4. GUIDANCE, CONTROL, AND PROPULSION SYSTEMS

By RICHARD R. CARLEY, *Gemini Program Office, NASA Manned Spacecraft Center;* NORMAN SCHULZE, *Propulsion and Power Division, NASA Manned Spacecraft Center;* BENJAMIN R. DRONE, *Gemini Program Office, NASA Manned Spacecraft Center;* DAVID W. CAMP, *Gemini Program Office, NASA Manned Spacecraft Center; and* JOHN F. HANAWAY, *Guidance and Control Division, NASA Manned Spacecraft Center*

Summary

In accomplishing the Gemini Program objectives, an onboard digital computer system, an inertial platform reference system, a radar system, and control systems using hypergolic bipropellant propulsion have been developed and successfully demonstrated.

Introduction

The program objectives of long-duration, rendezvous, and controlled-reentry missions have placed special requirements on the spacecraft guidance and control systems. These objectives required maximum reliability and flexibility in the equipment. This was accomplished by utilization of simple design concepts, and by careful selection and multiple application of the subsystems to be developed.

Guidance and Control System Features

In the development of an operational rendezvous capability, the geographical constraints on the mission are minimized by providing the capability for onboard control of the terminal rendezvous phase. To complete the rendezvous objectives, the spacecraft must be capable of maneuvering, with respect to the target, so that the target can be approached and a docking or mating operation can be accomplished.

For failures in the launch vehicle, such as engine hardover and launch vehicle overrates, where effects are too fast for manual reaction, the automatic portion of the launch-vehicle malfunction-detection system switches control from the primary to the secondary system. The secondary system receives command signals from the spacecraft system for launch guidance.

To develop an operational guided reentry, onboard control has been provided. The use of the flight crew for control mode selection and command of attitudes, as well as for detection of malfunctions and selection of redundant systems, simplifies the system design and reduces the need for complicated protective interlocks.

Guidance, Control, and Propulsion Systems Implementation

The features just discussed dictated the configuration of the Gemini guidance, control, and propulsion equipment. Figure 4–1 is a block diagram of the systems.

The guidance system consists of: (1) a digital computer and an inertial measuring unit operating together to provide an inertial guidance system, and (2) a radar system which provides range, range rate, and line-of-sight angles to the computer and to the crew-station displays. The ground stations and the spacecraft are equipped with a digital command system to relay information to the spacecraft digital computer.

The control system consists of: (1) redundant horizon-sensor systems, (2) an attitude controller, (3) two translation-maneuver hand controllers, and (4) the attitude-control and maneuvering electronics which provide commands to the reentry-control and to the orbit-attitude and maneuvering portions of the propulsion system. The retrorocket propulsion engines are normally fired by a signal from the spacecraft time-reference system.

Figure 4–2 shows the arrangement of the guidance, control, and propulsion equipment in the spacecraft. The locations are shown for the thrust chamber assemblies, or engines, for the reentry control system, and for the orbital attitude and maneuver system. The attitude controller is located between the two crewmembers,

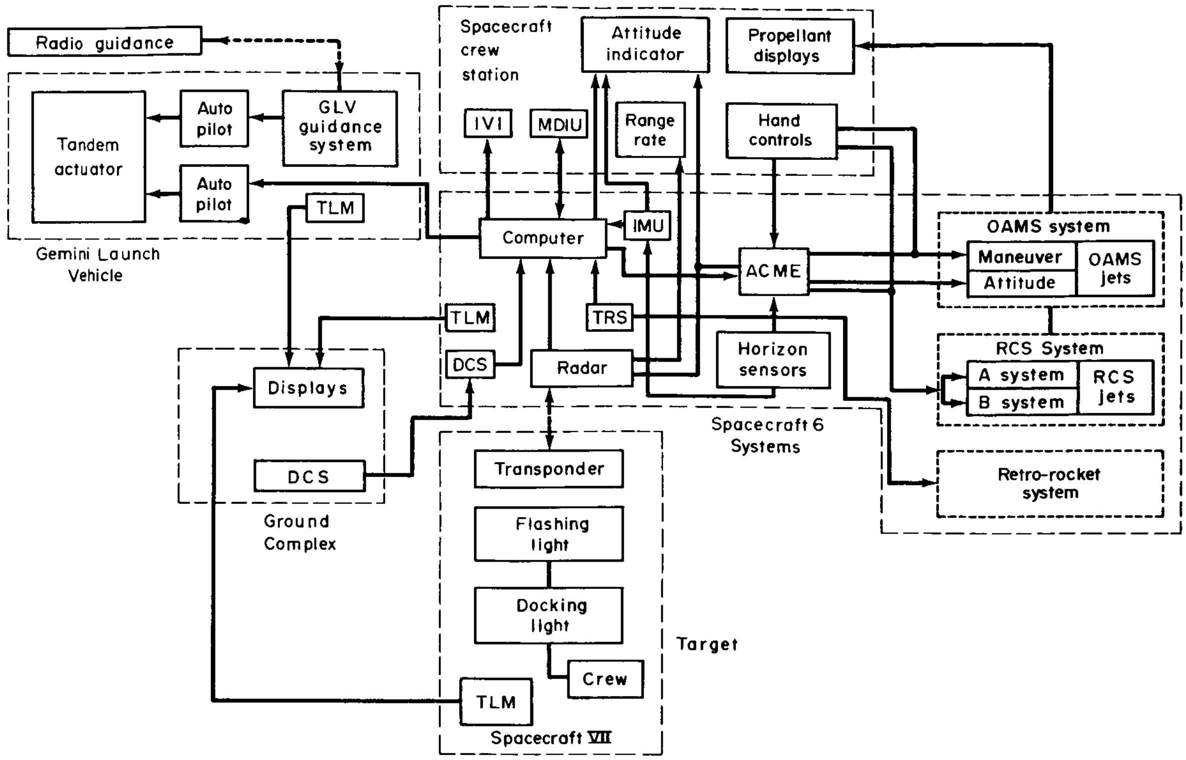


FIGURE 4-1.—Spacecraft guidance and control system.

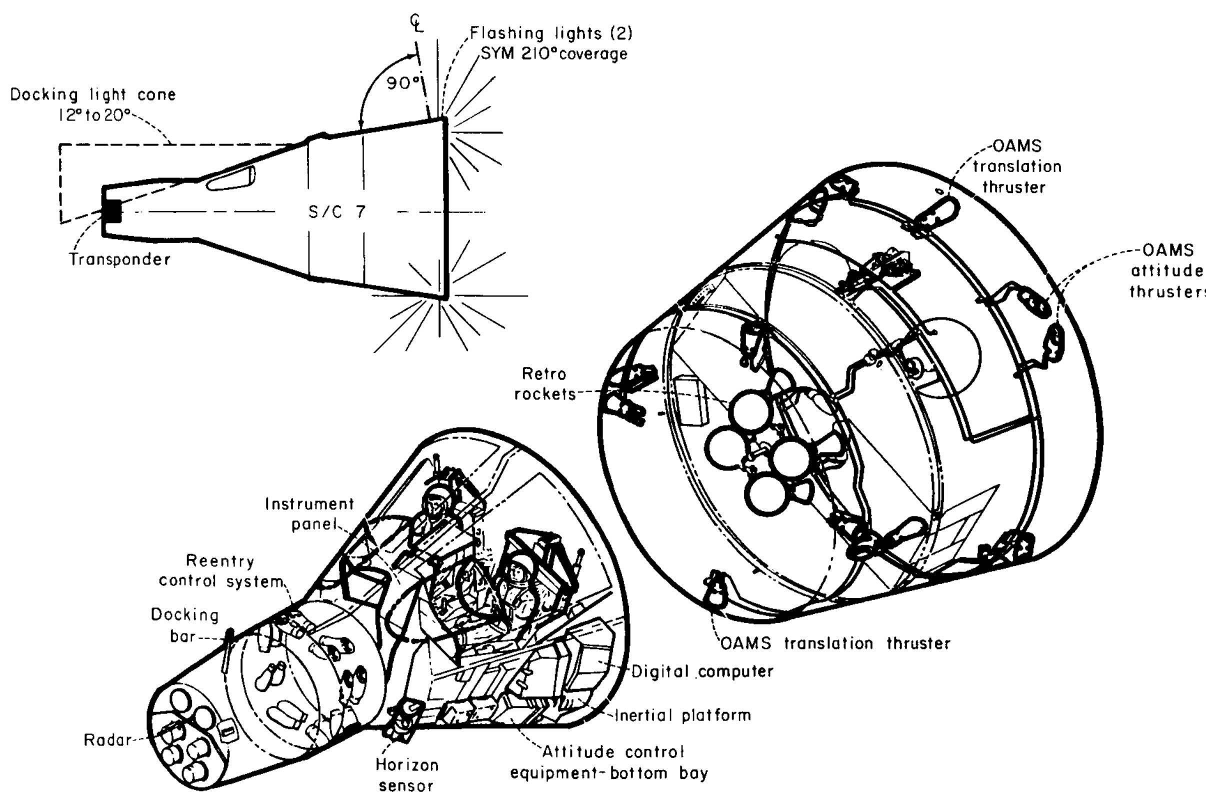


FIGURE 4-2.—Arrangement of guidance and control system components in the spacecraft.

and a translation controller is located on each side of the cabin.

Two attitude display groups, located on the instrument panel, use an eight-ball display for attitude orientation, and are equipped with three linear meter needles called flight director indicators. During launch or reentry, these needles can be used to indicate steering errors or commands and permit the flight crew to monitor the primary system performance. The needles can also be used to display attitude errors and to provide spacecraft attitude-orientation commands. The radar range and range-rate indicator used for the rendezvous missions is located on the left panel.

Gemini Guidance System

The inertial guidance system provides backup guidance to the launch vehicle during ascent. This system also determines the spacecraft orbit insertion conditions which are used in computing the velocity increment required for achieving the targeted orbit apogee and perigee. This computation is performed using the insertion velocity adjust routine.

A low-gain antenna, interferometric, pulsed radar utilizing a transponder on the target vehicle was selected to generate the information used by the computer to calculate the two impulse maneuvers required to achieve a rendezvous with the target.

The need to reference acceleration measurements and radar line-of-sight angles, as well as to provide unrestricted attitude reference to the crew, resulted in the selection of a four-gimbal stabilized platform containing three orthogonally mounted accelerometers. It provides an inertial reference for launch and reentry, and a local vertical earth-oriented reference for orbit attitude, using orbit-rate torquing.

The inertial guidance system also generates commands which, together with a cross-range and down-range steering display, are used to reach a landing point from dispersed initial conditions. Either an automatic mode, using the displays for monitoring, or a man-in-the-loop reentry-guidance technique can be flown.

The digital computer utilizes a random-access core memory with read-write, stored program, and nondestruct features. This memory has a capacity of 4096 39-bit words. The computer system provides the data processing necessary for launch guidance, rendezvous, reentry, and other calculations.

Control System

The control system (fig. 4–3) is basically a redundant rate-command system with the flight crew establishing an attitude reference and closing the loop. Direct electrical commands to the thrusters and a single-pulse-generation capability are also provided. The control system can be referenced to either of the two horizon-sensor systems to provide a redundant, low-power, pilot-relief mode. This mode controls the vehicle to the local vertical in pitch and in roll. Either horizon sensor can also supply the reference for alining the platform in a gyrocompassing-type automatic or manual mode as selected by the crew. To achieve the desired degree of reliability, the spacecraft is equipped with two separate reentry-control systems which include propellants, engines, and electrical-control capability. Either reentry-control system is adequate for controlling spacecraft attitude during the retrofire and reentry phases of the mission.

The control system was designed to operate with on-off rather than proportional commands to the propulsion engine solenoids. This simplified operation reduced the design requirements on the system electronics, solenoids, and valves, and on the dimensions and injector design of the thrust chamber assemblies, and also allowed the use of simple switch actuation for direct manual control. The engine thrust levels selected were those which would provide translation and rotational acceleration capability adequate for the completion of all tasks even with any one engine failed, and which would allow reasonable limit-cycle propellant-consumption rates for a long-period orbit operation.

Propulsion System

The orbital attitude and maneuver system (fig. 4–4) uses a hypergolic propellant combination of monomethylhydrazine and nitrogen tetroxide which is supplied to the engines by a regulated pressurization system that uses helium gas stored at 2800 psi. The choice of these propellants, along with the on-off mode of operation, minimized ignition requirements and permitted simplification of engine design. Controlled heating units prevent freezing of the

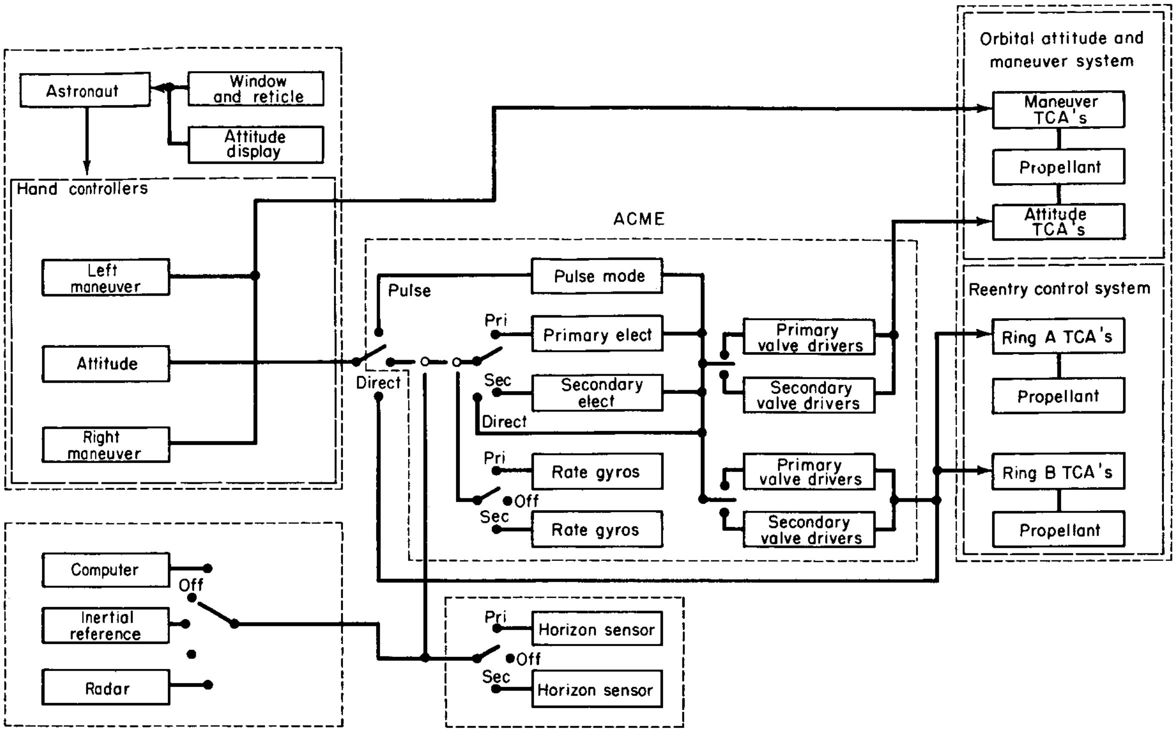
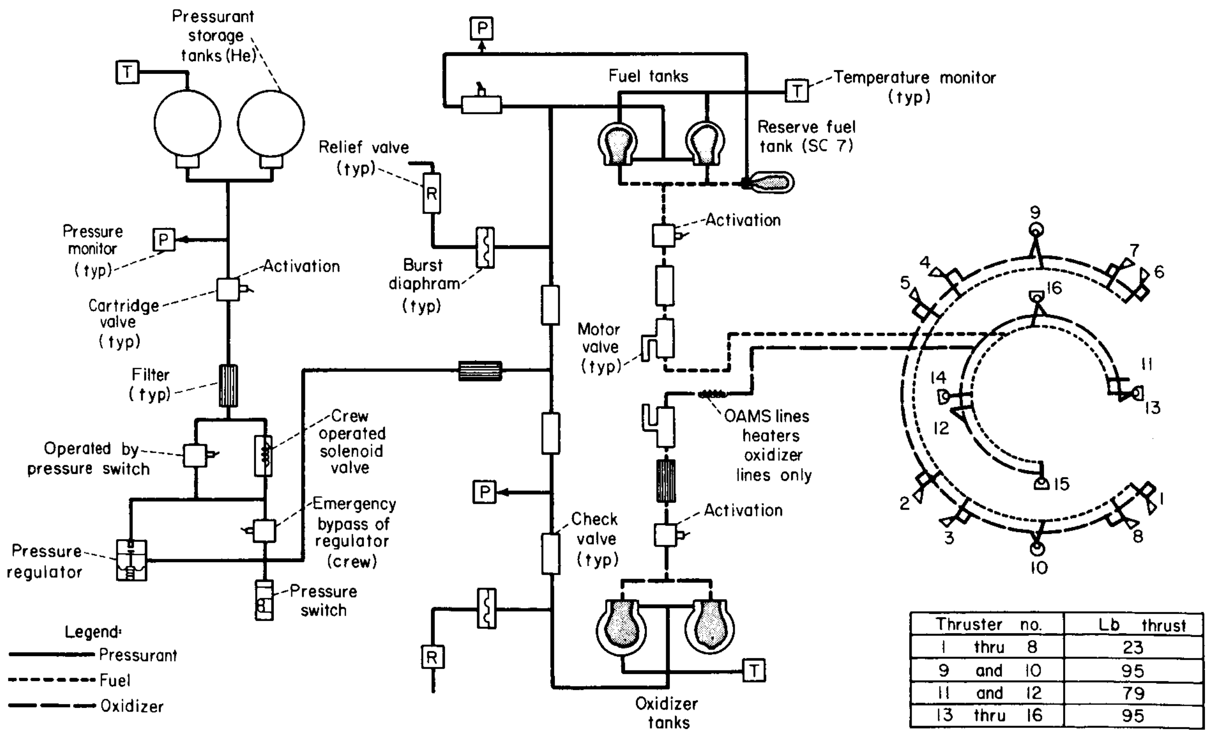


FIGURE 4-3.—Control system.



Thruster no.	Lb thrust
1 thru 8	23
9 and 10	95
11 and 12	79
13 thru 16	95

FIGURE 4-4.—Orbital attitude and maneuver system.

propellants. A brazed, stainless-steel plumbing system is used so that potential leakage points and contamination are eliminated. Positive expulsion bladders are installed in the propellant tanks. Table 4–I shows the system characteristics for steady-state engine operation.

The reentry-control system is of similar design to the orbital attitude and maneuver system. Ablative-type engines to limit reentry heating problems are used on the reentry vehicle. To reduce hardware development requirements and to permit a clean aerodynamic configuration, submerged engines, similar in design concept, are used in the orbital attitude and maneuver system.

The separate retrograde propulsion system consists of four spherical-case, polysulfide-ammonium-perchlorate, solid-propellant motors. The system is designed to assure safe reentry after any three of the four motors have been fired. The design also allows the system to be used for emergency separation of the spacecraft from the launch vehicle after lift-off.

Development Program

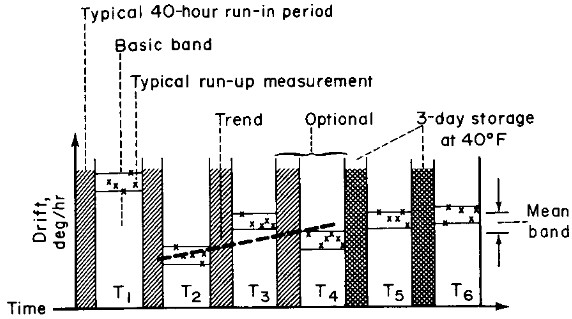
During the development phase, each guidance and control component underwent a comprehensive series of ground tests, both individually and after integration with interfacing components. These included engineering tests beyond the qualification level; qualification tests; and overstress, reliability, and complete systems tests at the vendor's plant. The computer and inertial-measurement-unit systems, engineering models as well as flight hardware, were integrated at the computer manufacturer's plant. Flight units were delivered to the prime contractor with the flight computer program loaded, for installation in the spacecraft prior to spacecraft systems tests. During the development of the guidance and control hardware, it was established that temperature and random vibration environments were needed as part of the predelivery acceptance tests on each flight unit to verify system capability and to establish and maintain effective quality control. A two-sigma flight environment was used to uncover conditions not apparent in the normal testing environment. Unsatisfactory conditions were corrected, and the units retested until proper operation was obtained as a means for insuring high reliability of the flight equipment.

For the Gemini guidance and control program, many special tests were developed. As an example, a special inertial component run-in test procedure (fig. 4–5) was used to determine gyro normal-trend data and also to reject unstable gyros before installation in platforms. After a 40-hour run-in period, five runup-to-runup drift measurements are obtained, followed by subsequent sets of run-in and runup-to-runup measurements. The units are rejected as having unstable characteristics if the drift trend is excessive, or if the effect of the run-in and the storage-temperature-soak on the performance of the gyro creates an unusual spread within the sets of measurement bands or the amount of shift of the bands. Tests of this nature assure adequate selection of inertial components and, along with 100 percent inspection of parts and similar techniques, have significantly improved system reliability.

TABLE 4–I.—*Gemini Propulsion System Characteristics*

Propulsion system	Number of engines	Thrust, lb_f (a)	Total impulse, lb_f-sec	Propellant weight, lb_m (b)	Specific impulse, lb_f-sec/lb_m
Orbital attitude and maneuver system	8	23	180 000	710	258
	2	79			273
	6	95			273
Reentry control system	16	23	18 500	72	283
Retrorockets	4	2490	56 800	220	255

a lb_f = pounds of force.
b lb_m = pounds of mass.



Trend-Least square fit of means of last three of $T_1T_2T_3T_4$

Mean band-Maximum spread of means of last four of $T_6T_5T_4T_3T_2$

Average basic band-Algebraic average of basic band values from last four of T $T_5T_4T_3T_2$

FIGURE 4-5.—Gyro test procedure.

Onboard Computer Program Development

An extensive development program for the computer-stored program was established to assure timely delivery, adequate verification, and good reflection of mission requirements. Figure 4-6 shows the basic organizational arrangement that was established. A critical feature is the monthly issue of the detailed system description authorized and provided to all users to assure common understanding, and integrated and coordinated implementation of supporting requirements. The programs are subjected to rigorous tests, including a mission verification simulation program. These tests provide dynamic simulation of the flight computer, which has been loaded with the operational program; all interfaces are exercised and all computer logic and mode operation thoroughly demonstrated. Figure 4-7 indicates a few of the detailed steps and iterations required in the development of a successful computer program. Figure 4-8 shows the computer-program development schedule, and also indicates the required lead time and development background.

Propulsion System Preflight Background

A similar, extensive ground-test program was conducted on the propulsion systems during research, development, qualification, reliability, and complete systems-test programs. A full-scale retrorocket abort test was conducted in an altitude chamber which determined the required nozzle-assembly design.

An analysis of the reentry control system and

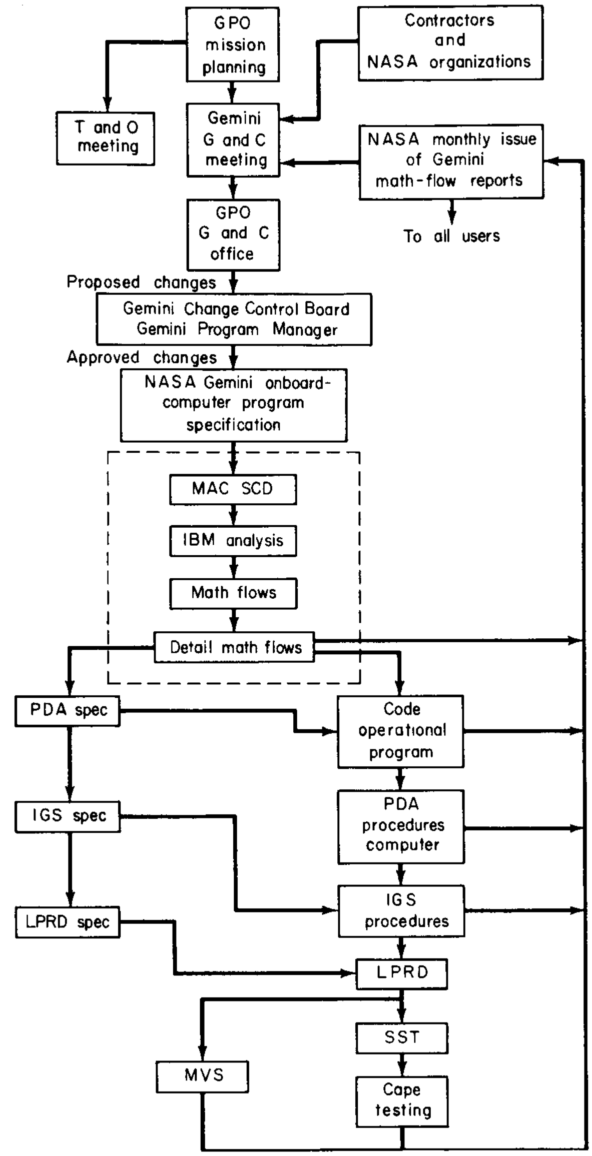


FIGURE 4-6.—Math flow control procedures and required intermediate goals.

the orbital attitude and maneuver system engine operation reveals that engine life is a function of the firing history (fig. 4-9). A long engine life results from low-percent duty cycles which, however, decrease specific impulse. To meet the duty-cycle requirements of the Gemini spacecraft, the mixture ratio of the propellants was decreased so that the combustion gas temperatures would be reduced. Major design changes also were instituted to provide greater engine integrity by permitting fuel-film-cooled walls and reorientation of the thrust-chamber-

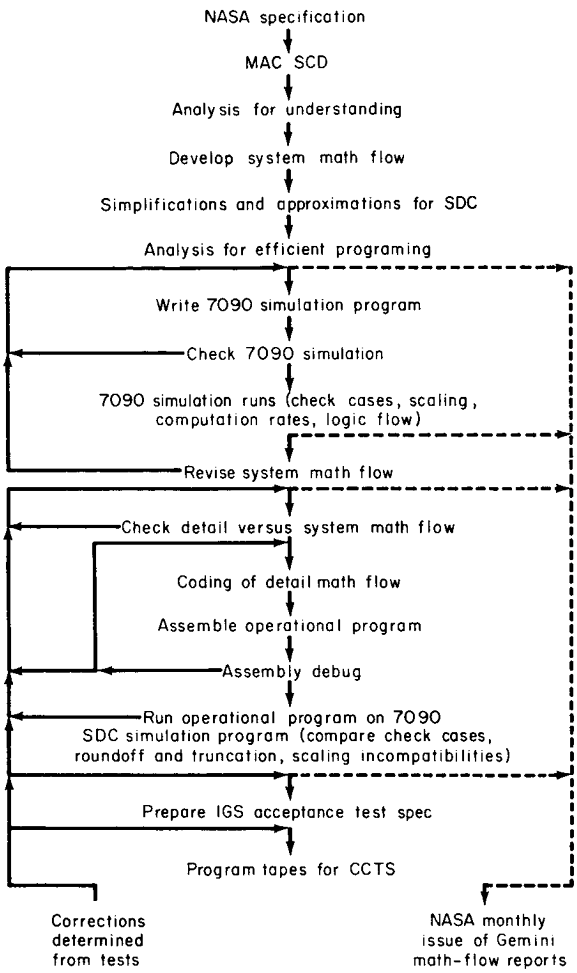


FIGURE 4-7.—Required intermediate goals in math flow development.

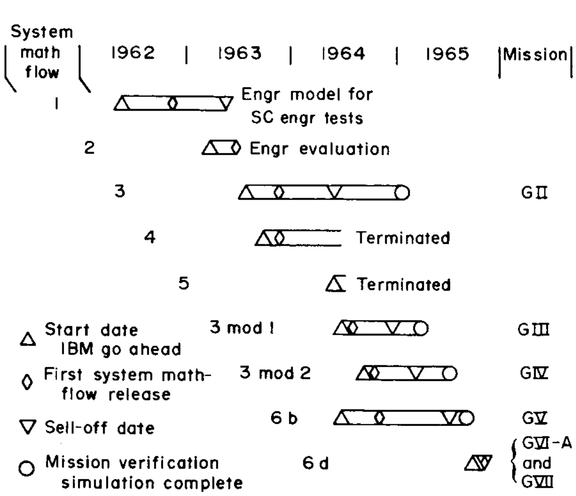


FIGURE 4-8.—Computer program development status chart.

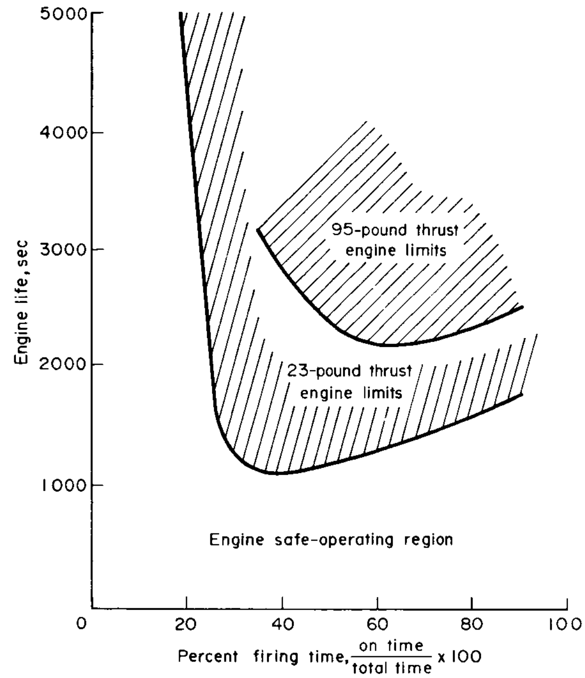


FIGURE 4-9.—Engine firing capability.

assembly ablative layers. Special hot-fire tests of the injector assemblies provided a basis for rejection of undesirable injectors prior to engine assembly.

Flight Performance

Guidance System Performance

The accumulated hours that the guidance and control system was in operation during the various missions are shown in table 4-II. Of all the missions, Gemini V required the maximum number of operating hours on the following systems and components:

(1) Platform—32 hours
(2) Attitude control and maneuver electronics—142 hours
(3) Primary horizon sensor—38 hours
(4) Secondary horizon sensor—45 hours

The maximum operating time required for the computer was 20 hours during the Gemini VI-A mission.

Beginning with the Gemini IV mission, the systems were subjected to repeated power-up and power-down cycling. After a periodic update of the emergency-reentry quantities for the Gemini IV computer, the flight crew was

TABLE 4–II.—*Gemini Component Operating Hours*

Component	Gemini II	Gemini III	Gemini IV	Gemini V	Gemini VI–A	Gemini VII	Total
Computer	0.2	4.7	6.3	16.0	20.0	6	53.2
Inertial measurement unit (platform)	.2	4.7	9.7	32.7	20.0	14	81.3
Attitude control and maneuver electronics	.2	4.7	37.0	142.0	25.7	91.5	301.1
Horizon scanner (primary)	.2	2.2	33.0	38.4	25.4	16.0	115.2
Horizon scanner (secondary)	.2	2.5	.1	45.0	.3	0	48.1

unable to power-down the computer system using normal procedures. Power was removed using an abnormal sequence which altered the computer memory and, therefore, prevented its subsequent use on the mission. Subsequent inflight cycling of the switch reestablished normal power operation. During postflight testing of the computer, 3000 normal cycles were demonstrated, both at the system level and with the system installed in the spacecraft. This testing was followed by a component disassembly program which revealed no anomalies within the computer, auxiliary computer power unit, or the static power supply.

The primary horizon sensor on the Gemini V spacecraft failed at the end of the second day of the mission. The mission was continued using the secondary system. The horizon-sensor head is jettisoned prior to reentry, which makes postflight analysis difficult; however, the remaining electronics which were recovered operated normally in postflight testing.

During ascent, the steering-error monitoring, along with selected navigation parameters which are available as onboard computer readouts, has given adequate information for onboard switchover and insertion go—no-go decisions. Table 4–III contains a comparison of the nominal preflight targeted apogee and perigee altitudes, with the flight values actually achieved. The table also shows, in the IVAR column, the values which would have resulted from the use of the insertion velocity adjust routine (IVAR) after insertion with the primary guidance system, and, in the IGS column, the values which would have been achieved had switchover to inertial-guidance-system (IGS) steering occurred early in stage II flight and assuming that no insertion correction had been made. A range of apogees from 130 to 191 nautical miles was targeted on the flights. Comparison of the actual values with those in the IVAR column shows that, after the Gemini III mission, the insertion velocity adjust routine would have reduced the dispersion of the actual from nominal. The IGS column shows that, had the backup system been selected, it would have given insertion conditions resulting in a safe orbit and a go-decision for all flights. Although the primary guidance was adequate on all flights, the inertial guidance system, subsequent to the Gemini III mission, would have provided guidance values closer to nominal than the primary system. The use of the insertion velocity adjust routine would have further reduced these dispersions.

Table 4–IV compares the nominal, actual, and inertial-guidance-system insertion values of total velocity and flight path angle. The actual value was computed postflight from a trajectory which included weighted consideration of all available data. The comparison indicates that, for missions after the Gemini III mission, the interial-guidance-system performance has been well within expectations.

During the orbital phases of flight, the inertial guidance system was utilized for attitude control and reference, for precise translation control, and for navigation and guidance in closed-loop rendezvous. Performance in all of these functions is dependent upon platform alinement. The alinement technique has proved to be satisfactory, with the residual errors, caused by equipment, in all axes being on the order of 0.5° or less.

TABLE 4-III.—*Comparison of Orbital Parameters at Insertion* [a]

Mission	Absolute value, nautical miles							
	Nominal		Actual		IVAR [b]		IGS [c]	
	Apogee	Perigee	Apogee	Perigee	Apogee	Perigee	Apogee	Perigee
Gemini II [d]	141	90	N/A	N/A	111 (−30)	87 (−3)	N/A	N/A
Gemini III	130.1	87.1	121.0 (−9.1)	87.0 (−0.1)	121 (−9.1)	90 (2.9)	128 (−2.1)	78 (9.1)
Gemini IV	161.0	87.0	152.2 (−8.8)	87.6 (0.6)	164.3 (3.3)	87.0 (0)	163.9 (2.9)	87.0 (0)
Gemini V	191.2	87.0	188.9 (−2.3)	87.4 (0.4)	189.9 (−1.3)	87.0 (0)	192.7 (1.5)	86.9 (−0.1)
Gemini VI-A	146.2	87.1	140.0 (−6.2)	87.0 (−0.1)	146.5 (0.3)	87.0 (−0.1)	140.5 (−5.7)	87.0 (−0.1)
Gemini VII	183.1	87.1	177.1 (−6.0)	87.1 (0)	181.0 (−2.1)	87.0 (−0.1)	180.0 (−3.1)	87.0 (−0.1)

[a] Values in parentheses are differences from nominal.
[b] Insertion velocity adjust routine.
[c] Inertial guidance system.
[d] Values shown from Gemini II are those targeted to exercise the IVAR routine.

TABLE 4-IV.—*Comparison of Insertion Conditions*

Mission	Insertion condition	Nominal (targeted)	Actual	Inertial guidance system
Gemini II	Total velocity, fps	25 731	25 736	25 798
	Flight path angle, deg	−2.28	−2.23	−2.20
	Time from lift-off, sec	356.5	352.2	351.8
Gemini III	Total velocity, fps	25 697	25 682	25 697
	Flight path angle, deg	+0.01	+0.01	+0.32
	Time from lift-off, sec	358.4	353.8	353.7
Gemini IV	Total velocity, fps	25 757	25 746	25 738
	Flight path angle, deg	+0.00	+0.04	+0.06
	Time from lift-off, sec	355.8	353.8	353.8
Gemini V	Total velocity, fps	25 812	25 805	25 808
	Flight path angle, deg	+0.02	0.00	−0.01
	Time from lift-off, sec	356.9	353.2	353.2
Gemini VI-A	Total velocity, fps	25 730	25 718	25 720
	Flight path angle, deg	0.00	+0.03	+0.03
	Time from lift-off, sec	356.7	358.7	358.7
Gemini VII	Total velocity, fps	25 806	25 793	25 801
	Flight path angle, deg	0.00	0.03	0.03
	Time from lift-off, sec	358.6	357.0	357.0

Figure 4–10 contains a time history of the radar digital range and computed range rates during the rendezvous approach for the Gemini VI–A mission. Rendezvous-approach criteria limit the permissible range rate as a function of range for the closing maneuver. The figure shows that, prior to the initial braking maneuver, the range was closing linearly at approximately 40 feet per second. If the effect of the braking thrust is ignored, an extrapolation of range and range rate to the nominal time of interception indicates that a miss of less than 300 feet would have occurred. A no-braking miss of this order is well within the requirements for an easy manual approach and docking with the target vehicle. Solid lock-on was achieved at 232 nautical miles and was maintained until the spacecraft had closed with the target and the radar was powered down.

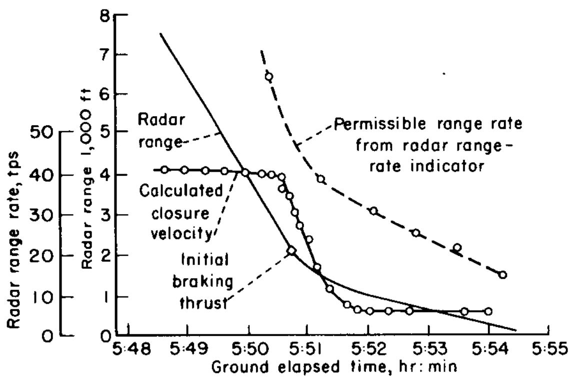


FIGURE 4–10.—Radar trajectory range comparison for Gemini VI–A and VII rendezvous.

The rendezvous performed on the Gemini VI–A/VII missions was nominal throughout. A computer simulation has been completed in which actual radar measurements were used to drive the onboard computer program. A representative value of the computed total velocity to rendezvous is compared with the telemetered values and shown in table 4–V. The close agreement verifies onboard computer operation. A trajectory simulation has verified total system operation. Using the state vectors obtained from the available tracking of the Gemini VI–A and VII spacecraft prior to the terminal phase, and assuming no radar, platform, alinement, or thrusting errors, the values of the total velocity to rendezvous and the two vernier midcourse corrections were computed. The simulated values and the actual values agree within the uncertainties of the spacecraft ground tracking for the conditions stated. The flyby miss distance resulting from this simulation was 96.6 feet.

The Gemini VI–A and VII spacecraft both demonstrated successful onboard-controlled re-

TABLE 4–V.—*Rendezvous Velocity Comparisons*

[Angle to rendezvous equals 130°]

Computer simulation			
Time from lift-off	Radar, nautical miles	Simulated ΔV_t,[a] feet per second	Data acquisition ΔV_t,[a] feet per second
5:15:20	36.20	70	69

Trajectory simulation			
First midcourse correction, incremental velocity indicators		Second midcourse correction, incremental velocity indicator	
Simulated, feet per second	Actual, feet per second	Simulated, feet per second	Actual, feet per second
3 aft 0 right/left 3 down	7 forward 5 left 7 up	2 aft 0 right/left 1 down	4 forward 6 right 2 up

[a] ΔV_t=total velocity to rendezvous.

entries. The cross-range and down-range error indications of the flight director indicator permitted both flight crews to control the spacecraft landing point to well within the expected tolerance of 12 nautical miles.

Table 4–VI is a summary of reentry navigation and guidance performance. The first line on the figure shows the inertial-guidance-system navigation error after the completion of steering at 80 000 feet and is obtained from comparisons with the best estimate trajectory. These values show that the system was navigating accurately. The next line shows the miss distances as a difference between the planned and actual landing points. The Gemini II mission had an unguided reentry from a low-altitude-insertive reentry condition which tended to reduce dispersions. Gemini III was planned and flown so that a fixed-bank angle, based on the postretrofire tracking as commanded from the ground, was held until the cross-range error was brought to zero. During this flight, however, the aerodynamic characteristics and the velocity of the retrograde maneuver performed with the orbital attitude and maneuver system differed from those expected. This difference reduced the spacecraft lifting capability to such an extent that, with the open-loop procedure flown, the targeted landing area could not be reached using the planned technique. The onboard computer predicted this condition and gave the correct commands to permit the flight crew to achieve the correct landing point. The Gemini IV reentry dispersion is that resulting from reentry from a circular orbit and being flown without guidance. The Gemini V reentry miss was caused by an incorrect quantity being sent from the ground. This quantity was used to initialize the inertial guidance system prior to reentry, and the incorrect quantity caused the inertial guidance system to show the incorrect range to the targeted landing area. The flight crew determined that a discrepancy existed in the system and, at that time, started flying a constant bank-angle reentry. The last two lines in table 4–VI indicate some of the factors causing shifts in the landing-area footprints for the Gemini missions. This table indicates generally good system performance.

Control and Propulsion System Performance

The control system has been thoroughly exercised, and all design objectives have been demonstrated. The platform mode has proved well suited for in-plane translations, for platform alinement, and for general pilot relief in busy exercises such as station keeping. The rate-command capability has been most useful for

TABLE 4–VI.—*Gemini Reentry Navigation Summary*

Flight	Gemini II	Gemini III	Gemini IV	Gemini V	Gemini VI–A	Gemini VII
	Trajectory difference, nautical miles					
Inertial guidance system—best estimate trajectory difference at 80 000 feet	1.2	0.8	([a])	[b] 1.1	[c] 2.5	2.3
Planned—best estimate trajectory difference at touchdown	18	64	47	97	[c]<7	6.6
Footprint shift, nautical miles						
Retrofire	14	48	50 [d]	5	22	41
Aerodynamics	([a])	160	([a])	([a])	([a])	40

[a] Not determined.
[b] With corrected value for ground update.
[c] Based on extrapolated radar data.
[d] Preretrofire and retrofire.

translations, such as retrofire and rendezvous maneuvers, and for damping aerodynamic oscillations during reentry in order to ease the reentry guidance task. Pulse mode has provided the fine control necessary for manual platform alinements, for station keeping, and for experiments and maneuvers requiring accurate pointing. Reentry rate command has been used on the Gemini II and IV missions for reentry control. The wide deadbands mechanized in this mode conserve propellants while retaining adequate control.

The horizon mode has been utilized extensively to provide pilot relief through automatic control of pitch and roll attitude based upon horizon-sensor outputs. Performance, in general, has been excellent, although several instances of susceptibility to sun interference have been noted. On the Gemini VI–A mission, this mode operated unattended for approximately 5 hours while the flight crew slept. The final or direct mode has been utilized effectively by the crew when they wished to perform a maneuver manually with the maximum possible control authority.

Typical retrofire maneuver performance is shown in table 4–VII. During the first manned mission, the Gemini III spacecraft retrofire maneuver was performed with the roll channel in direct mode and with the pitch and yaw channels in rate command. This method of operation provided additional yaw authority in anticipation of possible high-disturbance torques. Only nominal torques were experienced, however, and the remaining missions utilized rate-command mode in all axes. Attitude changes during retrofire have resulted in velocity errors well within the lifting capability of the spacecraft and would not have contributed to landing-point dispersions for a closed-loop reentry. A night retrofire was demonstrated during the Gemini VI–A and VII missions. In summary, the performance of the attitude-control and maneuvering electronics has been exceptional during ground tests as well as during all spacecraft flights.

The Gemini III spacecraft demonstrated the capability to provide orbital changes which included a retrograde maneuver that required a 111-second firing of the aft engines in the orbital attitude and maneuver system. The

TABLE 4–VII.—*Typical Gemini Retrofire Maneuver Velocity Comparison*

[Values in parentheses are differences from nominal]

Flight	ΔX, feet per second	ΔY, feet per second	ΔZ, feet per second	Total
Gemini VI–A	−308	0	117	329. 5
	(1)	(−1)	(−1)	(. 6)
Gemini VII	−296	0	113	316. 8
	(2)	(3)	(−1)	(1. 6)

propulsion system maneuvering capability was used for the rendezvous maneuvers during the Gemini VI–A mission.

There have been two flights with known anomalies which could definitely be attributed to the propulsion systems. The two yaw-left engines in the orbital attitude and maneuver system of the Gemini V spacecraft became inoperative by the 76th revolution, and neither engine recovered. Rate data also showed that other engines exhibited anomalous behavior but subsequently recovered, and this suggested the cause to be freezing of the oxidizer. During this flight the heater circuits had been cycled to conserve power. During the Gemini VII mission, the two yaw-right engines in the orbital attitude and maneuver system were reported inoperative by the crew approximately 283 hours after lift-off. Postflight analysis of rate data verified this condition. However, because these engines are not recovered, failure analysis is difficult, and inflight testing was insufficient to identify the cause of the failure on Gemini V and VII. Further studies are being conducted in an attempt to isolate the cause.

On the Gemini IV spacecraft, one of the pitch engines in the reentry control system was inoperative; however, postflight examination revealed a faulty electrical connector at the mating of the reentry-control-system section and the cabin section.

The propellant quantity remaining in the spacecraft during the flight is determined by calculating the expanded volume of the pressurizing gas using pressure and temperature measurements. Flight experience has shown that, due to inaccuracies in this quantity-gaging system, a significant quantity of propellants must be reserved for contingencies. A reserve propellant tank has been added to assure that a known quantity of propellant remains even though the main tanks have been depleted, thus insuring the capability of extending the mission to permit recovery in the planned primary landing area.

Conclusions

As a result of developing onboard capability, greater flexibility in mission planning and greater assurance of mission success have been achieved. In addition, information obtained from systems such as the inertial guidance system and the radar system has significantly improved the knowledge of the launch, orbital, and reentry phases of the mission and has made a thorough analysis more practical.

For the guidance, control, and propulsion systems, the design, development, implementation, and operating procedures have been accomplished, and the operational capabilities to meet the mission requirements have been successfully demonstrated.