N 71-10465

NASA TECHNICAL
MEMORANDUM

NASA TM X-58054
November 1970

PAST AND PRESENT MANNED SPACECRAFT ELECTRONICS AND IMPLICATIONS FOR THE SPACE SHUTTLE

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Presented to the
XXIst International Astronautical Congress
Constance, German Federal Republic
October 4-10, 1970

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

PAST AND PRESENT MANNED-SPACECRAFT ELECTRONICS AND IMPLICATIONS FOR THE SPACE SHUTTLE

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INTRODUCTION

The electronic systems for the NASA manned-spacecraft programs have varied from the relatively simple systems used in Project Mercury to the much more sophisticated systems developed for the Apollo Program. This increase in sophistication has been in consonance with the progressively increasing requirements in these programs and with the electronic state-of-the-art characteristic of the time period of each program. The purpose of this paper is to compare the electronic systems used in these programs on a subsystem by subsystem basis and to consider any apparent trends or unique characteristics in the context of the known requirements for the space shuttle. The four spacecraft considered are the Mercury spacecraft, the Gemini spacecraft, the Apollo lunar module (LM), and the Apollo command and service module (CSM).

The mission characteristics for each spacecraft program pertinent to the avionics systems are first compared; then the developmental schedules of each program are considered briefly to lay a basis for the subsequent discussion. The electronics systems compared in this paper are the following.

1. Power generation and distribution
2. Guidance and navigation
3. Stabilization and control
4. Display and control
5. Caution and warning
6. Sequencing
7. Instrumentation
8. Communication and tracking

For each subsystem, a table of basic characteristics for the four spacecraft is presented and is used for a comparative discussion. Included in the discussions are some considerations of the subsystem redundancies. After the discussion of each subsystem, a comparison of the weight of the electronic subsystems is made for each spacecraft. Finally, the unique avionic characteristics and trends are summarized and the implications for the shuttle are briefly discussed.

MISSION CHARACTERISTICS

As shown in table I, the basic 7-day mission for the space shuttle is encompassed by the mission durations of the previous spacecraft. The shuttle is to be manned by a crew of two in the orbiter and two in the booster, if the booster is manned. This factor is also within the realm of previous experience. However, the shuttle is to carry 10 passengers,

TABLE I.- MISSION CHARACTERISTICS

Mission characteristics	Program/vehicle				
	Mercury	Gemini	Apollo CSM	Apollo LM	Space-shuttle orbiter
Approximate active mission time, days	1 1/2	2 to 14	11	3 1/2	7
Number of crewmen	1	2	1 to 3	2	2 + 10 passengers
Extravehicular activity	No	Yes	No	Yes	No
Maximum earth-vehicle altitude, n. mi.	153	739	220 000	220 000	<500
Approximate earth orbital inclination, deg	32.5	28.8 to 32.5	15 to 40	--	55
Approximate space maneuver velocity change, ft/sec	345 retro	1100	7100	13 400	18 000
Rendezvous and docking					
Target vehicle	No	No	Yes	Backup	Possible
Active vehicle	No	Yes	Backup	Yes	Yes
Atmospheric entry inertial velocity, ft/sec	25 900	25 900	36 200	--	25 900
Atmospheric cross-range maneuver, n. mi.	0	±50	±150	--	150 to 1500
Typical earth-landing accuracy, n. mi.	5	2.5	1.5	--	0.005

which is a new factor. The shuttle, like the Mercury and Gemini spacecraft, will be a low-altitude, low-orbital-energy, and thus low-atmospheric-entry-velocity vehicle rather than one with high energy, such as the Apollo spacecraft. However, the shuttle may deliver payloads to earth orbit that could reach synchronous earth-orbital altitudes in which they would approach the Apollo orbital energy. The spacecraft-maneuver velocity-change (ΔV) comparison is interesting because the 1500- to 2000-ft/sec on-orbit ΔV for the shuttle added to the ΔV increment, which must be applied and guided by the second stage, will exceed the maneuver ΔV of any of the previous spacecraft, including the higher-orbital-energy Apollo spacecraft. Of course, this increase results from the shuttle orbiter design characteristic of serving as its own second booster stage.

The shuttle will have a somewhat higher orbital inclination than that of previous U.S. spacecraft because it is to be designed to rendezvous and dock with the space base, which will have a 55° inclination orbit. The atmospheric cross-range maneuver capability of the shuttle, which is yet to be decided, could be limited to that of the Gemini spacecraft (or Apollo spacecraft in low-altitude earth orbit), or it could be as much as 1500 nautical miles, depending on the final configuration chosen. The shuttle, because it is to land on runways, will have a landing-accuracy requirement three orders of magnitude better than that achieved by the Gemini or Apollo spacecraft. It is not anticipated at present that the shuttle will have an extravehicular-activity (EVA) requirement.

DEVELOPMENTAL SCHEDULES

The developmental schedules of the three programs are shown in figure 1. In each case, the detail design of the spacecraft and the associated subsystems started at the milestone labelled "Contract signed." At this point, definitive spacecraft and subsystem concepts had evolved from the preliminary design and trade-off studies. The Mercury spacecraft was developed using 1958-1959 technology, the Gemini spacecraft with 1961-1962 technology, and the Apollo spacecraft with 1964-1965 technological state of the art. The time from the beginning of detail design to the first manned flight ranged from approximately 2 years on Project Mercury to 3-1/4 years in the case of the Gemini Program to 7 years and 6-1/2 years, respectively, for the Apollo CSM and LM.

ELECTRICAL POWER SYSTEM

Silver-zinc batteries were the sole source of electrical power in the Mercury spacecraft, whereas a combination of silver-zinc batteries and hydrogen-oxygen fuel cells was used in the Gemini spacecraft. The LM also has only silver-zinc batteries, while the battery/fuel-cell combination is used in the CSM (table II). Each of the two Gemini fuel-cell powerplants could produce as much as 1 kilowatt of power. Each powerplant had three independent sections that could be individually tied to the power buses. The normal power range for each of the CSM fuel-cell powerplants is 400 watts to 1.42 kilowatts. Each powerplant can be independently switched to the power buses by the crew.

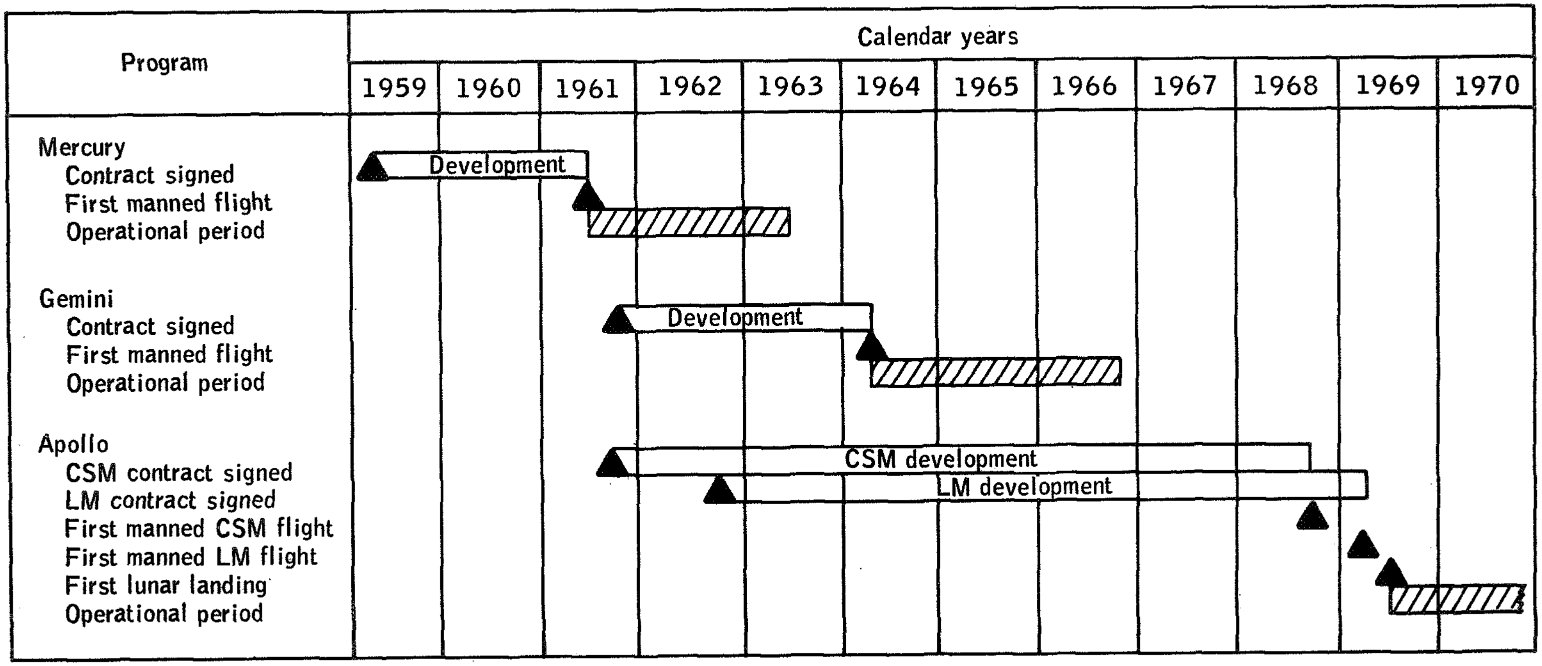

Figure 1.- Developmental schedules.

TABLE II.- ELECTRICAL POWER SYSTEM COMPARISON

Characteristics	Program/vehicle			
	Mercury	Gemini	Apollo CSM	Apollo LM
Power sources				
Fuel cells	None	2	3	None
Batteries (silver-zinc)	6	7	5	8
Battery capacity, W-h	16 500	6500	3500	6150
dc voltages or voltage ranges, V . . .	6, 12, 18, 24	22 to 30	25 to 30	25 to 32.5 (37)[a]
ac power supplies				
Number of central inverters	3	None	3	2
Capacity, VA, each inverter	150 and 250 (2)	--	1250	350
ac frequency, Hz	400 ± 1 percent	--	400 ± 0.5 percent	400 ± 1 percent
ac voltage, V	115 ± 5 percent	--	115 ± 1 percent	115 ± 1 percent
Power distribution and control				
dc buses	8	4	8	2
ac buses	2	None	2	2
Power switching	Power relays	Power relays	Power relays and motor switches	Power relays
Circuit protection	Fuses	Circuit breakers	Circuit breakers	Circuit breakers

[a]Voltage of pyrotechnic batteries.

Of the four spacecraft, only the CSM has an onboard battery charger. Of the three main batteries and two pyrotechnic batteries, only the main batteries can be charged. In the CSM, the main batteries are used to supplement the fuel-cell power output during peak demands. In the LM, the ascent-stage batteries are used in parallel with the descent-stage batteries during lunar descent to ensure adequate voltage levels. In both of these spacecraft, relatively sophisticated electrical-power control techniques are used in which high and low voltage, reverse current, and overtemperature conditions are sensed. In some cases, switching is accomplished automatically; in others, the caution and warning system alerts the crew. In the Mercury and Gemini spacecraft, the power sources were used more independently and sequentially. Diodes were used for reverse-current protection, but crew displays were principally "circuit on" indicators.

In the Gemini spacecraft, the pyrotechnic batteries were separate and isolated from the main batteries, as they are in the CSM and LM. The Gemini spacecraft had three pyrotechnic batteries, the CSM has two, and the LM has two. The Mercury spacecraft had a separate, isolated battery, which was for emergency squib firing and for emergency power to other circuits. In the Gemini spacecraft, as is presently the case in the CSM, the main batteries could also be used to back up the pyrotechnic batteries. Characteristically, in the Gemini spacecraft, power for initiating pyrotechnical functions was provided simultaneously from two separate power sources. This redundancy was carried over to the CSM and LM.

The Gemini spacecraft was the only one of the four spacecraft that has not had centralized alternating-current (ac) power conversion and distribution. Only direct-current (dc) power was distributed, and localized dc-to-ac inverters were used in those subsystems requiring ac power. In the CSM and the LM, as was the case in the Mercury spacecraft, 400-hertz, 115-volt centralized inverters are used to provide ac power to the subsystems via ac buses. The two primary inverters in the Mercury spacecraft (150-volt-ampere and 250-volt-ampere capacity) were backed up by a 250-volt-ampere standby inverter. In the CSM, three identical 1250-volt-ampere inverters are used, any one of which can satisfy all of the spacecraft ac power requirements. The LM has two inverters, each of which can provide the total power requirement.

In the Mercury spacecraft, multiple (main, standby, and isolated) buses were used to distribute dc power. The main bus operated at 24 volts and could be tied to the 24-volt standby and isolated buses. The 6-, 12-, and 18-volt standby buses and the 6- and 18-volt isolated buses were independent and could not be interconnected. Two ac buses were used, each powered by a main inverter. The standby inverter could feed either or both of the ac buses. In the Gemini spacecraft, a single dc main bus, a single dc control bus, and two redundant dc squib buses were used. No ac buses were used. The CSM has two redundant main dc buses powered by the three fuel cells or by the three batteries. In addition, the CSM is equipped with three battery buses that can be powered by two of the batteries, a nonessential system bus that can be powered by the main bus or by two of the batteries, and a flight and postlanding bus that can be powered from the main bus or from the batteries. Each of the two pyrotechnic buses has a separate battery and is isolated from the rest of the electrical power system. Two ac buses are used in the CSM. The LM has two redundant dc buses and two redundant ac buses. Most subsystems, particularly the critical ones, can receive power from both ac or both dc buses. The LM has two explosive-device (pyrotechnic) buses, each powered by a separate battery. Each explosive function is accomplished by two separate cartridges, each fired by one of the pyrotechnic buses.

Power relays were used for power switching in the Mercury and Gemini spacecraft, as they are in the CSM and LM. In addition, the CSM has motor switches for inverter switching, for switching the stabilization and control system (SCS) jet commands from the service module (SM) jets to the command module (CM) jets when the vehicles are separated, and for other switching functions.

The Gemini spacecraft was equipped with circuit breakers for circuit protection, as are the CSM and LM. Fuses and fuse switches were used in the Mercury spacecraft; but for some critical abort functions, solid connectors were used in lieu of fusing.

NAVIGATION AND GUIDANCE

Ground tracking was used in the Mercury and Gemini spacecraft, as it is in the CSM and LM, for on-orbit navigation (table III). The Gemini spacecraft, like the LM, also had the ability to propagate the ground-generated state vector in the onboard computer to predict future position.

TABLE III.- NAVIGATION AND GUIDANCE COMPARISON

Characteristics	Program/vehicle			
	Mercury	Gemini	Apollo CSM	Apollo LM
Spacecraft navigation-guidance functions				
Launch-vehicle guidance	--	Backup first and second stage	Backup Saturn launch vehicle GIMU[a]	Lunar ascent
On-orbit navigation	Ground	Ground plus onboard predictive	Ground plus star-landmark	Ground plus onboard predictive
Rendezvous guidance	--	Onboard	Optical backup	Onboard
Earth-return targeting	Ground	Ground	Ground plus onboard	--
Entry guidance	--	Onboard	Onboard	--
Inertial sensors				
Gimbaled inertial measurement units	--	1	1	1
Strapdown inertial measurement units	--	--	--	1
Body-mounted accelerometers	1 accelerometer switch	--	1	--
Body-mounted attitude gyroscopes	2	--	3	--
Rate gyroscopes	3	6	3	3
Computers (number)	--	1	1	2
Memory size, words	--	4096	38 912	38 912 and 4096
Word length, bits	--	39	16	16 and 18
Keyboard and display units	--	1	2	2
Optical sensors				
Horizon scanners	2	2	--	--
Sextants (field of view, deg)	--	--	1 (1.8)	--
Telescopes (field of view, deg)	--	--	1 (60)	1 (60)
Rendezvous radar (range, n. mi.)	--	1 (180)	--	1 (400)
Landing radar (range, feet)	--	--	--	50 000

[a]Gimbaled inertial measurement unit.

The CSM also has the capability of making onboard star-horizon and star-landmark measurements to determine and refine its state vector. The Mercury spacecraft had no launch-vehicle guidance capability, but the Gemini navigation and guidance system backed up the first- and second-stage launch-vehicle guidance system, and the CSM can back up a failure of the Saturn launch-vehicle gimbaled inertial measurement unit (GIMU). The LM launch-vehicle guidance from the lunar surface into lunar orbit is accomplished by the primary guidance, navigation, and control system (PGNCS) and is backed up by the abort guidance system (AGS).

The Gemini onboard computer solved the rendezvous equations using targeting information from the ground and the inputs from the rendezvous radar. This is also the case with the LM. The CSM has no radar but uses targeting information from the ground and line-of-sight angles (referenced to the inertial measurement unit (IMU)) generated with a manually operated sextant to solve the rendezvous equations. The CSM rendezvous capability is a backup to the LM capability, which is primary. The first Apollo manned rendezvous was a CSM optical rendezvous. The LM/CSM very-high-frequency (vhf) communications link is used as a supplementary source of range information for the rendezvous. The Mercury spacecraft, of course, had no rendezvous capability, nor did it have an entry guidance capability, because it flew a ballistic entry trajectory. Primary earth-return targeting was accomplished by ground tracking for the Mercury and Gemini spacecraft, as it is for the CSM. The CSM can also accomplish earth-return targeting on board.

The Mercury spacecraft had no navigation and guidance system as such; the sensors were all part of the SCS. The two body-mounted gyroscopes (vertical and directional) were free gyroscopes that could be slaved using the outputs of each of the single degree-of-freedom (pitch and yaw) horizon-sensor heads. In addition, three rate gyroscopes were used for flight-control-system damping, and an accelerometer switch with a threshold of 0.05g was provided to indicate entry initiation. None of this sensing equipment was redundant. A single GIMU having four gimbals, a computer having a memory containing 4096 39-bit words, two two-axis horizon-scanner heads, a 180-nautical-mile-range rendezvous radar, and two three-axis rate gyroscope packages constituted the guidance, navigation, and control sensing equipment on the Gemini spacecraft. The navigation and guidance system was simplex, but most of the stabilization and control equipment was duplicated. The later Gemini flights (GT-8 and the subsequent Gemini missions) also had an auxiliary tape memory in which additional computer programs could be stored and read into the core memory of the computer for use when required. The programs were stored on the tape in a triply redundant form, and a majority voting scheme was used to minimize the bit-dropout problem when transferring a program from the tape to the core memory of the computer.

The CSM and the LM use identical primary guidance, navigation, and control systems consisting of a three-gimbaled IMU; a computer having a fixed memory of 36 864 16-bit words and an erasable memory of 2048 16-bit words; an electroluminescent computer display and keyboard (DSKY) (two in the CSM); and identical power supplies, platform electronics, and interfacing equipment. In both the LM and the CSM, attitude-control and thrust-vector-control digital autopilots are used in the PGNCS computer as the primary means of spacecraft control. The LM PGNCS guidance function is backed up by an AGS, which consists of a strapdown inertial measurement unit (SIMU) and a 4096-word computer with its own display and keyboard. The LM also uses a single three-axis rate gyroscope package for the stabilization and control function and a landing radar for use during the terminal lunar descent. The CSM PGNCS guidance function is backed up by the analog SCS, which uses pulse-torqued body-mounted attitude gyroscopes as an attitude reference and a panel-mounted X-axis accelerometer feeding a ΔV counter as an incremental velocity indicator. Two triads of body-mounted attitude gyroscopes are provided. One set is normally used in an attitude-sensing mode but can be switched to a rate-sensing mode to back up the other body-mounted-attitude-gyroscope package, which normally provides rate information, if required.

In the Gemini spacecraft, the outputs of the horizon scanners and gyrocompassing were used for alining the GIMU. In the CSM, either the scanning telescope, which has a unity power, or the sextant, which has an optical power of 28, is used for GIMU alinement star sightings. The scanning telescope has a single articulated line of sight, and the sextant has two lines of sight: one is fixed to the spacecraft and the other is articulated. The two instruments have a computer angle readout and are slaved together. They can also be used independently. The sextant is also used to make star-horizon and star-landmark measurements for orbital navigation purposes and line-of-sight measurements to the

LM for rendezvous. The LM has a 60° field of view, unity-power, alinement optical telescope for GIMU alinement sightings. It can be used either in flight or on the lunar surface. The information is read out manually and entered into the PGNCS computer via the DSKY.

STABILIZATION AND CONTROL

The stabilization and control characteristics for the four spacecraft are shown in table IV. The control-system organizations for the Gemini and Mercury spacecraft were similar to each other and to those now in use for the CSM and LM. In general, control-system electronics for the Gemini and Mercury spacecraft, as for the CSM and LM, were analog and were interfaced with the reaction control system (RCS) jet solenoids and any servoactuators in the system through driver amplifiers. Any jet-select logic was included in the control-system electronics. The system was moded from the cockpit to close the control loops around any of the available control references, such as body-mounted gyroscopes, inertial measurement units, horizon scanners, and so on, and to accept commands from sources such as the rotational hand controllers, onboard computers, and, in some cases, from the ground. Also, the LM and CSM use digital autopilots in the computers to close the stabilization and control loops. In fact, this method is the primary control path for these vehicles. Jet-select logic is included in the digital autopilot (DAP) and allows the system to operate more efficiently than do the analog systems because, with the DAP, firing jets in opposition and thus wasting fuel is avoided. This usually cannot be avoided with the analog

TABLE IV.- STABILIZATION AND CONTROL COMPARISON

Characteristics	Program/vehicle			
	Mercury	Gemini	Apollo CSM	Apollo LM
Control references				
Local vertical	Vertical gyroscope/ horizon scanner	GIMU/horizon scanner	ORDEAL[a]	ORDEAL
Azimuth reference	Yaw gyroscope/ horizon scanner	Gyrocompass	GIMU/BMAG[b]	GIMU/SIMU
Inertial	--	GIMU	GIMU/BMAG	GIMU/SIMU
Line of sight	--	Radar	--	--
Attitude displays				
Three-axis rate and attitude needles	1 display	--	--	--
Flight director attitude indicator	--	2	2	2
Hand controllers				
Rotational (three-axis)	1	1	2	2
Translational (three-axis) . . .	--	1	1	2
Attitude control modes available				
Rate command	X	X	X	X
Pulse	--	X	X	X
Direct	Mechanical	X	X	X
Acceleration command	X	--	X	X
Automatic control	Analog system	Analog system	DAP[c] and analog system	DAP and analog system
Thrust vector control				
Gimbaled main engine	--	--	DAP, analog system, and direct	DAP and analog system
Fixed main engine	--	--	--	DAP and analog system
Translation thrusters	--	Direct only	DAP, analog system, and direct	DAP and analog system

[a]Orbital rate display, earth and lunar.
[b]Body-mounted attitude gyroscope.
[c]Digital autopilot.

systems. The commands from the DAP to the jets or the servoactuators still go through the driver-amplifier portion of the control electronics. An additional advantage of the DAP system is that changes in gains or filtering can be made with software changes rather than with hardware changes as in the analog system.

The Mercury spacecraft had only a local-vertical reference frame, which was provided by a combination of roll and pitch horizon scanners and vertical and directional free gyroscopes. The Gemini spacecraft used a local-vertical reference frame provided either by the GIMU alined-by-horizon sensors (with azimuth provided by gyrocompassing) or by the pitch and roll horizon scanners alone (with the azimuth axis unconstrained in attitude but rate limited). The Gemini spacecraft also had available an automatic, inertially fixed attitude-reference frame and a manually controlled line-of-sight reference frame using target elevation and azimuth information from the rendezvous radar and roll information from the GIMU. This latter reference frame was used for all rendezvous missions because the Gemini rendezvous radar was body-fixed and boresighted to the spacecraft X-axis. Neither the LM nor the CSM uses horizon sensors; and, although a local-vertical reference could have been provided computationally with a software routine in the PGNCS computer (this capability is currently being programed into the computers in the Skylab Program command and service modules), the priority for this reference was low enough that it was not included in the fixed memory. However, a unit called the orbital rate display, earth and lunar (ORDEAL) is included in both the LM and CSM. This unit electrically drives the flight director attitude indicator (FDAI) in pitch to provide a local-vertical reference display. Controls on the unit allow the selection of earth or lunar orbits. The principal control reference frame in the CSM and LM is the inertial reference provided by the spacecraft inertial sensors.

The Mercury spacecraft had a three-axis rate/attitude display with three rate needles and three attitude needles. The Gemini spacecraft had redundant flight director attitude indicators as do the CSM and LM. Manual control inputs were provided by means of three-axis rotational hand controllers in the Mercury and Gemini spacecraft, as in the CSM and LM. Three-axis translational controllers were used in the Gemini spacecraft as they are in the CSM and LM. Both rotational and translational hand controllers are redundant on the LM. The CSM has two rotational hand controllers and one translational controller, and the Gemini spacecraft was equipped with one of each. The Mercury hand controller was unique in that, in addition to the acceleration commands it provided by closing limit switches to fire the primary RCS thrusters and in addition to the rate-command input to the automatic SCS, the hand controller had a direct mechanical linkage to the valves of an independent RCS for emergency use. The LM and CSM have rate-command, pulse, and direct-control modes available to the crew, as did the Gemini spacecraft. The LM and the CSM have separate solenoid coils in the thruster valves; these coils are energized by the direct-control-mode switch closures in the hand controllers. The Gemini direct mode bypassed the control electronics and energized the single set of solenoid coils in the thruster valves. The LM and CSM also have an acceleration command available by deflecting the hand controllers beyond preset limits.

The Gemini analog-control-system electronics, including the solenoid drivers, power supplies, and rate-gyroscope packages, were duplicated. The duplicate equipment was wired in as spares that could be selected by panel switching. The six 100-pound and two 85-pound translation thrusters were independent of the 25-pound attitude-control thrusters. The translation thrusters could be fired only with the manual maneuvering controller, which applied dc voltages directly to the thruster coils. Sixteen 100-pound thrusters on the LM and SM provide both translational and rotational control. The 12 thrusters on the CM provide only rotational control. Either the digital autopilots or the jet-select logic in the LM and CSM analog stabilization and control systems can be used to fire the appropriate jets for the required functions. The LM has two sets of solenoid drivers for the thrusters; the CSM has one set.

The primary means for altering the velocity vector of the vehicle are the service propulsion system (SPS) engine on the CSM and the descent and ascent engines on the LM. The SPS engine is gimbaled, and electromechanical servoactuators can be controlled either by the DAP, by the SCS, or in a direct mode by the pilot. The LM descent engine is gimbaled only for trimming out misalinements with the center of gravity. The electromechanical trim servoactuators can be driven either by the descent DAP or by the SCS. Short-period attitude control with the fixed ascent engine is provided during both lunar descent and lunar ascent by use of the LM attitude-control thrusters.

DISPLAYS AND CONTROLS

The display and control panels (table V) show a marked increase in size and complexity from those on the Mercury spacecraft to those used on the CSM and LM. The Mercury spacecraft had three relatively small panels arranged as a main panel in front of the pilot and right and left consoles. The Gemini spacecraft had seven panels arranged as a center panel, a center console, a center overhead switch and circuit-breaker panel, command pilot and pilot panels, and right and left circuit-breaker panels. Piloting functions were concentrated at the left, or command pilot's, station; engineering (e.g., electrical-power control) and navigation functions were concentrated at the right, or pilot's, station. This same pattern tends to be maintained in the CSM and LM. On the Gemini spacecraft, either crewmember could manipulate switches and knobs at the other crewman's station by use of a "swizzle stick" stored overhead. This capability was important because the vehicle was operated by one crewman during the Gemini extravehicular activities. The majority of the CSM displays and controls are located on the main display console, which faces the three crewmen when they are in their couches and which is made up of 21 separate panels. The console is nearly 7 feet long and 3 feet high with two 3- by 2-foot wings on either side. Other panels are located in the lower equipment bay, where the guidance and navigation

TABLE V.- DISPLAY AND CONTROL COMPARISON

Characteristics	Program/vehicle			
	Mercury	Gemini	Apollo CSM	Apollo LM
Number of display-and-control-panel segments	3	7	25	12
Total number of switches and circuit breakers	(66)	(239)	(611)	(338)
Toggle switches	48	102	302	144
Circuit breakers	--	106	262	160
Rotary switches	6	11	13	14
Pushbutton switches	12	20	13	7
Potentiometers	--	--	21	13
Meters and display devices	(46)	(48)	(88)	(81)
Single meters	5	2	13	9
Dual meters	6	8	18	12
Event indicators	31	27	43	53
Unique displays	4	11	14	7
Meter movements	D'Arsonval	D'Arsonval	D'Arsonval	Servometric
Display lighting				
Meter displays	Floodlights	Floodlights	Electroluminescent	Electroluminescent
Annunciators	Incandescent	Incandescent	Incandescent	Incandescent

equipment is located, and in the right and left equipment bays, where the environmental control system and waste management panels are located. The LM displays and controls are arranged to be operated by standing crewmen. Four panels are located in the center between the crewmen. Two are approximately vertical at about eye level, and two are located below these and are slanted at approximately 45° to the horizontal. A panel is located in front of each crewman at about waist height, and the rest of the panels are located on the right and left sides of the cabin.

The total number of switches and circuit breakers per crewman varied from 66 per man in the Mercury spacecraft to 119 per man in the Gemini spacecraft to 169 per man on the LM and 203 per man on the CSM with three crewmembers in the cockpit. The Mercury and Gemini spacecraft had no potentiometers, but a number are used in the LM and CSM for such functions as volume control and temperature control. Although the total number of switches and circuit breakers varied almost an order of magnitude between the Mercury spacecraft and the CSM, the total number of meters and display devices differs only by a factor of less than two. In terms of economy of displays per man, the Gemini spacecraft had the fewest, at 24 per man, and the LM has the highest, at approximately 40 per man. There is not a large range of unique devices used, such as attitude indicators, range and range-rate indicators, clocks, computer displays and keyboards — that is, devices other than voltage and current indicators or quantity gages. Meter movements are servometric on the LM and were D'Arsonval on the Mercury and Gemini spacecraft, as they are on the CSM.

Instruments were rear-mounted to the panels in the Mercury and Gemini spacecraft, and this method has continued to be used for the CSM and LM. The Mercury and Gemini instruments were environmentally sealed, and those of the LM and CSM are hermetically sealed. Meter displays are electroluminescent on the LM and CSM and were floodlighted on the Mercury and Gemini spacecraft. Incandescent annunciators are used on the CSM and LM, as they were on the Mercury and Gemini spacecraft.

CAUTION AND WARNING SYSTEMS

The caution and warning function (table VI) is defined as the process of alerting or notifying the crew that an out-of-nominal condition exists. The Mercury and Gemini spacecraft had no separately identifiable caution and warning system. Some caution and warning functions were included as part of the Mercury and Gemini sequencing system, which used telelights to provide the crew with indications of nominal events and sequences. Thus, only the Mercury and Gemini functions that are caution and warning are included in the comparison with the LM and CSM caution and warning systems. The LM and CSM systems are separately identifiable.

The Mercury spacecraft had nine inputs, all discretes, which used relay logic to light advisory indicators on the main instrument panel when out-of-nominal conditions occurred. The inputs came from the critical subsystems, such as the environmental control system, and were available to the crewman throughout the mission. Most of the Gemini caution and warning functions were concentrated on the launch phase and were provided as inputs to the crew for use in the onboard launch-abort decision that they could

TABLE VI.- CAUTION AND WARNING SYSTEM COMPARISON

Characteristics	Program/vehicle			
	Mercury	Gemini	Apollo CSM	Apollo LM
Inputs	(9)	(10)	(64)	(145)
Discretes	9	10	22	60
Analog signals	--	--	42	45
Inhibit signals	--	--	--	10
Resetable inputs	--	--	--	22
Enable signals	--	--	--	8
Outputs	(9)	(10)	(35)	(35)
Advisory indicators	9	10	34	33
Master-alarm indicators	--	--	3	1
Tone generator (audible)	--	--	1	1
Logic				
Relay output closures	X	X	--	X
Solid-state output closure . . .	--	--	X	--
Automatic master-alarm reset	--	--	--	X
Inhibits and enables	--	--	--	X

make, either in parallel with the ground or independently. Seven discrete inputs provided information about out-of-tolerance conditions in the first- and second-stage launch-vehicle propulsion and guidance and control systems. Launch-vehicle fuel quantity and longitudinal acceleration meters were also available to the crew, but without logic and automatic-annunciation capability. The crew had to monitor these meters to gain the information desired. Relay output closure was used to light the seven advisory indicators. In addition to these, two telelights driven by logic in the IMU electronics were used to indicate a malfunction in the acceleration portion and in the attitude portions of the GIMU, and a self-test routine was used in the computer to detect a failure and to light the computer malfunction telelight. The LM and CSM use relatively sophisticated caution and warning systems wherein multiple inputs are provided to transistor/diode logic voltage comparators. In addition, the LM has inhibit and enable signals as well as automatic resets in the logic. In the CSM, solid-state output closures are used to light the annunciators, and relay closures are used in the LM. The LM has 145 inputs to the caution and warning system, and the CSM has 64. In both spacecraft, an audible tone warning is provided in the crewmen's headsets and a resettable master-alarm indicator is lit. When the crewman pushes the master-alarm light, it is extinguished and the tone is turned off, but the discrete caution and warning lights remain on until the problem is remedied. The CSM has three master-alarm indicators in different places in the spacecraft. The LM has one.

SEQUENCING

Dual redundancy was used in sequencing all critical functions in the Mercury and Gemini spacecraft, and the redundant auotmatic sequencing systems were backed up by manual switches (table VII). This approach also is used in the CSM and LM. In addition, the Mercury spacecraft had a separate, manually operated parachute and landing-bag deployment system in which the pyrotechnical deployment devices were mechanically initiated. The approximate number of relays devoted to the sequencing functions range from a minimum of 80 on the LM to a maximum of 160 on the CSM; the Mercury and Gemini spacecraft had 100 and 150, respectively. The CSM has a dual-series relay arrangement in which the sensitive axes of the relays are normal to each other to minimize the probability of inadvertent operation. The Mercury and Gemini spacecraft had single relays per function in each system as does the LM. The CSM has a separately identifiable sequencing subsystem with 12 sequence controllers. The Gemini system also was identified separately and consisted of 15 relay panels. The Mercury sequencing relays were distributed throughout the spacecraft in the subsystems containing the functions. The LM has two explosive-device relay boxes as part of the dual arrangement of sequencers. However, the rest of the sequencing functions, such as those interfacing with the explosive-device relay boxes, are included in the interfacing subsystems.

TABLE VII.- SEQUENCING COMPARISON

Characteristic	Program/vehicle			
	Mercury	Gemini	Apollo CSM	Apollo LM
System redundancy				
Automatic	Dual	Dual	Dual	Dual
Manual backup	Manual switch	Manual switch	Manual switch	Manual switch
Emergency system	Parachute deploy	--	--	--
Number of relays (approximate)	100	150	160	80
Relay configuration per function . . .	Single	Single	Dual-series	Single
System assemblies	Distrib-uted relays	15 relay panels	12 sequence controllers	2 explosive-device relay boxes plus distributed relays
Sequence timing				
Time-delay relays	X	X	--	--
Time-delay circuits	--	--	X	X
Event timers	--	X	X	--

Sequence timing was provided by time-delay relays in the Mercury spacecraft. The Gemini spacecraft used event timers and time-delay relays. The LM uses time-delay electronic circuits, and the CSM uses event timers and time-delay electronic circuits.

INSTRUMENTATION SYSTEM

In general, the instrumentation consists of measurement transducers, signal conditioners, encoders, and recorders. After the information collected by the transducers is conditioned to some standard reference, it is used to drive display devices in the cabin, either encoded and recorded or encoded and telemetered to the ground. The approximate total number of inflight measurement points used on each spacecraft and the number of measurements used for several functions are shown in table VIII. The total number of onboard measurements per spacecraft has roughly doubled from program to program.

The Mercury instrumentation system consisted of a number of electronic packages located in the cabin. The signal-conditioning function was accomplished in these packages. No standard signal-conditioner modules were used. Circuitry, as necessary, was added to condition each measurement to a 0- to 3-volt dc base before the measurement was commutated. In some cases, the transducers themselves (e.g., cabin air temperature and pressure) were included in the packages because they were mounted in the cabin. Instrumentation data voltages were commutated

TABLE VIII.- INSTRUMENTATION SYSTEM COMPARISON

Characteristics	Program/vehicle			
	Mercury	Gemini	Apollo CSM	Apollo LM
Measurement points (approximate total in flight)	100	225	469	473
Caution and warning	9	10	64	145
On board display	53	74	274	214
Telemetry	85	202	336	279
Signal conditioners				
Number of signal-conditioner packages	--	2	1	2
Number of modules or channels	Not standaridzed	79 modules	145 channels	246 channels
Encoding method	PAM/PDM	PCM	PCM	PCM
Time-reference equipment	Clock analog output	Crystal oscillator	Crystal oscillator and time pulses	Crystal oscillator and time pulses
Tape recorders (number)	1	4	2	1
Characteristics	7 channel record only 1 min/10 min	(1) PCM, 2 channel, 4 hr (1) Voice, 2 channel, 1 hr (2) Biomedical, 7 channel, 75 hr	(1) 5 PCM channels 1 LM data channel 1 voice channel 7 analog channels 30 min/2 hr (1) Hand-held voice recorder	(1) 1 voice channel 1 data channel 10 hr

and converted to pulse-duration-modulation (PDM) and pulse-amplitude-modulation (PAM) outputs. The PDM data were recorded on the spacecraft tape recorder. The PAM data and other instrumentation voltages were applied to several voltage-controlled oscillators. A mixer circuit was used to combine the subcarrier-oscillator outputs; then the mixer output was applied to either the telemetry transmitter or the tape recorder. The Gemini signal conditioners were modularized, and 79 of them were included in two signal-conditioning packages. Six basic types of signal-conditioning modules were used, and several of these had additional variations for different signal-handling capabilities. A multiplexer/encoder system was used to convert the signals to a serial binary-coded signal for recording on the pulse-code-modulation (PCM) data-dump recorder or for presentation to the real-time transmitter.

Eight different types of conditioners are included in two signal-conditioning packages on the LM. Each signal-conditioning package has 22 modules and, together, they provide 246 signal-conditioning channels. In the CSM, all signal conditioners are in one package, which has a redundant power supply. Four different signal-conditioner types are used to provide 145 signal-conditioning channels. Both the LM and CSM use a 1024-kilohertz signal from the guidance and navigation computers to synchronize PCM-timing electronics. If the synchronization pulse fails, both the CSM and LM have a separate crystal oscillator. Input signals to the PCM equipment in both spacecraft are high-level analog, parallel digital, and serial digital. The PCM data rates for both spacecraft include a normal rate of 51.2 kbps and a reduced rate of 1.6 kbps. Tape recorders are used in the CSM and LM as they were on the Mercury and Gemini spacecraft. The Mercury spacecraft had a seven-channel recorder that operated continuously from umbilical release until 5 minutes after spacecraft separation and continuously from 30 minutes before retrograde firing until 10 minutes after splashdown. During the rest of the mission, the recorder was on for 1 minute every 10 minutes. In addition to this automatic operation, the crewman could operate the recorder manually whenever he chose to do so. The Gemini spacecraft used four different recorders. One was a two-channel PCM recorder that had a 4-hour capacity and that was used to record PCM data for delayed-time dump. A cartridge-loaded two-channel voice recorder was available to the crew. Each cartridge, which could be easily changed, provided a 1-hour capacity. Two seven-channel biomedical recorders were provided. Each was controlled by the crew and could operate 75 hours at normal tape speed. The CSM data recorder has 14 channels: five PCM channels, one LM-data channel, one voice channel, and seven analog channels. Operating time is 30 minutes at high speed and 2 hours at low speed. The recorder is used principally for recording information for delayed dump. The LM recorder has two channels, one for voice and one for data, and can operate for 10 hours. Operation is either manual or semiautomatic.

COMMUNICATIONS AND TRACKING SYSTEMS

As shown in table IX, the high-frequency (hf) and vhf bands were used for voice communication and the ultrahigh-frequency (uhf) band for receiving command signals in the Mercury and Gemini spacecraft. The vhf band and the S-band are used in the CSM and LM radio-frequency (rf) requirements. The Mercury spacecraft was equipped with one hf transceiver and two vhf transceivers for voice, two vhf transmitters for telemetry, and two uhf frequency-modulation (FM) receiver/decoders for receiving commands. The Gemini spacecraft also had one hf and one vhf transceiver in conjunction with a voice-control center for voice communications, but had three separate vhf telemetry transmitters — a real-time transmitter, a delayed-time transmitter, and a standby transmitter for use if the other two failed. The Gemini spacecraft had two separate uhf command receivers feeding a common decoder. The CSM and the LM are equipped with a unified S-band system (USBS) which is used primarily for communications with the Manned Space Flight Network (MSFN) on earth, and vhf systems which are used primarily for LM/CSM communications and for EVA communications. The USBS in both spacecraft has primary and secondary transmitter-receiver and power-amplifier assemblies that serve the multiple functions of providing a transponder for MSFN space tracking and providing voice transceiving, telemetry-transmitting, and command-receiving functions. In addition, the CSM has a separate

TABLE IX.- COMMUNICATIONS AND TRACKING SYSTEMS COMPARISON

Characteristics	Program/vehicle			
	Mercury	Gemini	Apollo CSM	Apollo LM
Voice (number of transceivers)	3	2	4	4
hf, MHz	(1) 15	(1) 15.016		
vhf, MHz	(2) 296.8	(1) 296.8	(1) 296.8 (1) 259.7	(1) 296.8 (1) 259.7
S-band, MHz			(2) 2106.4 R[a] 2287.5 T[b} USBE[c]	(2) 2101.8 R 2282.5 T USBE
Telemetry (number of transmitters)	2	3	3	3
vhf, MHz	(1) 225.7 (1) 259.7	(1) 230.4 (1) 246.3 (1) 259.7		(1) 259.7
S-band, MHz			(2) 2287.5 USBE (1) 2272.5	(2) 2282.5 USBE
Command (number of receivers)	2	2	2	2
uhf, MHz	(2) 450	(2) 450		
S-band, MHz			(2) 2106.4 R 2287.5 T USBE	(2) 2101.8 R 2282.5 T USBE
Beacons (number)	5	5	4	2
hf, MHz (recovery and rescue)	(1) 8.364			
vhf, MHz (recovery and rescue)	(2) 243	(1) 246.3 (1) 243	(1) 243	
C-band, MHz (space tracking)	(1) 5400 to 5900	(1) 5765 T 5690 R		
S-band, MHz (space tracking)	(1) 2700 to 2900	(1) 2840 R 2910 T	(2) 2106.4 R 2287.5 T USBE	(2) 2101.8 R 2282.5 T USBE
L-band, MHz (rendezvous tracking)		(1) 1528 R 1428 T		
X-band, MHz (rendezvous tracking)			(1) 9832.8 R 9792 T	

[a]Receive.
[b]Transmit.
[c]Unified S-band equipment.

TABLE IX.- COMMUNICATIONS AND TRACKING SYSTEMS COMPARISON - Concluded

Characteristics	Program/vehicle			
	Mercury	Gemini	Apollo CSM	Apollo LM
Radar (number)		1		2
L-band, MHz (pulsed interferometer)		(1) 1528 T[a] 1428 R[b]		
X-band, MHz (monopulse rendezvous)				(1) 9832.8 T 9792 R
X-band, GHz (landing)				(1) 9.58 and 10.51
Antennas (number)	9	10	10	8
hf	(1) Whip	(2) Whip		
vhf	(1) Bicone[c] (1) Fan	(1) Stub (2) Blades	(2) Blade (2) Scimitar	(2) Helix
uhf		(2) Whip		(1) EVA omnidirectional
C-band	(3) Omnidirectional helix	(1) Slotted		
S-band	(3) Omnidirectional helix	(1) Slotted	(4) Helical (1) Steerable, 4 31-in.-diam parabolas	(2) Omnidirectional (1) Steerable 26-in.-diam parabola (1) Erectable 10-ft diameter
L-band		(1) Helix		
X-band			(1) Beam splitter	(1) Steerable 24-in.-diam parabola

[a]Transmit.
[b]Receive.
[c]Used also for hf and vhf.

S-band FM telemetry transmitter. Each spacecraft has two vhf/amplitude-modulation (AM) transceivers, which are used for the following functions.

1. LM to CSM, two-way voice

2. LM to CSM, low-bit-rate telemetry

3. CSM to MSFN, two-way voice

4. EVA astronaut to LM, two-way voice and data to LM

5. EVA astronaut to CSM, two-way voice and data to CSM

6. LM to CSM ranging (using time delay of transmitted round-trip tones)

In addition to these functions, a voice link is possible among EVA astronauts, the MSFN, and the CSM using the CSM or the LM as a relay and using a combination of the S-band and vhf capabilities.

Both the LM and CSM have audio centers fed by individual audio stations, one for each astronaut. A premodulation processor in each spacecraft provides signal modulation, mixing, and switching for all data and voice communications in accordance with the selected mode of operation. The command link (or up-data link) on both the LM and the CSM is used to update the spacecraft computers and the central timing equipment when necessary.

Both the Gemini and Mercury spacecraft had five rf beacons each, while the CSM has four and the LM has two. The Mercury spacecraft had one hf beacon and two vhf beacons for recovery and rescue, while the Gemini had two vhf beacons and the CSM has one vhf beacon for this purpose. The CSM and LM have the unified S-band transponder for MSFN space tracking, and the Gemini and Mercury spacecraft each had one C-band beacon and one S-band beacon. For rendezvous, the CSM carries one X-band beacon for use with the LM rendezvous radar. The Gemini spacecraft normally did not carry a rendezvous beacon. However, on the Gemini VII and VI missions, Gemini VII was fitted with an L-band beacon for use with the Gemini VI rendezvous radar. This was the same beacon that was normally affixed to the Agena target vehicle used in the program.

A body-fixed pulsed interferometer L-band radar was carried on the Gemini spacecraft for rendezvous. A gimbaled X-band monopulse rendezvous radar and an X-band landing radar that provides altitude and landing-velocity information are carried on the LM.

Nine antennas were carried on the Mercury spacecraft. A main bicone antenna was used for the hf, vhf, and uhf equipment. In addition, a uhf recovery fan antenna was deployed during entry, and an hf recovery whip antenna was deployed after splashdown. Six omnidirectional helices were used, three for the C-band and three for the S-band beacons. The Gemini spacecraft had 10 antennas. An extendable hf whip antenna 13 to 16 feet long was used in orbit and another hf antenna, 13 feet long when deployed, was used for recovery. Two vhf descent and recovery blade antennas were used only from the time that the vehicle was on a two-point suspension on the parachute through recovery. A vhf stub antenna was used for telemetry, for the voice transceivers, and for reception for one of the digital command receivers. Two slotted antennas were used, one for the C-band beacon and one for the S-band beacon. Finally, the L-band rendezvous-radar antenna was a helix. The CSM also has 10 antennas. Two vhf scimitar omnidirectional antennas are used in orbit, and two recovery vhf blade antennas are deployed automatically shortly after the main parachutes are deployed. The spacecraft has five S-band antennas. Four are omnidirectional helical antennas and the fifth is the high-gain steerable antenna that is stored at the base of the SM adjacent to the SPS engine bell and that is deployed after the LM has been withdrawn from the Saturn IVB stage. This antenna consists of four 31-inch-diameter parabolas on a gimbal structure. The 10th CSM antenna is a single X-band beam-slitter antenna used for the rendezvous transponder. The LM has eight antennas. Four of these are S-band antennas: two omnidirectional helices, one steerable high-gain 24-inch-diameter parabola, and one 10-foot-diameter errectable parabola for use on the lunar surface. Three vhf antennas are used on the LM; two are circularly polarized helices for use in flight and the other is a conical omnidirectional EVA antenna. The eighth antenna is the 24-inch-diameter steerable parabola that serves as the rendezvous-radar antenna. Some form of antenna multiplexing was used in the Mercury and Gemini spacecraft and is used in the CSM and LM.

ELECTRONIC-SUBSYSTEM WEIGHTS

A summary of the weights of each of the electronic subsystems considered is presented in table X. The percentage of the respective spacecraft weights that these weight increments represented are also indicated. The spacecraft weights used for reference were dry weights and did not include launch escape towers, booster adapters, or propellants and expendables. It should be noted that all of the weights in this table are representative weights, and that the weights of any given spacecraft and subsystem might vary somewhat from those given, depending on the specific mission. Also, some caution must be used because it is difficult to get truly comparable weights from one program to the next. Notwithstanding these reservations, it is interesting to make comparisons both on the basis of absolute weight and on the basis of percentage of spacecraft weight. In general, the Mercury spacecraft had the largest percentage of spacecraft weight allocated to electronic systems. This was the case despite the facts that the Mercury spacecraft carried only one crewman instead of two or three like the other spacecraft, had a relatively short mission time, and carried no guidance and navigation system. This large percentage of spacecraft weight allocated to electronics was attributable to a conservative design approach because this was the first manned spacecraft and because the earlier state of the art was used. No caution and warning or sequencing-system weights could be explicitly identified because these functions

TABLE X.- ELECTRONIC-SYSTEM WEIGHT SUMMARY

Subsystem	Vehicle weight							
	Mercury		Gemini		Apollo CSM		Apollo LM	
	lb	Percent	lb	Percent	lb	Percent	lb	Percent
Spacecraft dry weight, lb (a)	2993	100	7163	100	19 719	100	8773	100
Power generation and distribution	500	16.7	704	9.8	3082	15.6	1375	15.7
Guidance and navigation	--	--	183	2.6	411	2.1	326	3.7
Stabilization and control	102	3.4	112	1.6	258	1.3	47	.54
Display and controls	95	3.2	98	1.4	405	2.1	271	3.1
Caution and warning	--	--	8	.1	21	.11	18	.21
Sequencing	--	--	121	1.7	168	.85	14	.16
Instrumentation	98	3.3	165	2.3	111	.56	117	1.3
Communication and tracking	167	5.6	216	3.0	426	2.2	265	3.0
Totals	962	32.2	1607	22.5	4882	24.8	2433	27.7

[a]Less launch escape system, booster adapter, propellant and expendables.

were distributed among the individual subsystems. The weights of these two subsystems are small percentages of the spacecraft weights for all spacecraft. The LM sequencing weight in the table is so low because only the explosive-devices sequencing-system weight was separately identifiable. Much of the sequencing function is distributed among the subsystems involved in the sequencing. The weight of the CSM electronic subsystems is three times that of the Gemini spacecraft and twice that carried by the LM. In terms of percentage of spacecraft weight, however, it is not greatly different from the other two spacecraft. Interestingly, the total weight for the instrumentation function is relatively constant for the spacecraft, varying from a minimum of 98 pounds for the Mercury spacecraft to a maximum of 165 pounds for the Gemini spacecraft. The power-generation and -distribution system has the largest portion of spacecraft weight allocated to it on all four spacecraft. Approximately one-half the spacecraft electronics weight is in this subsystem.

AVIONICS IMPLICATIONS FOR THE SPACE SHUTTLE

As indicated in the comparison of mission characteristics, many of the performance requirements of the shuttle are encompassed by the performance requirements that have been satisfied in the previous manned-spacecraft programs. For the space shuttle, a maneuver ΔV that is somewhat more than the ΔV of any previous spacecraft will have to be applied and directed by the orbiter. However, this is a difference in magnitude rather than in kind. The entry and terminal phases of the flightpath of the shuttle may be somewhat different from those of previous programs. If a high-cross-range orbiter is adopted, the entry flightpath will be extended in time, and will have lower vehicle accelerations and higher vehicle temperatures. Although the steering equations to be used with this type of flightpath undoubtedly will be different from those used in the previous semiballistic entries (principally because the magnitude of the lift vector of the vehicle, as well as the direction, will be modulated), the character of the entry requirement will not be altered. The differentiation between a small amount and a large amount of lift does not significantly affect the mechanization of the navigation and guidance system. If the shuttle is a lifting vehicle, the lift must be directed intelligently during entry regardless of the magnitude of the lift. Of course, the landing-accuracy requirement that is associated with landing on a runway is more demanding than the area landing constraints placed on previous manned spacecraft. In general, this requirement will require inputs from the ground. However, because the shuttle must have a guidance and navigation system with relatively good performance to negotiate the entry profile, only minor additions will be required to solve the landing-approach problem. With a position fix shortly after blackout to eliminate the entry dispersions and position fixes during terminal approach to give an end-of-the-runway reference, it should be possible to meet the requirements of the terminal phase through flare-out and touchdown with the onboard guidance, navigation, and control system.

The review of the electronics of four NASA manned spacecraft suggests the following implications for the space shuttle.

Electrical Power System

Studies of the space shuttle to date indicate that the on-orbit power requirements for the orbiter are approximately 6 kilowatts, and total energy requirements are approximately 600 kW-h. With these electrical-power and energy requirements and in view of the fact that a hydrogen-oxygen primary propulsion system and RCS will be used on the orbiter, it appears probable that the primary electrical-power source on the orbiter will be fuel cells, possibly with supplemented silver-zinc batteries. However, the orbiter also has some peak power requirements (approximately 15 kilowatts) during launch and during entry, approach, and landing. The launch peak results from the fact that the orbiter carries the second-stage rocket engines and must provide hydraulic power for the engine servoactuators. The entry, approach, and landing peak results from the hydraulic power required to drive the aerodynamic control surfaces. A number of power sources are being considered to meet these requirements. Hydrogen-oxygen-fueled auxiliary power units driving hydraulic pumps and alternators are the primary system being considered. Auxiliary power units also are being considered as the sole power source on the booster, which has a relatively short flight duration.

The Mercury spacecraft was equipped with a centralized ac power supply, and both the CSM and the LM are similarly equipped. This is the approach currently most-favored for the shuttle because the wiring weight is minimized. The wiring weight is a factor of some concern and attention on the shuttle. For example, the total wiring weight on the CSM is approximately 1300 pounds, which is a significant percentage of the 3082 pounds of electrical-power generation, control, and distribution system weight. The use of a wiring system similar to that of the CSM — one where the power wires are routed through circuit breakers in the cockpit before going to the using equipment — could have a substantial weight penalty on a vehicle such as the shuttle where power-using equipment is widely dispersed in the wingtips and tail. To minimize this potential wiring-weight penalty, solid-state, remotely controlled power distribution and control systems are being considered. In this type of system, minimum-length paths are used for power-distribution buses, and only relatively lightweight signal-control wires are routed to the cockpit for electrical-power control and switching.

Guidance, Navigation, and Control

Although it is not readily obvious from the comparison tables, the guidance and control systems for the Mercury and Gemini spacecraft were structured such that mission success was dependent on a relatively sophisticated simplex system, and crew safety was ensured by relying on simpler backup or abort systems. This is also true of the CSM and LM.

There were two reasons for this approach. First, all four programs have faced relatively tight weight constraints such that it has not been possible to meet mission-success requirements as high as the crew-safety requirements. Secondly, the use of generically different hardware mechanizations has avoided the risk of the same hidden defect in all success paths. Although this approach has worked quite successfully, some limitations became evident particularly with the later spacecraft. The fuel budgets have always been defined by the lowest-performance systems; dispersions have always been higher, thus impacting operations; and the crew-training function has been complicated because the crew has had to learn to fly the spacecraft with multiple systems. Because the avowed goal for the space shuttle is to provide low-operational-cost transportation to orbit and return, economic viability cannot be achieved with an approach in which mission success is sacrificed if a single component in the guidance and control system fails. Multiple guidance and control systems, each having the requisite performance to meet the mission-success goals, must be used. The number of systems to be used will depend on a study of the economic factors involved. In the previous manned spacecraft, at least two, and usually three, flightpath control systems have always been provided. Also, attempts have been made to eliminate all potential single-point failures, though some still remain. It appears likely that the guidance, navigation, and control system of the shuttle, as well as some of the other critical subsystems, will have a similar redundant structure.

A substantial step was taken with the CSM and LM toward an integrated guidance, navigation, and control system with the use of the DAP in the computers. It is probable that this approach also will be used for the shuttle because it provides good performance and flexibility, and is adaptable to the implementation of redundancy. The availability of specific inertial, optical, computational, and radar equipment to meet the shuttle requirements will not be a problem.

Displays and Controls and Caution and Warning

One of the principal space-shuttle requirements is that the orbiter be operated with a crew of two and be flyable, if necessary, by one crewman. For this requirement to be successfully met, it would appear that the increasing proliferation of switches, circuit breakers, displays, and telelights that has characterized the displays and controls evolution from Project Mercury through the Apollo Program will have to be reversed. The weight of displays and controls on the CSM approaches the weight of the guidance and navigation system. Much of this weight is attributable to single-purpose or discrete displays that are used only at single points in the mission. The use of multipurpose displays should minimize the amount of panel space required and should provide a means for presenting more easily assimilated information to the crew. One of the display references that crews of all spacecraft have found natural and desirable in earth orbit is the local-vertical reference, and it should definitely be included in future display systems.

Although it still may be necessary to have discrete circuit-breaker switching for troubleshooting and failure isolation, a much greater use of mode switching and computer-controlled sequencing must be used to relieve the crew of burdensome checklists.

It appears likely that the caution and warning functions will be expanded to include equipment-trend indications so that the crew may be alerted to impending problems as well as to imperative ones. In line with minimizing the proliferation of unique displays and flashing lights, consideration will have to be given to the use of alphanumeric displays and possibly even to the expansion of audible warnings to more than just tones.

Instrumentation and Sequencing

The number of instrumentation points and the amount of data transmitted to the ground have increased substantially from program to program. However, there appears to be little reason to believe that the amount of space-shuttle-subsystem data transmitted to the ground will increase over that of the Apollo Program, because the system requirements are not as demanding as those in the Apollo Program. In fact, with the goal of minimizing operational costs, it is likely that the amount of data transmitted to the ground and the ground complex analyzing the data will be reduced to decrease the operational costs. However, the number of onboard data or instrumentation points will be greater than the inflight measurement points. For example, in addition to the

469 inflight measurement points on the CSM, another 423 instrumentation points were required for the aerospace checkout equipment (ACE) stations that were used before flight. The ACE stations are quite expensive, and the requirement to minimize operating costs will lead to the use of onboard checkout techniques to eliminate or minimize the number and size of ACE stations required. This use of onboard checkout and the requirements for improved caution and warning techniques, coupled with the need to streamline onboard display and switching, leads to the concept of an onboard-data management system that will use a computer to process the data on board. With the availability of the data-management computer, the function of sequencing also becomes a logical candidate for inclusion in the data-management system. The data-management system is covered in much more detail in the paper by Bradford and Chambers.

Ideally, no shuttle-subsystem data would have to be telemetered to the ground from an operational shuttle. Realistically, however, it is a large step from the present manned-space-flight operational concept, where virtually all of the real-time subsystem data monitoring and analysis is done on the ground, to an approach where all monitoring and analysis is done on board. Therefore, it is expected that, although the shuttle will move in the direction of self-sufficiency, the need for some onboard data recording for periodic telemetering will still exist and some real-time subsystem monitoring will be required on the ground. This will certainly be true in the shuttle flight-test program, for which additional developmental flight-test instrumentation (DFI) will be required. How much of the DFI remains installed and becomes part of the operational shuttle is the question that still has to be answered.

Communications and Tracking

Much of the redundant and multiple communications equipment used in the early manned-spacecraft programs can be attributed to the need to ensure communications while exploring the unknown. Some of this concern is also reflected in the complement of communications equipment in the Apollo Program. Moreover, the requirement to have two spacecraft simultaneously communicating with each other and with the ground significantly affected the distribution of equipment used in the CSM and LM communications systems. There should be little concern with the unknown in the space shuttle mission profile; and, once the orbiter separates from the booster, there will be little need for communications between the two craft. In addition, as the shuttle goal of autonomy is approached to minimize the operating costs, it would be expected that the need for telemetry and command receivers would tend to be minimized. Just how small the complement of communications equipment on the shuttle may be will, of course, depend on the system requirements as well as on several other electronic subsystem areas. For example, the shuttle will ultimately be operating in conjunction with the space station and with other space vehicles, so communications-frequency compatibility among them must be considered. Also, the shuttle has a flight phase that previous spacecraft did not and do not have — terminal approach and

and horizontal landing. Some navigational aids will be required to achieve the required three-orders-of-magnitude improvement in touchdown accuracy. Although the improved touchdown capabilities should ameliorate the need for recovery and rescue beacons, it is expected that some space-tracking beacons will continue to be required.

Rendezvous has been accomplished with and without the use of radar in the previous programs. One of the requirements for the shuttle is to rendezvous with failed satellites, which represent passive targets. For this requirement, a skin tracking rendezvous radar would probably be required. However, for the cooperative type of rendezvous that the shuttle will normally be flying, there is some merit in the possible use of a rendezvous technique similar to the one used by the CSM. The IMU alinement optical device also could be used for tracking either the visible reflected sunlight from the target or, if necessary, a flashing beacon. Thus, it would not be necessary to carry the radar weight on every flight. On the flights to failed satellites, the radar could be included as part of the shuttle payload for that mission.

The four spacecraft under consideration have or had eight to 10 antennas each to service the various communications and tracking systems. Hopefully, a communications system for the shuttle that minimized and unified the equipment would also minimize the need for antennas, because antennas tend to be incompatible with entry-vehicle heat-shield requirements.

CONCLUDING REMARKS

The space-shuttle mission characteristics present little or no increase in avionic system requirements over the requirements that have already been met by the electronic systems in the past and present manned-spacecraft programs. Therefore, the demands that the shuttle places on the avionics will not be for increased performance but for increased efficiency and economy of operation. The electrical-power generation technique developed in past and present programs will form the basis for the space-shuttle electrical-power sources. However, more advanced electrical-power distribution and control techniques will be required to minimize power-distribution-system weight penalties. Integrated guidance, navigation, and control system techniques, such as are used on the lunar module and command and service module, will also be used on the space shuttle because they lend themselves to good performance, flexibility, and the implementation of redundancy. Display and control systems must be simplified and made more functional if the shuttle is to be flown by two crewmen with little help from the ground. The functions of instrumentation, sequencing, caution and warning sensing, and checkout will have to be combined into a computer-controlled data-management-system approach if the space shuttle is to be operated completely independently of operational ground stations. However, the step from the present Apollo approach, which maximizes the use of ground support, to the goal of no ground support at all is a big step and may not be completely achievable with the initial shuttle development. Finally, it should be possible to simplify and minimize some of the communications-equipment redundancy and multiple spacecraft antennas used in the past because the operating regime of the shuttle will contain few unknowns.

ACKNOWLEDGMENTS

The authors would like to gratefully acknowledge the efforts of Robert Swint and Gene Holloway, who helped in organizing the data for this paper, and the contributions of Louis Leopold, Earl Tiedt, Kent Castle, and Gary Johnson, who supported the data-collection process.