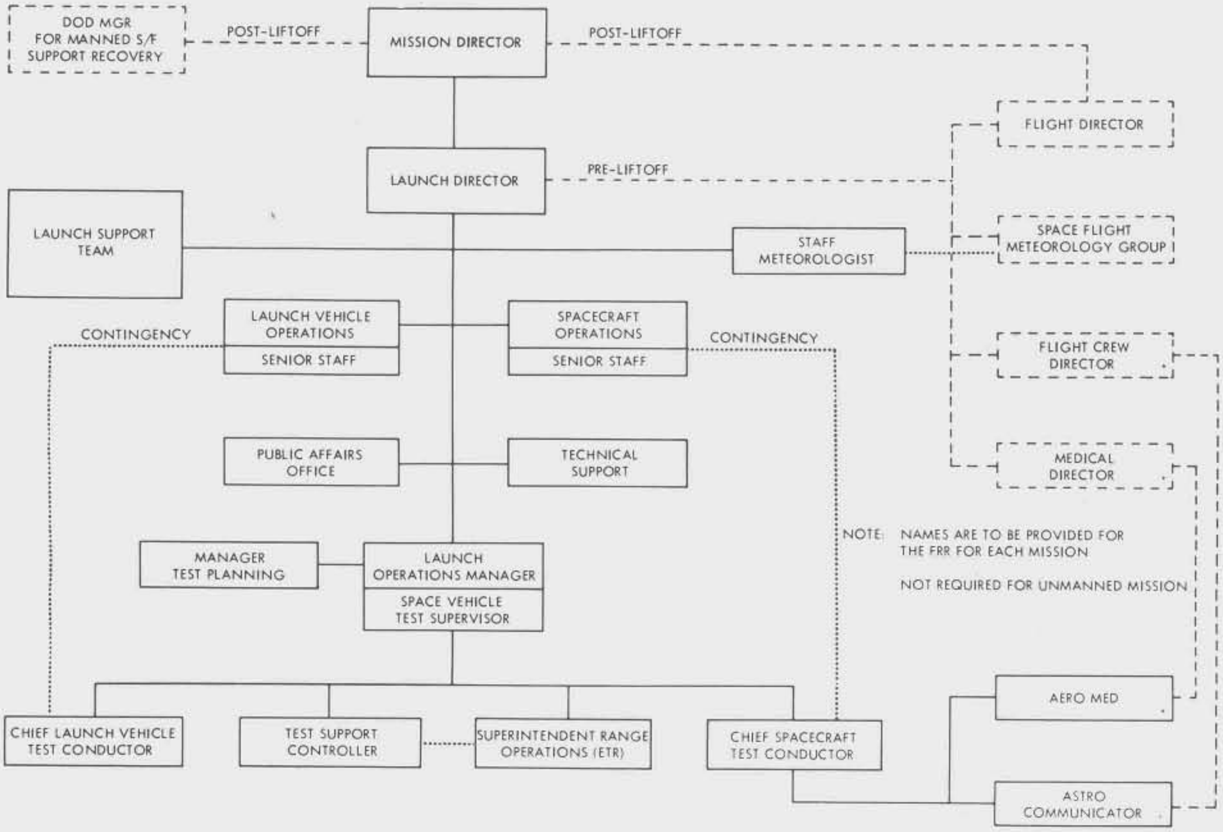


Fig. 9 - Launch operations organization

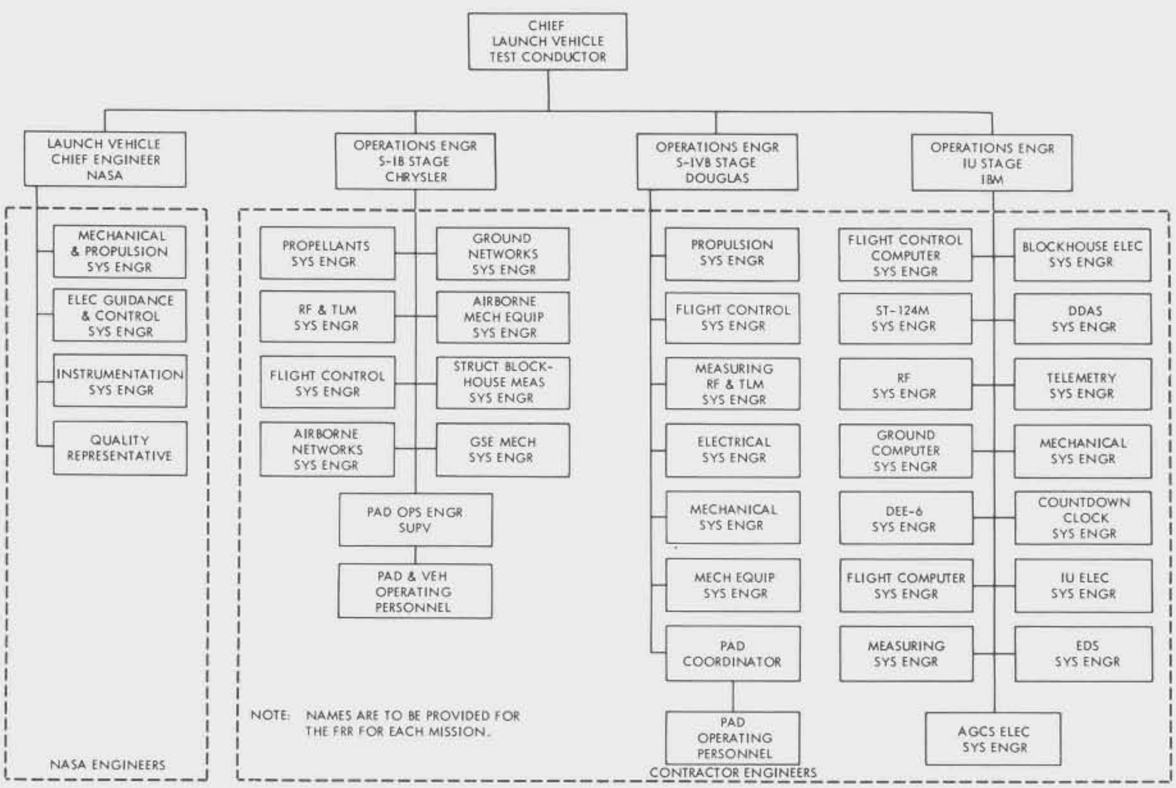


Fig. 10 - Launch team test organization

Director and the Medical Director support the Launch Director in prelaunch checkout and terminal countdown, as required. The Launch Director interfaces directly with the Flight Director who assumes responsibility when the space vehicle clears the umbilical tower.

The launch team organization is shown in Fig. 9. The Launch Operations Manager assists the Launch Director and is responsible for the management and technical execution of all work tasks, preflight tests, and launch. He acts for the Launch Director in matters pertaining to execution of space vehicle preflight tests and launch.

The Chief Launch Vehicle Test Conductor is responsible

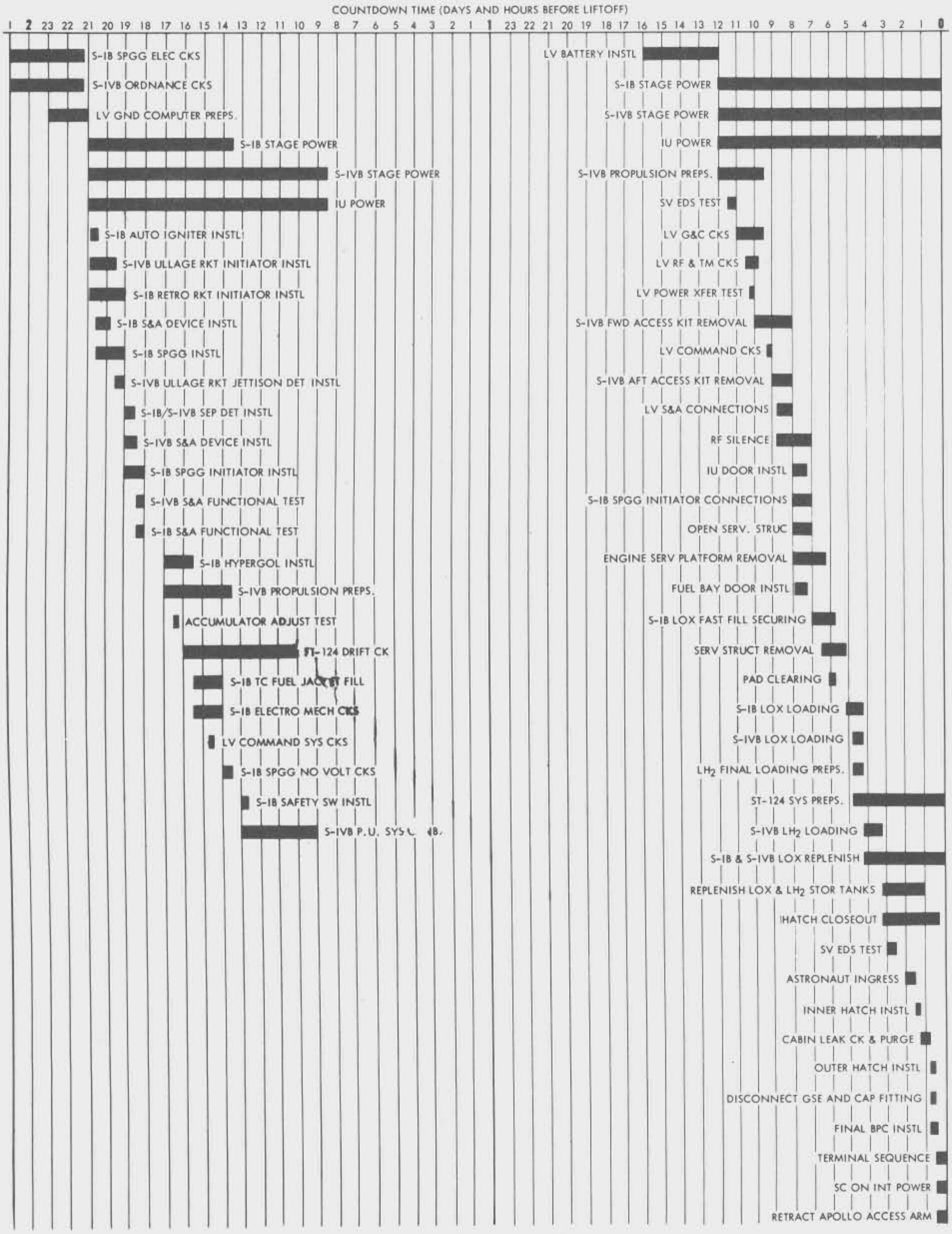


Fig. 11 - Launch vehicle countdown sequence

to the Launch Operations Manager through the Space Vehicle Test Supervisor. The Launch Vehicle Test Conductor is on a parallel level with the Test Support Controller, ETR Range Operations Superintendent, and the Chief Spacecraft Test Conductor. Fig. 10 shows the organizational breakdown under the Chief Launch Vehicle Test Conductor. The Launch Vehicle Chief Engineer, through the Stage System Engineer, monitors the following systems: mechanical and propulsion; electrical; guidance and control; and instrumentation. Subsystems peculiar to each stage are monitored by the contractor stage operations engineers.

COUNTDOWN SEQUENCE - The countdown is divided into two phases, and the two phases are accomplished on successive days. Fig. 11 identifies significant events in sequence and illustrates the approximate duration of each event.

The first phase of the countdown (first day) consists of space vehicle preflight preparations and the exercising of various systems to obtain final verification. These checks are followed by the installation and partial connection of ordnance. The second phase of the launch vehicle countdown (second day) consists of battery installation, power transfer (in which all systems are exercised), final range safety command checks, RF and telemetry checks, connection of remaining ordnance, removal of tape and covers, installation of hatches, positioning of the service structure, LOX loading of the S-IB and S-IVB blockhouse sealing, and then liquid hydrogen loading of the S-IVB. During the terminal countdown, which commences in the final minutes of phase two of the countdown (see Fig. 12), all systems are activiated and evaluated for performance prior to initiation of the launch sequence. Upon completion of all systems evaluation, automatic launch sequence is started at 2 minutes 43 sec before thrust commit (T-0). During automatic sequence, the S-IB and S-IVB stage propellant tanks are pressurized, launch vehicle power is transferred from external to internal power supplies, the S-IB stage engine purges are started, and guidance release is initiated.

The time-for-ignition signal is issued by the launch sequencer at T-3 sec; and, if the ready-for-ignition requirements have been met, the S-IB stage ignition command will be automatically issued from the AGCS program distributor to the ignition sequencer panel which is also located in the AGCS. The ignition sequencer issues four H-1 engine ignition commands 100 ms apart, thus starting the S-IB engines in pairs.

The time-for-commit signal is issued by the launch sequencer at T-0 sec. If the all-engines-running indication is present and the sequence-cutoff indication is absent, launch commit will automatically result. The commit signal turns on umbilical swing arm power and initiates LOX and fuel mast ejection, holddown arm release, and retraction of cable masts 2 and 4. Each of the four umbilical swing arms retracts upon receipt of a signal from either of two swing arm control switches, which are part of the firing accessories at the base of the S-IB stage. Vehicle vertical movement of approximately 0.75 in. causes these switches to actuate, signaling the swing arms to retract away from the vehicle. The liftoff indication is derived from the interruption of an electrical circuit across the swing arm-vehicle interface at the instant the umbilicals disconnect.

FLIGHT OPERATIONS

Mission success requires that launch and flight monitoring occur during real-time, so decisions can be made rapidly.

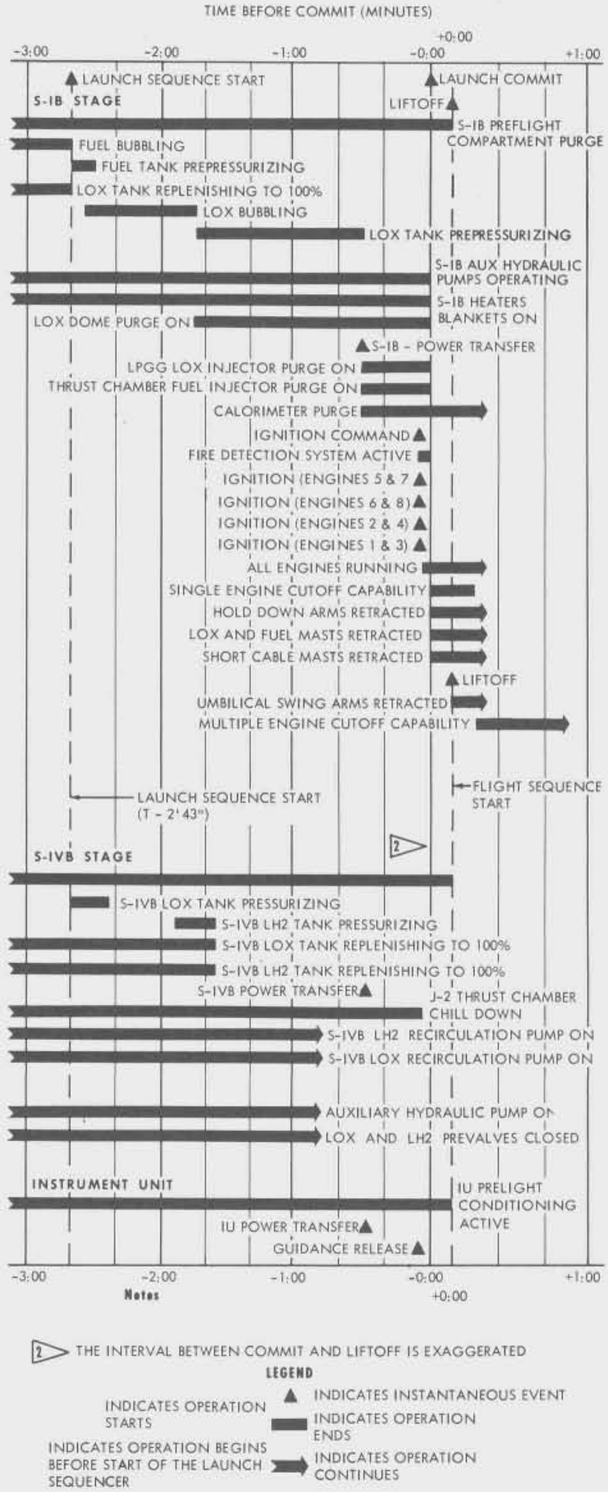


Fig. 12 - Terminal countdown sequence

Data must be analyzed quickly to identify malfunctions and deviations, and effects must be correlated with causes accurately, so launch and flight operations can be corrected. Mission rules are finalized prior to the final countdown. These rules are designed to minimize the real-time rationalizations required to cope with abnormal situations.

MISSION CONTROL MANAGEMENT - The Mission Director (MD) has overall authority during the mission (Fig. 13). During mission operations, the MD is usually located at the Mission Control Center (MCC) in Houston, but he may operate from the Launch Control Center at KSC (LCC-KSC). The Flight Director in the MCC controls flight operations from liftoff of the space vehicle through spacecraft recovery. The Launch Director at LCC-KSC exercises prime control over launch operations from the beginning of the final launch countdown until the space vehicle has cleared the umbilical tower. The Flight and Launch Directors are responsible to the MD. Centralized mission control is accomplished at MCC, and is supported by the network remote stations of the Manned Space Flight Network (MSFN), the Huntsville Operations Support Center (HOSC) at Marshall Space Flight Center (MSFC), and the LCC-KSC. Specialists in the areas

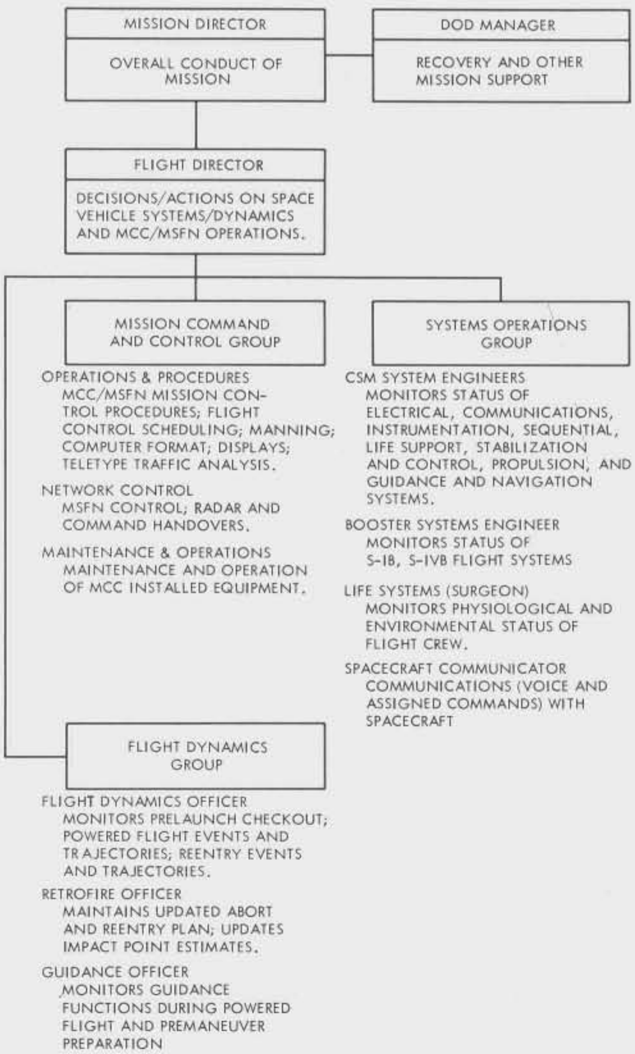


Fig. 13 - Flight operations organization

of navigation, electrical networks, instrumentation, and propulsion are located at the LCC and HOSC to provide MCC with timely information.

The MCC provides mission control and technical management in the areas of vehicle systems, vehicle dynamics, life support systems, flight crew activities, recovery support, MCC operations, and MSFN operations. The MCC:

1. Receives input data from MSFN elements over a combination of high speed data, low speed data, teletype, voice, and television channels.

2. Processes the data, which is principally composed of vehicle and spacecraft telemetry and radar tracking data.

3. Displays the processed data to provide visual representations of important aspects of the mission in progress.

4. Transmits acquisition data for the remote sites.

5. Exercises network support control in conjunction with Goddard Space Flight Center (GSFC).

Based on data prepared in advance and data processed in real-time, the MCC develops recommendations and initiates action to support the spacecraft crew and to control MSFN mission operations. Detailed mission control takes place within the Mission Operation Control Room (MOCR); detailed support in various technical disciplines is provided to the MOCR by a group of technical specialists in Staff Support Rooms, a Recovery Room, and a Weather Room. The MOCR staff is divided into three groups: the Mission Command and Control Group, the Systems Operation Group, and the Flight Dynamics Group.

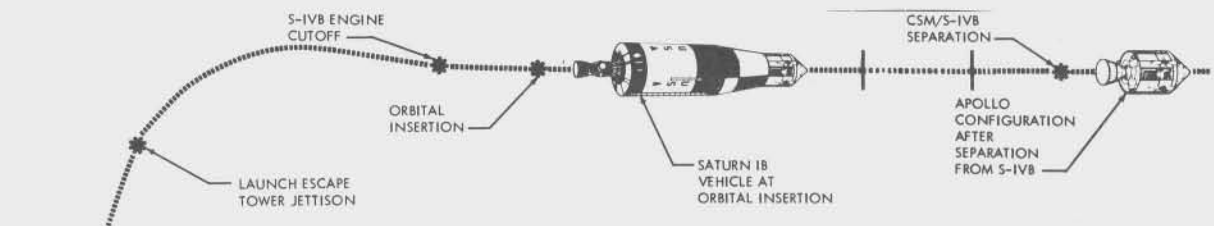
Mission Command and Control Group - This group exercises detailed real-time mission control from operational positions in the MOCR. The group monitors and analyzes mission status, implements appropriate actions to support flight plans and mission objectives, and provides detailed direction and control of all major system elements.

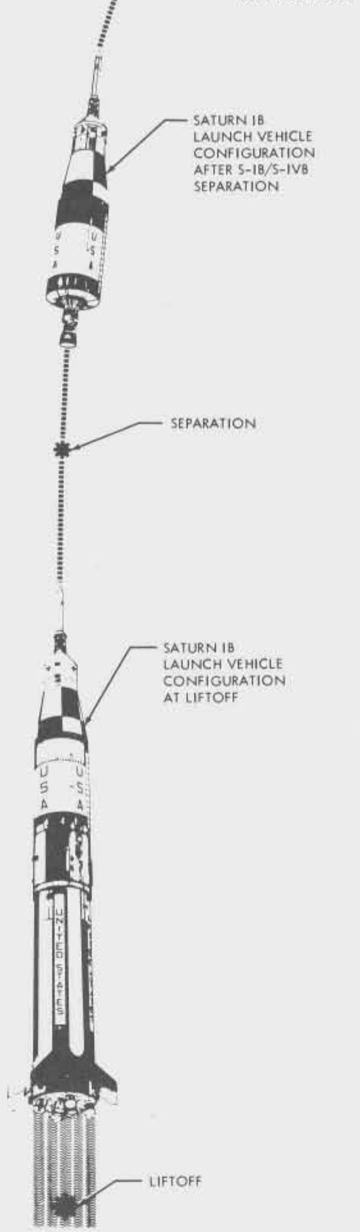
Systems Operations Group - Systems flight controllers are responsible for monitoring and evaluating the status of the flight crew and for analyzing the performance of all electrical, mechanical, and life support systems aboard the spacecraft and the launch vehicle. In addition, these flight controllers are responsible for issuing vehicle systems commands, for determining preventive and/or remedial actions if contingencies and malfunctions occur, and for conducting voice communications between the spacecraft and the ground. Key participants in the System Operations Group are the CSM Systems Engineer No. 1 (electrical, communications, instrumentation, environmental and life support systems), CSM Systems Engineer No. 2 (stabilization, control, propulsion), Booster Systems Engineer No. 1 (propulsion, pressurization, hydraulics) and No. 2 (guidance, navigation, altitude control), Life Systems Officer, and the Spacecraft Communicator.

Flight Dynamics Group - The Flight Dynamics Group is concerned primarily with the launch vehicle trajectory. Flight controllers monitor and evaluate all aspects of powered flight concerning crew safety and orbital insertion, evaluate and recommend modification of the orbital trajectory to meet mission objectives, and continuously update retrofire information for both planned and contingency reentry situations.

Key participants in the flight Dynamics Group and their responsibilities are: the Flight Dynamics Officer, who monitors powered flight events, trajectories, and reentry events for mission feasibility and updated impact points; the Retrofire Officer, who maintains an updated reentry plan; and the Guidance Officer, who performs the guidance monitor functions.

OBJECTIVES - Flight control monitoring provides real-time in-flight analysis of the mission trajectory and vehicle systems (see Fig. 14). This analysis is accomplished through use of facilities of the Air Force Eastern Test Range (AFETR), augmented by portions of the MSFN. In addition, flight control monitoring serves to control progress of the in-flight phase of the mission, using command capabilities at MCC and at remote facilities. Mission direction is given, through the Mission Director, for terminal countdown and recovery





FLIGHT SEQUENCE (101 BY 107 NM ORBIT)

NOMINAL FLIGHT TIME (HR:MIN:SEC)	(SEC)	PROGRAM TIME (SEC)	EVENT
-0:0:05.0	- 5.00	----	GUIDANCE REFERENCE RELEASE (GRR).
-0:0:03.1	- 3.10	----	INITIATE S-IB MAINSTAGE IGNITION SEQUENCE
0:0:00.0	0.00	----	FIRST MOTION.
0:0:00.2	0.20	$(0.0)_1$	LIFT-OFF SIGNAL. INITIATE TIME BASE 1.
0:0:10.2	10.20	$(10.0)_1$	INITIATE PITCH AND ROLL MANEUVERS.
0:1:15.0	75.00	----	MAXIMUM DYNAMIC PRESSURE.
0:1:35.2	95.20	$(95.0)_1$	CONTROL GAIN SWITCH POINT.
0:1:45.2	105.20	$(105.0)_1$	CONTROL GAIN SWITCH POINT.
0:2:00.2	120.20	$(120.0)_1$	CONTROL GAIN SWITCH POINT.
0:2:13.6	133.64	$(133.4)_1$	ENABLE S-IB PROPELLANT LEVEL SENSORS.
0:2:17.0	137.00	$(136.8)_1$	TILT ARREST.
0:2:17.1	137.14	$(0.0)_2$	LEVEL SENSOR ACTIVATION. INITIATE TIME BASE 2.
0:2:20.2	140.24	$(3.1)_2$	INBOARD ENGINE CUTOFF (IECO).
0:2:23.2	143.24	$(0.0)_3$	OUTBOARD ENGINE CUTOFF (OECO). INITIATE TIME BASE 3.
0:2:24.4	144.44	$(1.2)_3$	ULLAGE ROCKETS IGNITION.
0:2:24.5	144.54	$(1.3)_3$	SEPARATION SIGNAL; RETROMOTOR IGNITION.
0:2:24.6	144.61	----	S-IB/S-IVB PHYSICAL SEPARATION; CONTROL GAIN SWITCH POINT.
0:2:25.9	145.94	$(2.7)_3$	J-2 ENGINE START COMMAND.
0:2:29.0	149.00	----	TERMINATE ULLAGE THRUST.
0:2:29.3	149.34	----	90% J-2 THRUST LEVEL.
0:2:31.9	151.94	$(8.7)_3$	COMMAND PU SYSTEM ACTIVATION.
0:2:36.5	156.54	$(13.3)_3$	JETTISON ULLAGE ROCKET MOTORS.
0:3:04.2	184.24	----	JETTISON LES.
0:3:07.2	187.24	$(44.0)_3$	COMMAND ACTIVE GUIDANCE INITIATION.
0:5:49.3	349.34	$(206.1)_3$	CONTROL GAIN SWITCH POINT.
0:8:00.8	480.75	----	EMR SHIFT SENSED BY IGM.
0:9:47.2	587.19	----	GUIDANCE CUTOFF SIGNAL (GCS).
0:9:47.4	587.39	$(0.0)_4$	INITIATE TIME BASE 4. (REFLECTS AN APPROXIMATE 0.2 SECOND SYSTEMS DELAY)
0:9:57.2	597.19	----	ORBITAL INSERTION.
3:10:00.0	11400.00	----	CSM SEPARATION.

Fig. 14 - Launch vehicle configuration and flight sequence

phases where those phases are directly related to mission success. In-flight command backup of certain vehicle sequential events is initiated through remote facilities of the MCC and the MSFN.

FUNCTIONS - Specific tasks of the flight control monitoring activity include making Go/No-Go decisions; determining whether to revert to alternate flight plans, contingency plans, or aborts; and transmitting appropriate commands to carry out the decisions. In order to make the Go/No-Go decisions, the launch vehicle and spacecraft are tracked to provide empirical data. The vehicle systems and trajectory data, as described in the Flight Control Data Acquisition Requirements, are monitored and evaluated. Based upon these data, decisions are made concerning mission progress toward the satisfaction of primary mission objectives. Under special circumstances, decisions are made for reverting to alternate flight plans, contingency paths, or aborts. When an abnormal situation arises, commands are given to rectify the situation.

Commands are divided into two general types: abort commands and sequential/miscellaneous commands. There are three types of abort commands: automatic, manual, and flight dynamic. The sequential and miscellaneous commands initiate functions considered necessary to mission success. In some instances, these commands back up for onboard commands and may be sent at specified times in the mission in the event the onboard function cannot be confirmed on the ground.

PROCEDURES BY MISSION PHASE - Launch vehicle flight control procedures can be divided into the prelaunch, launch, and flight phases.

Prelaunch flight control activities and associated responsibilities begin during final countdown. Responsibilities include review of space vehicle systems appraisal and review of checkout progress, as determined by KSC. Launch vehicle checkout status information is relayed to HOSC and the MCC. This information, with spacecraft checkout information and network evaluation, provides the basis for mission Go or No-Go. Specific flight control activities performed during the prelaunch phase are as follows:

1. Monitor spacecraft checkout (LCC and MCC).
2. Monitor launch vehicle systems status (LCC, HOSC, and MCC).
3. Verify MSFN Status (MCC).
4. Monitor LCC and MCC control display status.
5. Determine recovery force status and make weather decisions.
6. Make Go/No-Go decisions.

Flight control procedures carried out during the launch and flight phase include evaluation of the launch vehicle systems, spacecraft systems, and vehicle dynamics parameters (see Figs. 15-18 for examples). Mission abort can be requested in case of adverse trajectory conditions. Abort command request becomes the responsibility of the Flight Director, the Booster Systems Engineer, and the Flight Dynamics Officer after liftoff plus approximately 10 sec. The Flight Director and Launch Director have equal abort request responsibility from liftoff until liftoff plus 10 sec. The problems of greatest concern for this 10 sec period are umbilical tower collision and pad fallback. In order to get the most complete coverage, a telescope observer is used in addition to the normal TV and blockhouse observers. The telescope observer is stationed as close as practicable to the vehicle and perpendicular to the launch vehicle umbilical tower plane. He is equipped with a high-powered tracking telescope and communications to the blockhouse.

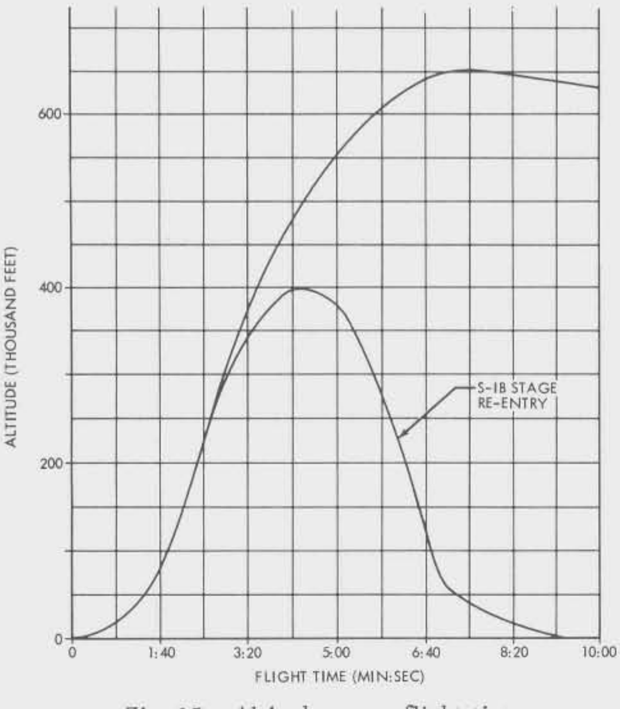


Fig. 15 - Altitude versus flight time

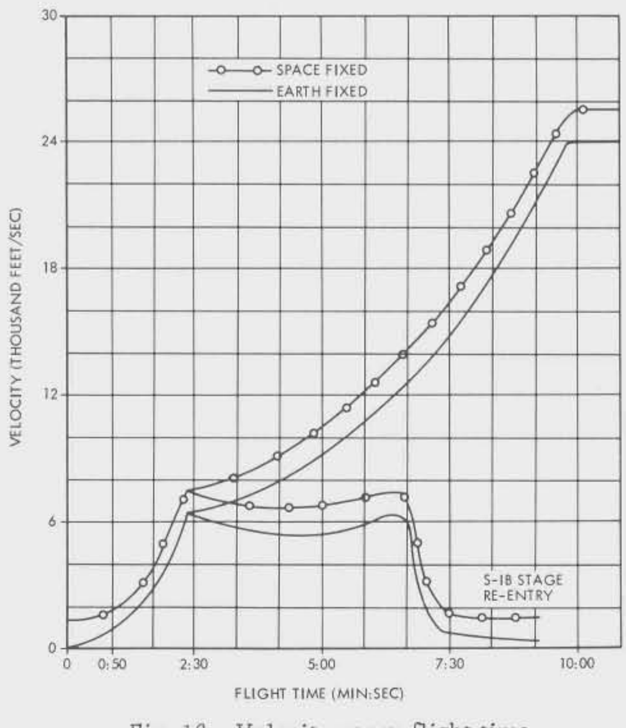


Fig. 16 - Velocity versus flight time

Booster Systems Engineers monitor and evaluate launch vehicle systems performance during the launch phase, and report their findings to the Flight Director.

The Flight Dynamics and Guidance Officers monitor the launch phase to determine adequacy of the guidance and mission trajectory. Specific flight control procedures which are accomplished between liftoff and the first spacecraft engine burn phase are:

1. Monitor and evaluate vehicle and spacecraft systems status.

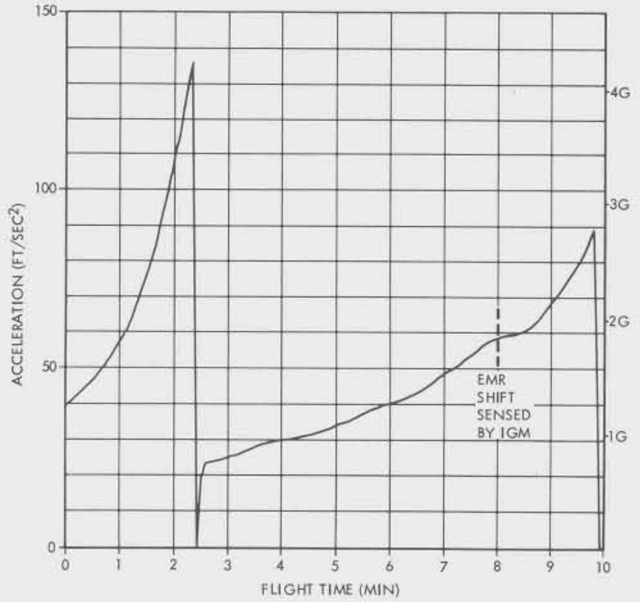


Fig. 17 - Acceleration versus flight time

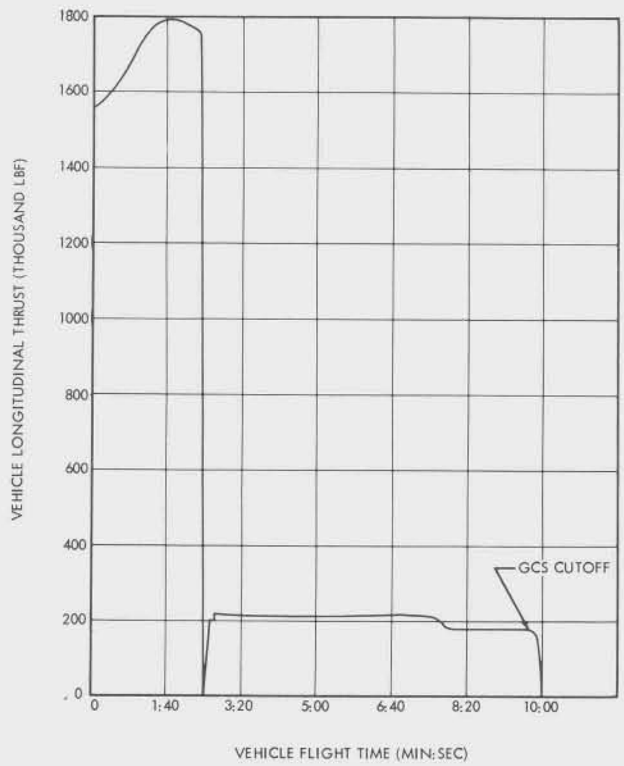


Fig. 18 - Vehicle composite longitudinal thrust (typical)

2. Monitor and evaluate trajectory for mission completion.

3. Transmit trajectory and system information to recovery ships and remote sites.

PARAMETERS MONITORED - The parameters monitored in the MCC during the flight phase may be displayed on several different scales, depending upon the need for monitoring the parameter. Data originating at the LCC and transmitted to the MCC are displayed approximately 2.5 sec after the event has occurred. These parameters are:

1. Launch and Flight Phase

a. Altitude versus downrange distance.

b. Impact points and present position (latitude versus longitude of each point)

c. Inertial flight-path-angle versus inertial velocity.

2. Launch Abort Phase

a. Spacecraft ground track.

b. Height above oblate versus longitude of present position.

c. Latitude versus longitude of present position.

POSTFLIGHT ACTIVITY

POSTFLIGHT EVALUATION - Postflight evaluation provides the only valid data source which represents the entire system in its true environment for the duration of the flight. The basic objectives of flight evaluation are to analyze the vehicle systems in terms of mission accomplishments, detection and interpretation of malfunctions, confirmation of predicted design parameters, and establishment of the vehicle environment; and to document the results both for reliability analyses and for recognizing vehicle characteristics, which become obvious only after a series of flights.

The complete flight testing of a space vehicle requires a thorough evaluation and analysis of all previously proved measurements, as well as the onboard measurements allocated to each vehicle. The completeness and effectiveness of the flight test analysis depends on coordinated evaluation procedures, which include all activities from data handling and reduction to report publication.

EVALUATION MANAGEMENT RESPONSIBILITIES - In the preparation of data handling and analyses which are associated with a flight, it is necessary to delineate areas of responsibilities, to identify data requirements, and to establish means of handling and analyzing the flight data.

The Office of Manned Space Flight (OMSF) has the responsibility for the overall planning and development of respective programs, establishing schedules, defining both mission assignments and mission objectives, as well as levying support requirements and schedules, and coordinating all ground support systems and forces required for operational support.

The Marshall Space Flight Center (MDFC) is responsible for the development of Saturn launch vehicles, engines, associated ground support equipment, and for flight operations support as it relates to the launch vehicle. The Manned Spacecraft Center (MSC) is responsible for developing manned spacecraft and conducting flight operations associated with manned missions. The Kennedy Space Center (KSC) is responsible for the overall planning, preflight preparation, and supervision of the integration, test, checkout, and launch operations. The Goddard Space Flight Center (GSFC) has the centralized responsibility for the planning, implementation, and technical operations of tracking and data acquisition facilities.

In order to identify the data requirements, each prime contractor submits to his respective lead Center the requirements for proper performance analysis of his contract end item. The Centers, in turn, compile the requirements from their contractors and add requirements of their own. This is then submitted to NASA Headquarters, where the Saturn I/IB Program Support Requirements Document (PSRD) is prepared and delivered to KSC six months prior to launch. KSC has the responsibility to satisfy these requirements in support of the upcoming launch.

FLIGHT EVALUATION WORKING GROUP - The MSFC Flight Evaluation Working Group (FEWG) directs the launch vehicle postflight evaluation. The stage contractors are responsible for the evaluation of their stages. The FEWG monitors the evaluation of all flight data to identify the cause of problems noted in each vehicle flight, and coordinates the related activities necessary to meet the objective of the flight evaluation. The completeness and effectiveness of the flight test analysis depend on coordinated evaluation procedures for all activities, from data requirements to final report publication.

The FEWG has the overall responsibility for flight evaluation efforts associated with the Saturn IB vehicles. The principal preflight responsibilities of the FEWG are defined as follows:

1. Solve the interface evaluation problems.

a. Establish coordinated evaluation procedures for all stages.

b. Establish evaluation and data reduction requirements.

c. Establish contact with evaluation groups of major subsystem contractors.

d. Discuss and aid the Mission Operations Office in the coordination of data required for flight evaluation from KSC or other data sources (that is, orbital tracking and data acquisition) for MSFC and the stage contractors.

e. Provide for compatibility of reduction and evaluation equipment.

f. Discuss requirements and impact the instrumentation modifications on the flight evaluation.

2. Assign primary evaluation missions to each group according to the respective capabilities (the term "group" is to include all organizational segments of MSFC and the contractors directly engaged in Saturn flight evaluation work).

FEWG responsibilities after each flight are as follows:

1. Direct and paticipate in the early engineering evaluation of the Saturn IB Launch Vehicle. Coordinate all steps of the overall vehicle systems evaluation through all MSFC stage contractor stages.

2. Provide for adequate participation in the evaluation meetings and inform design groups concerned with the results of the flight test.

3. Disseminate the Vehicle Flight Evaluation Report as a coordinated effort of all evaluation groups. The FEWG will stress the complete recognition of all problem areas of each test flight.

DATA REDUCTION - There are four basic data sources for the Saturn IB vehicles: tracking, telemetry, ground measurements, and tape recorder data systems, as listed below. There are a number of basic processes which translate the physical data actually received, such as cycle counts, frequencies, or phase differences, into the desired raw measurements.

1. Tracking - Tracking data reduction is the translation from the electrical or optical primary observation into a single tracking parameter. Sufficient single parameters are combined to yield a complete position determination. Vehicle velocity is obtained by numerical processes from consecutive positions. The reduction also includes the proper consideration of all known factors influencing the observation, such as geometry, refraction, and radio interference.

2. Telemetry - Telemetry data reduction includes the filtering, decommutation, digitizing, and linearization of the information received during the flight. The data output is available not only in tabulated form, but also on magnetic tapes for either automatic plotting or further processing on high-speed electronic computers.

3. Ground Measurements - Reduction of ground measurements data includes, in part, the interpretation of launch weight measurements and the transformation of general atmospheric data, ground acoustics, hardwire measurements, DDAS measurements, and camera coverage to a form applicable to the flight evaluation.

4. Tape Recorder Data Systems - These systems use an onboard recorder to store the measured information, which is later recovered at a remote location and brought to a central station for presentation and evaluation. A disadvantage of these systems is that the data are not immediately available for evaluation.

The processed telemetry data which have been linearized, converted to engineering units, separated by telemetry package, and smoothed and/or interpolated as requested are delivered in the form of SC 4020 plots and tabulated listings. Raw telemetry data are supplied in the form of analog and digital oscillograms. Selected data are plotted, tabulated, or reformatted, as required, from linearized digital data. Flight analysis programs are then used to study pressures, temperatures, guidance, and tracking information.

FLIGHT DATA ANALYSES - The evaluation activities which utilize the flight data analyze the performance of systems in the areas of propulsion, trajectory, atmospheric, mass characteristics, guidance and control, separation dynamics, structures, environmental temperatures and pressures, aerodynamics, electrical networks, instrumentation, camera coverage, and emergency detection system.

Propulsion - The propulsion system performance is evaluated for deviations from predicted sea level thrust, specific impulse and flowrate as determined for the vehicle and reduced to sea level condition. Deviations from predicted S-IB stage Inboard Engine Cutoff (IECO) and Outboard Engine Cutoff (OECO) times are analyzed together with the amount of propellant expended, and these are compared with the prior predictions. The S-IVB stage performance of the propellant utilization system and thrust fluctuations, as well as the Auxiliary Propulsion System (APS) performance during powered flight, coast, and orbit, are analyzed.

Trajectory - The actual trajectory is compared with the final preflight prediction: total velocity at OECO and S-IVB cutoff, altitude at S-IVB cutoff, range at S-IVB cutoff, and the theoretical free-flight trajectory of the booster after separation, including the calculated impact ground range and assumed impact time. A comparison of parameters of S-IVB and payload at orbital insertion (or S-IVB cutoff for ballistic trajectory missions) with nominal is made, including space-fixed velocity, perigee altitude, apogee altitude, and estimated orbital lifetime for orbital flight, as well as the conditions at S-IVB cutoff for suborbital flights.

Atmospheric - The atmospheric evaluation includes the establishment of both surface and flight meteorological parameters. Wind speed, wind direction, and wind components (obtained from rawinsonde, FPS-16 Radar, spherical balloon, and rocketsondes) are included in the atmospheric data. Stage contractors and MSC are provided a meteorological magnetic tape plus one print-out of all atmospheric data. These data are utilized by the contractors and MSC in performing their own evaluations.

Mass Characteristics - The mass characteristics evaluation includes the analysis of the vehicle weights and mass characteristics from ground ignition through spacecraft separation. Reconstruction values are compared with those of the final predicted values.

Guidance and Control - The control system performance, evaluated during the vehicle powered flight, analyzes pitch and yaw commands; pitch, yaw, and roll attitude errors and angular rates; pitch and yaw actuator positions; and the APS roll control parameters.

The guidance system is analyzed and a guidance error analysis based on the postflight trajectory is provided. The telemetered output of the launch vehicle guidance computer (LVDC/LVDA) is widely used by the various Saturn IB flight evaluation segments.

Separation Dynamics - The separation dynamics evaluation includes analyses of the separation of stages, separation times, payload/spacecraft separation scheme, and anomalies from the predicted.

Structures - The structural evaluation includes determining the bending moments, vehicle and stage loads, vibration environments, and acoustic environments.

Environmental Temperatures and Pressures - Internal and external environmental temperatures and pressures on the respective stages are evaluated. Determination of the IU prelaunch and flight internal environments is made by using telemetered data and ambient conditions from the postflight trajectory.

Aerodynamics - The aerodynamics evaluation includes the analysis of stability and drag characteristics and, when instrumentation is available, aerodynamic loading and compartment venting are evaluated.

Electrical Networks - The electrical networks evaluation is an analysis of stage electrical system component operation, using telemetered data and predicted characteristics, and includes the following major stage electrical networks: electrical control system performance, electrical power system performance, command destruct system performance, and exploding bridgewire system performance.

Instrumentation - Vehicle instrumentation is evaluated to determine the quality of telemetered data through analysis of predicted operating ranges and characteristics of instruments. Inflight calibration is used to detect system malfunctions. All measurement malfunctions are compiled, listing times (referenced to range zero) at which the measurement failed. An RF evaluation of instrumentation systems is necessary to encompass an overall instrumentation systems evaluation.

Camera Coverage - The camera coverage evaluation includes vehicle first motion, vehicle vertical motion, swing arm operation, engine and retrorocket exhaust envelopes, and a review of all onboard TV and photographic coverage.

Emergency Detection System - The EDS evaluation includes EDS event times and rate gyro outputs, abort cutoff enable times, Q-ball angle of attack sensor outputs, EDS distributor measurements, and thrust and guidance failure indications.

SATURN I/IB FLIGHT RESULTS

The 14 successful Saturn I/IB flights have proved many new design concepts and provided a high degree of design confidence, greatly increased our flight experience, and have resulted in significant improvements in flight simulation techniques. Through these improved techniques, real-time meteorological data can be incorporated into a flight simulation on the day of launch, thereby improving accuracies in the predicted flight.

The Saturn I vehicles carried considerably more in-flight measuring instrumentation than previous vehicles. This was done to shorten the developmental phase of the program. A conservative estimate of the amount of information received from the ten Saturn I flights was in excess of one billion bits. This huge volume of data consists of telemetry, tracking, meteorological, acoustic, and other information that has been compiled and used in the design and development of the Saturn IB and Saturn V vehicles. The data for the Saturn IB flights are being compiled and utilized to improve the existing programs, and to gain additional knowledge of flight conditions for future programs.

The primary mission of all Block I vehicles was to prove vehicle design by evaluation of in-flight performance of the booster engines, structural integrity of the vehicle airframe, control performance, reliability, and verification of vehicle and launch complex compatibility.

Block II vehicles SA-5 through -7 tested the S-I/S-IV stage separation, the live S-IV second stage, and the R&D version of the Saturn I Instrument Unit. In addition, SA-6 and SA-7 tested the Saturn/Apollo configuration and the ST-124 stabilized platform path adaptive guidance scheme. Vehicles SA-8, SA-9, and SA-10 tested the unpressurized prototype production model Instrument Unit, provided data for the evaluation of S-IVB/IU/SM adapter exterior thermal control coating, Apollo boilerplate CM/SM separation from the S-IV/IU/SM adapter, and the S-IV stage nonpropulsive venting system.

Excessive propellant sloshing occurred in the late stages of SA-1 powered flight; therefore, additional baffles were installed in the outer tanks of SA-3 and subsequent boosters to correct this condition. SA-2 experienced turbine erosion because of a long liquid propellant gas generator LOX lead time. Link valves were installed on SA-3 and subsequent vehicles to prevent LOX lead.

Two additional unsatisfactory conditions occurring on this flight were an open LOX emergency vent valve, indicating excessive LOX tank pressure, and intermittent tilt program disturbances. The LOX tank overpressure condition for later flights was corrected by reducing the number of coils in the LOX-to-GOX heat exchanger, which provides the gaseous oxygen for pressurizing the LOX tanks. The tilt disturbance was corrected by reducing the tilt cam loading.

On flights SA-1 through SA-4, a clockwise roll moment was observed during powered flight. This resulted in a change of actuators for SA-5 and subsequent vehicles. Vehicles SA-3 and SA-4 showed strong bending oscillations after cutoff, and to correct this, the control gains were changed.

On SA-4, inboard engine 5 was intentionally cut off 13 sec early to demonstrate the reliability of the Saturn I booster after a major subsystem failure. An engine malfunctioned on the SA-6 flight, causing number 8 inboard engine to shut down about 26 sec prematurely. This malfunction proved the ability of the complete system to adjust to a failure of some magnitude. In this instance, the guidance system was required to overcome a large deviation from the planned trajectory at S-I stage cutoff. The mission objectives were accomplished when the S-IV stage cutoff, initiated by the guidance system, occurred very near the prescribed velocity and within the expected accuracies of the system.

On SA-8 the average temperature of the GN_2 supplied to the ST-124 Inertial Platform was lower than expected; this was corrected by insulating the lines between the heater regulator assembly and the ST-124 platform.

In addition to the prime missions of qualifying the S-I booster concept under flight conditions, there were several experiments flown on the Saturn I vehicles. The most significant experiment was the test of the single-engine-out capability of the S-I stage, which was successfully demonstrated on SA-4. Another was the water cloud experiments, called Project Highwater, which was conducted on the SA-2 and SA-3 flights. In these experiments 30,000 gal of water were released near the trajectory apex to permit the measurement of ionosphere perturbances and to monitor the ionosphere's return to equilibrium.

The major mission objectives of the SA-201, the first Saturn IB flight, were:

1. To flight test the initial redesign of the S-I stage (now designated the S-IB stage)

2. To verify the S-IB/S-IVB interface compatibility.

3. To flight test the S-IVB stage with the new 200,000 lb-thrust J-2 engine.

4. To accomplish a reentry test of the spacecraft heat shield.

The only significant problem encountered on SA-201 occurred before launch in the control pressure system of the S-IB stage. Nitrogen from the 3000 psig control pressure system sphere is used to operate valves, to purge certain components, and to provide pressurization for engine turbine gearboxes. The problem in the system was that the normal ground supply pressure could not replenish the sphere as fast as the pressure was being depleted by the operating system. A test revealed that if the sphere pressure could be brought up to the required 3000 psig before launch, this volume would be sufficient for the flight even though the depletion rate would be somewhat excessive. The sphere was brought up to the required pressure by increasing ground supply pressure, and the launch and flight proceeded without incident. In addition to the achievements mentioned above, the SA-201 flight provided information on aerodynamic, dynamic, structural, and thermal characteristics of the vehicle. The flight demonstrated acceptable compatibility throughout the vehicle and payload interfaces; and although conditions of high noise levels were detected in the open loop emergency detection system (EDS) test, which is part of the overall crew safety systems, it did achieve the objective of actual flight test conditions.

The SA-203 flight was primarily concerned with determining the cryogenic behavior in a near zero "g" environment, thus verifying the orbital conditioning characteristics of the second stage propulsion system, which uses LH_2 as its fuel. Information on the reseating characteristics of the S-IVB stage propellant (LH_2 and LOX) in space was needed for future applications in which the S-IVB stage would be required to restart in earth orbit. The test revealed that a very small amount of thrust could be achieved, in a weightless environment, by venting S-IVB stage propellant. The resulting small acceleration was sufficient to keep the propellant settled in the bottom of the LOX and LH_2 tanks for engine restart.

Another major objective of the flight was to fail the common bulkhead between the LOX and LH_2 tanks to verify the results of a ground structural test. The open loop EDS was successfully flight tested after filters were installed to lower noise disturbances.

In addition to the major achievements mentioned above, the SA-203 flight provided information similar to that received from the SA-201 vehicle.

On the SA-202 flight, the major objective was to perform a suborbital flight test of the Apollo spacecraft heat shield. A smaller angle of reentry was employed for the spacecraft on this flight. This caused a "skipping" effect upon entering the atmosphere, thereby creating higher heat loads than experienced in the SA-201 test.

Again, launch vehicle systems were evaluated, and interface compatibilities were analyzed. The Apollo Saturn IB proved successful throughout. The EDS was flown closed loop, as is required before man-rating, and it too was successful.

The SA-204 flight objectives were to verify the operation and integrity of the Lunar Module (LM) subsystems and to evaluate LM staging and S-IVB/IU orbital performance. The flight data received confirmed that the system performed satisfactorily throughout.

Table 5 summarizes the entire Saturn flight accomplishments through SA-204. Missions and accomplishments of each flight are summarized individually in the following paragraphs.

SA-1 (OCTOBER 27, 1961) - Saturn SA-1 demonstrated the clustered-engine concept to be valid. The eight H-1 engines produced a combined thrust of 1,300,000 lb. Since this was a flight test of S-I stage only, the dummy upper stages were not separated after first stage cutoff.

The mission highlights included:

1. Launched with no technical holds.
2. Maximum altitude of 85 nautical miles.
3. Maximum range of 214 nautical miles.
4. Airframe structural rigidity demonstrated.
5. Control performance and reliability demonstrated.
6. Compatibility to ground support equipment demonstrated.

SA-2 (APRIL 25, 1962), PROJECT HIGHWATER - SA-2 continued the development tests of the SA-1 flight. In addition, a special experiment with widespread scientific interest, Project Highwater, authorized by the NASA Office of Space Science and Applications, was conducted. The experiment consisted of releasing nearly 30,000 gal of ballast water in the upper atmosphere. Release of this vast quantity of water in a near-space environment marked the first purely scientific large-scale experiment concerned with space environments that was ever conducted. The water was relased at an altitude of 65 nautical miles, where, within only 5 sec, it expanded into a massive ice cloud 4.6 miles in diameter. This cloud continued to climb to a height of 90 miles.

The highlights of the mission were:

1. Launched with no vehicle technical holds.
2. Maximum altitude of 65 nautical miles.
3. Maximum range of 50 nautical miles.
4. Airframe structural integrity verified.
5. Control performance and reliability verified.
6. Compatibility to ground support equipment verified.
7. Project Highwater experiment completed successfully.

SA-3 (NOVEMBER 16, 1962) - SA-3 repeated the flight objectives of the SA-2 flight, including another Project Highwater experiment. In addition, one of the first experiments necessary for the development of the Block II vehicle phase was conducted: the firing of four solid-fuel retrorockets identical to those to be used in the Block II-phase stage separation sequence, (although the upper stages were not intended to separate in the Block I vehicles). Other experiments related to the Block II vehicles were performed successfully, as was an experiment related to the Centaur program.

These mission highlights included:

1. Maximum altitude of 104 nautical miles.
2. Maximum range of 131 nautical miles.
3. Airframe structural integrity verified.
4. Control performance and reliability verified.
5. Compatibility to ground support equipment verified.
6. Test of Block II swing arms.
7. Test of prototype horizon sensor.
8. Flight test of passenger stabilized platform.
9. Test of pulse code modulation (PCM) and UHF telemetry.
10. Test of Block II heat shield panel.
11. Test firing of retrorockets (no separation).
12. Simulation of Block II ullage volumes.
13. Achievement of significant increase in propellant utilization efficiency through the use of a new cutoff method.
14. Centaur rocket pressure distribution study.
15. Project Highwater experiment successfully completed.

SA-4 (MARCH 28, 1963) - SA-4 was the final vehicle of the Block I phase. In addition to carrying on the flight objectives of the first three Saturns, it carried a considerable amount of equipment to be used in the Block II phase. The external characteristics of the SA-4 vehicle reflected the Block II design more closely than any previous vehicle. In addition to the retrorockets added to the design on SA-3, dummy camera pods were attached to the S-I stage to simulate the camera pods that would be used on some of the Block II vehicles. Also, and far more obvious, dummy aerodynamic protuberances were added to the inert S-IV stage to simulate the ullage rockets that would be used on the live rocket stage.

A special feature of the clustered engine concept is that if an engine fails past a critical point on a flight path, it may be possible to carry out the mission by utilizing the remaining operating engines. The Saturn I was specifically designed to have this engine-out capability.

Because the Saturn engine-out capability had not been demonstrated, it was decided to program a premature cutoff of one engine as an experiment on the SA-4 flight to demonstrate the validity of the design calculations. One of the engines was deliberately cut off after 100 sec of flight time, and the remaining portion of the flight was made using only seven engines. The flight performed as calculated, and the engine-out principle was proven to be a valid engineering approach.

The highlights of this mission included:

1. Maximum altitude of 81 nautical miles.
2. Maximum range of 219 nautical miles.

Table 5 - Summary of Saturn Flight Program

SUBORBITAL FLIGHTS						
	Date	Duration	Altitude	Distance	Burn Distance	Remarks
SATURN I						
SA-1	10-27-61	408 sec.	85 mi.	207 mi.	116 sec.	Successful ballistic flight
SA-2	4-26-62	162 sec.	65 mi.	50 mi.	117 sec.	Project High Water I. 96 tons of water exploded.
SA-3	11-16-62	292 sec.	104 mi.	131 mi.	149 sec.	Project High Water II. 95 tons of water exploded.
SA-4	3-28-63	398 sec.	81 mi.	219 mi.	121 sec.	One inboard engine shut down intentionally after 100 sec. and flight continued successfully.
SATURN IB						
AS-201	2-26-66	1,917 sec.	306 mi.	5,400 mi.	602.9 sec.	Successful suborbital lob shot to position Spacecraft for earth reentry heat shield test.
AS-202	8-25-66	5,582.2 sec.	617 mi.	17,800 mi.	588.5 sec.	Successful suborbital flight to test spacecraft's heat shield, and check launch vehicle.

ORBITAL FLIGHTS							
	Date	Perigee	Apogee	Orbital Period	Burn Duration 1st Stg.	Burn Duration 2nd Stg.	Remarks
SATURN I							
SA-5	1-29-64	163 mi.	479 mi.	95 min.	146 sec.	481 sec.	First flight with live second stage. 37,900 lbs. into orbit.
SA-6	5-28-64	114 mi.	149 mi.	88 min.	149 sec.	473 sec.	Boilerplate Apollo Spacecraft. One inboard engine unexpectedly shut down 26 sec. early but did not impair flight.
SA-7	9-18-64	112 mi.	145 mi.	88 min.	147 sec.	471 sec.	Boilerplate Apollo Spacecraft. 39,000 lbs. into orbit. Declared operational three flights early.
SA-9	2-16-65	309 mi.	463 mi.	97 min.	145 sec.	473 sec.	First operational flight. Pegasus I placed into orbit.
SA-8	5-25-65	315 mi.	465 mi.	97 min.	148 sec.	473 sec.	Pegasus II. First night launch
SA-10	7-30-65	328 mi.	330 mi.	95 min.	148 sec.	479 sec.	Pegasus III.
SATURN IB							
AS-203	7-5-66	*115 mi.	*117 mi.	*88 min.	142 sec.	288 sec.	Test liquid hydrogen behavior. Simulation of Saturn V restart conditions.
* Orbital parameters given are for initial orbit only; propulsive experiments caused small variations.							
AS-204	1-22-68	98 mi.	138 mi.	*88 min.	142 sec.	451 sec.	Successful Flight Test of Apollo Lunar Module

3. Airframe structural integrity was verified for use of some Block II aerodynamic protrusions.

4. Control performance and reliability verified.

5. Test of prototype horizon sensor.

6. Thermal flight test of selected sensing devices.

7. Test of Block II antenna panels.

8. Use of onboard playback recorder for data gathering.

9. Passenger test of Q-ball angle-of-attack device.

10. Passenger test of radar altimeter.

11. Passenger test of mistram device.

12. Successful engine-out test.

13. Verification of propellant utilization improvement by cutoff method.

14. Test of PCM and UHF telemetry.

15. Test of Block II heat shield panels continued.

16. Retrorocket firing (no separation).

17. Flight test of passenger stabilized platform.

SA-5 (JANUARY 29, 1964) - SA-5, the first of the Block II vehicles, was a milestone in the orderly evolution of the Saturn I design. Although it still retained a Jupiter nose cone and had a payload compartment only slightly different from the dummy S-V stages of the Block I vehicles, it had a live second stage that gave the Saturn I orbital capabilities for the first time.

The S-IV stage itself was a major developmental milestone, being powered by a propellant combination of LH_2 and LOX.

In performance, the clustered engine concept further proved its validity. Cameras recovered from the SA-5 flight recorded the S-IV stage separation and the ignition of the second stage engines.

There were many firsts on the SA-5 flight. A new Instrument Unit, prototype of those to be flown on later vehicles, was used. The IU, located immediately above the S-IV stage, acted as the overall brain of the vehicle. The path of the powered portion of the flight was directed from the IU, and many sensing and evaluating instruments were controlled through this central point.

Beginning with SA-5, the S-I stage used the uprated H-1 engines, each of which had a thrust of 188,000 lb. Thus, SA-5 was the first vehicle launched with a first-stage thrust of 1,500,000 lb.

The flight was an unqualified success, injecting the largest payload then recorded (38,000 lb) into an Earth orbit.

Mission highlights included:

1. First launch of Block II vehicle.

2. First flight test of S-IV stage.

3. First flight test of Instrument Unit.

4. Use of S-I stage fins for control stability demonstrated.

5. Successful passenger flight of ST-124 guidance system.

6. Successful stage separation.

7. First orbital Saturn vehicle.

8. First Saturn vehicle using uprated H-1 engines.

9. Successful recovery of motion picture camera pods from S-I stage.

10. Injection of 38,000 lb satellite into orbit.

SA-6 (MAY 28, 1964) - SA-6 was the second of the developmental Block II vehicles, and the first to loft a dummy Apollo capsule. The vehicle continued the developmental experimentation of the previous missions, and was another step in the development of a flight qualified vehicle for its primary mission of support to the Apollo manned flight program.

The mission objectives were extensions of those of the SA-5 mission. In addition to the basic objectives of SA-5, the sixth Saturn vehicle demonstrated capabilities, such as the in-flight environmental parameters of the spacecraft and the normal jettisoning of the Launch Escape System tower, that would assist in designing the manned Apollo spacecraft.

Motion picture cameras on SA-6 recorded the propellant consumption of the S-I stage and the separation of the S-I and S-IV stages. These cameras were attached to the vehicle in special recoverable camera pods. After the separation of the stages, the camera pods were ejected from the S-I stage and, with it, continued in a ballistic trajectory. Unlike the stage, however, the camera pods were equipped with special recovery parachute devices that enabled the camera pods to land relatively intact in the ocean down range, where they were recovered by ship.

SA-6 demonstrated the validity of the clustered engine concept and the value of the engineering planning of an engine-out capability. One of the engines aboard the vehicle cut off prematurely. Although this was not part of the programmed flight, the performance of the remaining seven engines ensured the successful completion of the mission.

The spent S-IV stage, the Instrument Unit, and the Apollo spacecraft assembly, having a total weight in excess of 18 tons, were successfully injected into orbit.

The high points of this mission included:

1. Demonstration of physical compatibility of the launch vehicle and the first Apollo boilerplate.

2. Partial active test of the ST-124 system.

3. First test of guidance velocity cutoff.

4. Successful mission with planned large angle of attack.

5. Recovery of motion picture camera pods from S-I stage.

6. Demonstration of Launch Escape System under flight conditions.

7. Engine-out capability demonstrated by actual engine casualty.

8. Injection of 37,000 lb satellite into orbit.

SA-7 (SEPTEMBER 18, 1964) - The SA-7 was the first of the Block II vehicles to be considered operational. The unparalleled success of the Saturn I program permitted the research and development phase of the Saturn I program to be telescoped.

An important mission of SA-7 was to demonstrate a method of jettisoning the Launch Escape System (LES) tower different from the method used aboard SA-6. It is necessary to jettison the LES once the flight vehicle reaches an altitude where the tower is no longer required, and the successful demonstration of an alternate method of accomplishing this critical function separation would provide a design flexibility that could eventually prove very useful.

Many of the SA-7 mission objectives were extensions of SA-6 and earlier flight missions. The movie camera setup was the same, although the results were a little different than had been planned due to the intervention of a hurricane. When the camera pods had been ejected, they reentered flawlessly, but landed in a region of the sea close to Hurricane Gladys. The storm closed in before the capsules could be recovered; and, though given up for lost, two of the three capsules washed ashore after seven weeks, none the worse for wear except for the acquisition of a few barnacles.

The overall mission of SA-7, however, was the same as previous missions; that is, the injection into orbit of a dummy Apollo spacecraft, an S-IV stage, and the Instrument Unit--a total weight of 19-1/2 tons.

The highlights of this mission included:

1. First complete flight test of ST-124 system using closed loop.

2. First flight demonstration of the spacecraft alternate LES tower jettison mode.

3. First test of S-IV stage nonpropulsive venting system.

4. Passenger test of S-I area fire detection system.

5. First flight not to use S-IV stage LOX backup pressurization system.

6. Third-flight test of Instrument Unit.

7. First flight of active time-tilt polynomial system for the S-I stage.

8. Third orbital flight; second orbital flight of expended S-IV stage, Instrument Unit, and Apollo boilerplate.

9. The approximate orbital weight was 39,100 lb.

SA-9 (FEBRUARY 16, 1965) - The SA-9 vehicle, which was launched before SA-8, was the last Saturn I vehicle with an MSFC-manufactured first stage. Boosters for the last two Saturn I's (SA-8 and SA-10) were produced by Chrysler Corp. Space Div.

SA-9 was the first vehicle of the final Block II configuration. One of the major differences between the first and final Block II vehicle configurations was in the design of the Instrument Unit. The earlier Instrument Unit was divided into "conditioned" tubular sections. Each section was pressurized and the components were enclosed within the sections, surrounded by an inert gas for environmental control. The later Instrument Unit was not pressurized; the components, which were mounted to the wall of the IU, had no in-flight external temperature control. However, some of the components incorporated individual prelaunch heating elements in their design.

SA-9 launched the Meteoroid Technology Satellite (MTS), sponsored by the NASA Office of Advanced Research and Technology. This satellite was nicknamed Pegasus after the winged horse of Greek mythology, because it had two panels or "wings." These huge panels detected and recorded meteoroid impacts and transmitted the information back to Earth. In addition to the general scientific value of such data, the information relating to meteoroid density is of extreme importance for our manned space program, since it gives us an idea of the penetration hazards of meteoroids to be encountered in an Earth orbit environment.

Mission highlights included:

1. First Saturn rocket with an operational payload.

2. Pegasus satellite demonstrated functional operations.

3. Evaluation of closed-loop guidance accuracy.

4. First flight utilization of iterative guidance scheme.

5. First flight of an improved, nonpressurized Instrument Unit.

6. Evaluation of a thermal control coating for S-IV stage, Instrument Unit, and adapter of Service Module.

7. Demonstration of separation of the boilerplate Service and Command Modules from the S-IV stage and Instrument Unit.

8. First live, high-resolution, fast-scan, television broadcast originating on an orbiting satellite.

9. Orbiting of 33,900 lb payload consisting of spent S-IV stage, Instrument Unit, Pegasus satellite, and the Apollo Command Module.

SA-8 (MAY 25, 1965) - The SA-8 flight was a milestone in the Saturn program. This was the first Saturn launch vehicle to use two powered stages manufactured by private industry -- the S-I stage by Chrysler Corp. Space Div., and the S-IV stage by Douglas Aircraft. As an operational vehicle, SA-8 carried a Pegasus satellite.

In addition to being the first industry-build Saturn, SA-8 was the first Saturn rocket to be launched at night.

Like SA-9 and SA-10, SA-8 carried a live television system to observe the deployment of the Pegasus wings. In addition to its basic mission, SA-8 continued the studies of SA-9.

The SA-8 lofted the Pegasus satellite, the spent S-IV stage, and the Instrument Unit into an orbit with a minimum altitude (perigee) of 273 nautical miles, and a maximum altitude (apogee) of 420 nautical miles.

Its mission highlights included:

1. First Saturn launched with both stages built by private industry.

2. First night launch of a Saturn rocket.

3. Evaluation of meteoroid data sampling in near-Earth orbit.

4. Continuation of demonstration of functional operations of the mechanical, structural, and electrical subsystems of the Pegasus meteoroid satellite.

5. Continuation of the evaluation of the thermal coating for the exterior of the S-IV stage, Instrument Unit, and Service Module adapter extension.

6. Television coverage of the separation of the boilerplate Command Module and Service Module from the Pegasus satellite area.

7. Orbiting of 34,100 lb satellite consisting of spent S-IV stage, Instrument Unit, Pegasus satellite, and the Apollo Command Module.

SA-10 (JULY 30, 1965) - The SA-10 vehicle flight marked the close of the most phenomenally successful of the United States' space programs.

Like the two previous vehicles, SA-10 carried a Meteoroid Technology Satellite (Pegasus). Its mission was more complex as it not only measured meteoroid hazards, but also checked out the validity of its own data and those of SA-8 and SA-9. In addition, removable panels were attached to the satellite for possible later recovery by space-maneuvering astronauts.

The recovery of these panels would be of great scientific value, allowing examination of material that has been exposed to space conditions for prolonged periods of time. The information extracted from the recovered panels could contribute materially to future space station and lunar base design specifications.

SA-10 lofted its payload into an almost perfectly circular orbit, with a perigee of just under 286 nautical miles and an apogee of slightly more than 287 nautical miles.

The mission highlights included:

1. Final launch of the Saturn I series.

2. Determination of meteoroid penetration for three different thicknesses of aluminum.

3. Measurement of satellite radiation environment and panel temperature to evaluate the validity of data from meteoroid impacts.

4. Determination of the position and orientation of the satellite relative to the time of meteoroid impact.

5. Continued demonstration of the iterative guidance mode and evaluation of system accuracy.

6. Continuation of the evaluation of the thermal coating for the exterior of the S-IV stage, Instrument Unit, and Service module adapter extension.

7. Orbiting of 34,000 lb Pegasus satellite, including S-IV stage and Instrument Unit, into nearly perfect circular orbit.

SA-201 (FEBRUARY 26, 1966) - The flight of the first Apollo/Saturn IB (SA-201) followed in the tradition of the completely successful 10 missions of the Saturn I series.

This launch marked the first flight tests of a powered Apollo spacecraft, an S-IB stage, an S-IVB stage, and a J-2 engine. The two-stage vehicle achieved all test objectives in its 32 minute suborbital flight down the Atlantic Missile Range.

The Apollo spacecraft, which reached a peak altitude of 306 miles, splashed down about 200 miles southeast of Ascension Island in the South Atlantic.

Its mission highlights included:

1. Initial launch of the Saturn IB series.

2. First test of S-IB stage.

3. First test of S-IVB stage.

4. First test of J-2 engine.

5. First test of a powered Apollo spacecraft.

6. Confirmation of aerodynamic, dynamic, structural, and thermal characteristics.

7. Demonstrated S-IB/S-IVB/IU compatibility.

8. Evaluated S-IB/S-IVB short coast separation mode.

9. Evaluated the emergency detection system in open loop operation.

SA-203 (JULY 5, 1966) - The second Apollo/Saturn IB (SA-203) added to the impressive record of Saturn vehicles by making the 12th consecutive successful flight in as many launch attempts.

The SA-203 was topped by a simple aerodynamic shroud (nose cone) instead of an Apollo spacecraft. The S-IVB stage, Instrument Unit, and nose cone weighed a total of about 58,500 lb, the heaviest object launched into orbit by the United States. This included 10 tons of LH_2 in the S-IVB stage as the primary payload.

The primary purpose of the flight was to determine the cryogenic behavior in a near zero "g" environment, thus verifying the orbital conditioning characteristics of the second stage propulsion system, which uses liquid hydrogen as its fuel. This information was needed for future Saturn V applications in which the S-IVB stage must restart in Earth orbit.

The hydrogen continuous vent system was arranged to provide a very slight amount of thrust as the gaseous hydrogen produced by boiloff escaped. Additional thrust was provided periodically by opening a liquid oxygen tank propulsive vent valve. The information obtained was needed to determine if the thrust and resulting slight acceleration would keep the fuel settled in the bottom of the tank, where it would be available for use in restarting the J-2 engine.

The second stage was broken up near the beginning of the fifth orbit during a test of the hydrogen tank pressure rise rate and common bulkhead test. The structural failure of the bulkhead was planned, but the time of failure was uncertain.

All aspects of the flight, including general tests of the vehicle's propulsion and guidance systems and observation of the Instrument Unit's operation in orbit, were carried out satisfactorily.

Mission highlights included:

1. Heaviest object launched into orbit by the United States.

2. Test to determine the behavior of liquid hydrogen under weightless conditions.

3. Evaluated the structural integrity of the S-IVB common bulkhead.

4. Evaluated the emergency detection system in open loop operation.

5. Confirmation of aerodynamic, dynamic, structural, and thermal characteristics.

SA-202 (AUGUST 25, 1966) - The third unmanned Apollo/Saturn IB (SA-202) was the 13th successfully launched vehicle of the Saturn Program.

The primary purpose of this mission was a suborbital flight to test the Apollo spacecraft's heat shield. About 93 minutes after launch, the Saturn IB had hurled its payload three-fourths of the way around the earth. Then the Apollo's 21,500 lb thrust service engine carried the spacecraft to an altitude of more than 700 miles.

The Apollo command module made a "skipping" reentry into the atmosphere, somewhat like a roller coaster ride, subjecting the heat shield to extended high heat loads. The previous reentry test had been at a sharper angle, reducing time of reentry.

The emergency detection system was successfully flown "closed loop" on this flight.

Mission highlights included:

1. Extended high heat load test of the Apollo spacecraft heat shield.

2. Evaluated the emergency detection system in closed loop operation.

3. Continued demonstration of S-IB/S-IVB/IU compatibility.

4. Further evaluation of use of service propulsion system.

SA-204 (JANUARY 22, 1968) - The flight test of the SA-204 vehicle was the fourth in a series of Saturn IB R&D test flights.

The primary purpose of the mission was to verify operation and integrity of Lunar Module subsystems, Lunar Module staging, and S-IVB/IU orbital performance.

All portions of the orbital safing experiment were performed successfully, including propellant venting, propellant dump, cold helium dump, and stage and engine pneumatic supply dump. The stage pneumatic sphere pressure did not decrease to the expected level due to a higher than expected initial pressure. However, the rate and manner in which the sphere was vented were satisfactory.

Mission highlights included:

1. Verified operation and integrity of the lunar module subsystems.

2. Evaluated lunar module staging.

3. Evaluated S-IVB/IU orbital performance.

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