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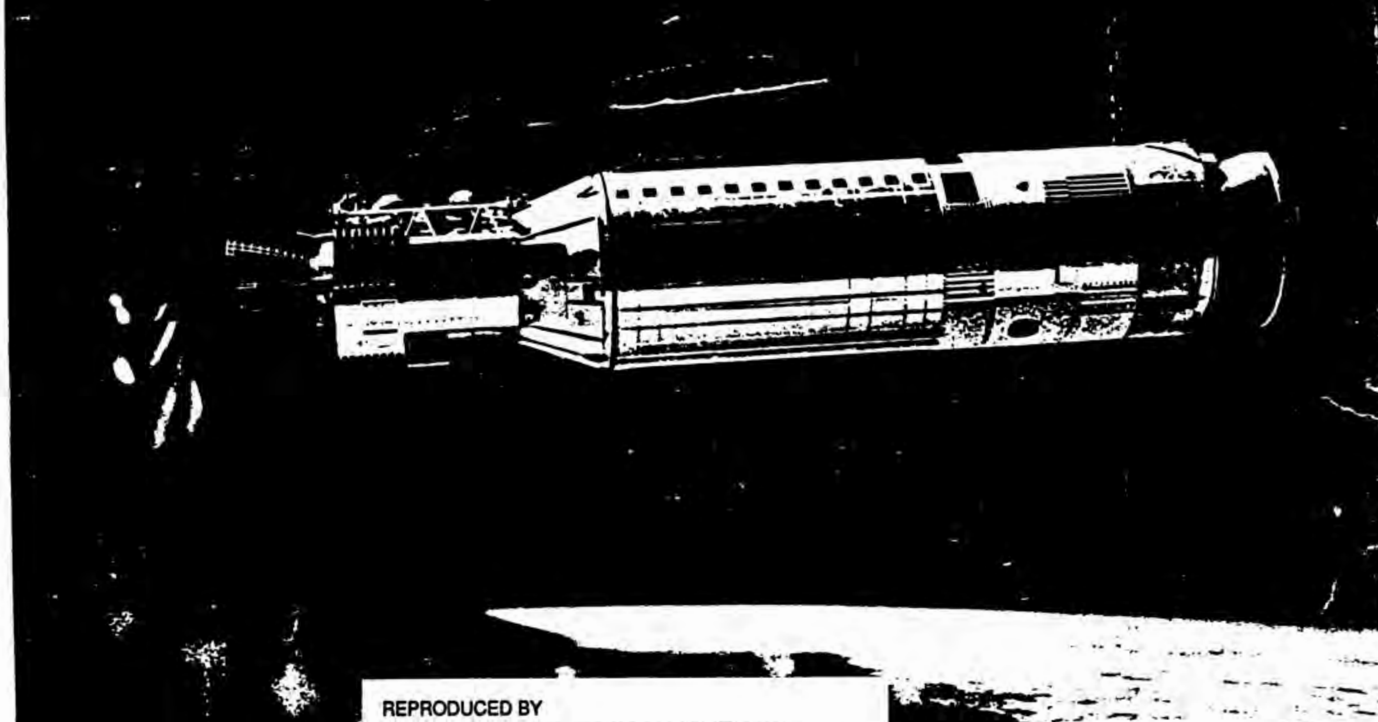
GEMINI SUMMARY CONFERENCE



February 1-2, 1967 Manned Spacecraft Center Houston, Texas

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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Scientific and Technical Information Division
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GEMINI SPACECRAFT FLIGHT HISTORY

MISSION	DESCRIPTION	LAUNCH DATE	MAJOR ACCOMPLISHMENTS
Gemini I	Unmanned 64 orbits Qualification	Apr. 8, 1964	Demonstrated structural integrity, and launch vehicle systems performance.
Gemini II	Unmanned Suborbital Qualification	Jan. 19, 1965	Demonstrated spacecraft systems performance.
Gemini III	Manned 3 orbits Qualification	Mar. 23, 1965	Demonstrated manned qualification of the Gemini spacecraft.
Gemini IV	Manned 4 days Long duration	June 3, 1965	Demonstrated spacecraft systems performance and crew capability for 4 days in space, and demonstrated extravehicular activity.
Gemini V	Manned 8 days Long duration	Aug. 21, 1965	Demonstrated long-duration flight, demonstrated rendezvous radar capability, and rendezvous maneuvers.
Gemini VI	Manned 2 days Rendezvous (Canceled after failure of Gemini Agena Target Vehicle)	Oct. 25, 1965	Demonstrated dual countdown procedures (Gemini Atlas-Agena Target Vehicle and Gemini Launch Vehicle/spacecraft), and flight performance of the Target Launch Vehicle and flight readiness of the Gemini Agena Target Vehicle secondary propulsion system.
Gemini VII	Manned 14 days Long duration	Dec. 4, 1965	Demonstrated 2-week duration flight and station keeping with Gemini Launch Vehicle Stage II, evaluated shirt-sleeve environment, acted as the rendezvous target for Spacecraft 6, and demonstrated controlled reentry to within 7 miles of planned landing point.
Gemini VI-A	Manned 1 day Rendezvous	Dec. 15, 1965	Demonstrated on-time launch procedures, closed-loop rendezvous capability, station-keeping technique with Spacecraft 7.

(Continued inside back cover)

FOREWORD

The Gemini Summary Conference was held on February 1 and 2, 1967, at the NASA Manned Spacecraft Center, Houston, Tex. The conference emphasized the highlights of the Gemini Program and especially the flight results of the last five missions. This report contains the 21 technical papers presented at the conference as well as an introduction by George E. Mueller and concluding remarks by George M. Low.

The technical papers are divided into five sections: the first describes the rendezvous, docking, and tethered-vehicle operations involving the spacecraft and a target vehicle; the second presents various aspects of extravehicular activity; the third concerns the operational support of the missions; the fourth covers the experiments conducted during the missions; and the fifth compares the astronaut flight and simulation experiences and relates the Gemini results to the Apollo Program.

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1. INTRODUCTION

By George E. Mueller, *Associate Administrator for Manned Space Flight, NASA*

The Gemini Program is over. The papers in this report summarizing the program were prepared by some of the people who contributed to the overall success. In each case, the authors were actual participants and provide a cross section of what may be called the Gemini team. As is true in any undertaking of this magnitude, involving many diverse organizations and literally thousands of people, a vital element of the Gemini success may be traced to teamwork. In the purest definition of the word, wherein individual interests and opinions are subordinate to the unity and efficiency of the group, the Gemini team has truly excelled.

Much has already been written concerning the Gemini achievements, and many of the achievements are presented again in greater depth within this report. By way of introduction, and to set the stage for the following papers, a few words are necessary to assess the achievements in the context of the goals of the national manned space-flight program. Only in this way is it possible to evaluate the significance of the Gemini accomplishments.

The Gemini Program was undertaken for the purpose of advancing the United States manned space-flight capabilities during the period between Mercury and Apollo. Simply stated, the Gemini objectives were to conduct the development and test program necessary to (1) demonstrate the feasibility of long-duration space flight for at least that period required to complete a lunar landing mission; (2) perfect the techniques and procedures for achieving rendezvous and docking of two spacecraft in orbit; (3) achieve precisely controlled reentry and landing capability; (4) establish capability in extravehicular activity; and (5) achieve the less obvious, but no less significant, flight and ground crew proficiency in manned space flight. The very successful flight program of the United States has provided vivid demonstration of the achievements in each of these objective areas.

The long-duration flight objective of Gemini was achieved with the successful completion of Gemini VII in December 1965. The progressive buildup of flight duration from 4 days with Gemini IV, to 8 days with Gemini V and 14 days with Gemini VII, has removed all doubts, and there were many, of the capability of the flight crews and spacecraft to function satisfactorily for a period equal to that needed to reach the lunar surface and return. Further, this aspect of Gemini provides high confidence in flight-crew ability to perform satisfactorily on much longer missions. The long-duration flights have also provided greater insight into, and appreciation of, the vital role played by the astronauts, the value of flexibility in mission planning and execution, and the excellent capability of the manned space-flight control system. As originally conceived, the Gemini Program called for completion of the long-duration flights with Gemini VII, which was accomplished on schedule.

One of the more dramatic achievements has been the successful development of a variety of techniques for the in-orbit rendezvous of two manned spacecraft. The preparation for this most complex facet of Gemini missions was more time consuming than any other. That it was performed with such perfection is a distinct tribute to the Gemini team that made it possible: the spacecraft and launch-vehicle developers and builders, the checkout and launch teams, the flight crews and their training support, and the mission-planning and mission-control people.

The ability to accomplish a rendezvous in space is fundamental to the success of Apollo, and rendezvous was a primary mission objective on each mission after Gemini VII. Ten rendezvous were completed and seven different rendezvous modes or techniques were employed. Nine different dockings of a spacecraft with a target vehicle were achieved. Eleven different astronauts gained rendezvous experience in this most important objective. Several of the rendezvous were designed to simulate some facet of an Apollo rendezvous requirement. The principal focus of the rendezvous activities was, however, designed to verify theoretical determinations over a wide spectrum. Gemini developed a broad base of knowledge and experience in orbital rendezvous and this base will pay generous dividends in years to come.

A related accomplishment of singular importance to future manned space-flight programs was the experience gained in performing docked maneuvers using the target-vehicle propulsion system. This is a striking example of Gemini pioneering activities—the assembly and maneuvering of two orbiting space vehicles.

The first attempt at extravehicular activity during Gemini IV was believed successful, and although difficulties were encountered with extravehicular activity during Gemini IX–A, X, and XI, the objective was achieved with resounding success on Gemini XII. This in itself is indicative of the Gemini Program in that lessons learned during the flight program were vigorously applied to subsequent missions. The extravehicular activity on Gemini XII was, indeed, the result of all that had been learned on the earlier missions.

The first rendezvous and docking mission, although temporarily thwarted by the Gemini VI target-vehicle failure, was accomplished with great success during the Gemini VII/VI–A mission. This mission also demonstrated the operational proficiency achieved by the program. The term "operational proficiency" as applied to Gemini achievements means far more than just the acceleration of production rates and compressing of launch schedules. In addition and perhaps more importantly, operational proficiency means the ability to respond to the unexpected, to prepare and execute alternate and contingency plans, and to maintain flexibility while not slackening the drive toward the objective. Time and again Gemini responded to such a situation in a manner that can only be described as outstanding.

A few comments are in order on what the Gemini accomplishments mean in terms of value to other programs. There is almost no facet of Gemini that does not contribute in some way to the Apollo Program. Aside from the actual proof testing of such items as the manned space-flight control center, the manned space-flight communications net, the development and perfection of recovery techniques, the training of the astronauts, and many others which apply directly, the Gemini Program has provided a high level of confidence in the ability to accomplish the Apollo Program objectives before the end of this decade. The Apollo task is much easier now, due to the outstanding performance and accomplishments of the Gemini team.

Similarly, the Apollo Applications Program has been inspired in large part by the Gemini experiments program, which has sparked the imagination of the scientific community. In addition to the contributions to Apollo hardware development which provide the basis for the Apollo Applications Program, it has been discovered, or rather proved, that man in space can serve many extremely useful and important functions. These functions have been referred to as technological fallout, but it is perhaps more accurate to identify them as accomplishments—that is, accomplishments deliberately sought and achieved by the combined hard labor of many thousands of people. Some of these people have reviewed their work in this report.

The Manned Orbiting Laboratory Program has been undertaken by the Department of Defense for the purpose of applying manned space-flight technology to national defense and is making significant use of the

Gemini accomplishments. This may be considered as a partial repayment for the marvelous support that NASA has received and continues to receive from the DOD. The success of the NASA programs is in no small measure due to the direct participation of the DOD in all phases of the manned space-flight program. This support has been, and will continue to be, invaluable.

The combined Government industry university team that makes up the manned space-flight program totals about 240 000 people. In addition, thousands more are employed in NASA unmanned space efforts, and in programs of the Department of Defense, the Department of Commerce, the Atomic Energy Commission, and other agencies involved in total national space endeavors. These people, in acquiring new scientific knowledge, developing new techniques, and working on new problems with goals ever enlarged by the magnitude of their task, form the living, growing capability of this Nation for space exploration.

For the last quarter century, this Nation has been experiencing a technological revolution. Cooperative efforts on the part of the Government, the universities, the scientific community, and industry have been the prime movers. This cooperation has provided tremendous capability for technological research and development which is available now and which will continue to grow to meet national requirements of the future. The influence of this technological progress and prowess is, and has been, a deciding factor in keeping the peace. Preeminence in this field is an important instrument in international relations and vitally influences this country's dealings with other nations involving peace and freedom in the world. Political realities which can neither be wished away nor ignored make the capability to explore space a matter of strategic importance as well as a challenge to the scientific and engineering ingenuity of man. This Nation can no more afford to falter in space than it can in any earthly pursuit on which the security and future of the Nation and the world depend.

The space effort is really a research and development competition, a competition for technological preeminence which demands and creates the quest for excellence.

The Mercury program, which laid the groundwork for Gemini and the rest of this Nation's manned space-flight activity, appears at this point relatively modest. However, Mercury accomplishments at the time were as significant to national objectives as the Gemini accomplishments are today and as those that are planned for Apollo in the years ahead.

That these programs have been, and will be, conducted in complete openness with an international, real-time audience makes them all the more effective. In this environment, the degree of perfection achieved is even more meaningful. Each person involved can take richly deserved pride in what has been accomplished. Using past experience as a foundation, the exploration of space must continue to advance. The American public will not permit otherwise, or better yet, history will not permit otherwise.

SPACE ORBITAL MANEUVERING

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2. SUMMARY OF RENDEZVOUS OPERATIONS

By W. BERNARD EVANS, *Office of Vehicles and Missions, Gemini Program Office, NASA Manned Spacecraft Center;* and MARVIN R. CZARNIK, *Dynamics Group Engineer, McDonnell Aircraft Corp.*

Introduction

One of the major objectives of the Gemini Program was to develop and to demonstrate techniques for the rendezvous and docking of space vehicles. This objective is of vital importance since rendezvous and docking is mandatory for success in many future manned space-flight programs. For example, lunar orbital rendezvous has been selected as the primary mode for the Apollo lunar-landing mission which requires one rendezvous and two dockings. Other programs requiring rendezvous are planetary missions, manned space stations, and unmanned satellite inspection and repair missions.

During the Gemini Program, the following types of rendezvous techniques were evaluated: fourth orbit ($M = 4$), third orbit ($M = 3$), first orbit ($M = 1$), optical rendezvous, rendezvous from above, stable orbit rendezvous, and optical dual rendezvous. These techniques were used successfully in the completion of 10 rendezvous operations (table 2–I). A major factor in achieving success during these operations can be attributed to the implementation of an extensive analysis, simulation, and training program leading first to the Gemini VI–A rendezvous mission, and subsequently to more complex missions. During the Gemini III mission, the spacecraft propulsion system and the guidance and control system were evaluated. On the Gemini IV mission, a plan was developed and an attempt was made to station keep and rendezvous with the spent second stage of the launch vehicle. During Gemini V, a phantom rendezvous and a spacecraft radar-to-ground transponder tracking test were performed. The phantom rendezvous involved a series of maneuvers based upon ground tracking and computations, and precisely duplicated the maneuver sequence and procedures planned for the midcourse phase of the Gemini VI–A mission.

TABLE 2–I.—*Mission Summary*

Gemini mission	Type of rendezvous
VI–A	Fourth orbit ($M = 4$)
VIII	Fourth orbit ($M = 4$)
IX–A	Third orbit ($M = 3$)
	Optical re-rendezvous
	Re-rendezvous from above
X	Fourth orbit ($M = 4$)
	Optical dual
XI	First orbit ($M = 1$)
	Stable orbit
XII	Third orbit ($M = 3$)

Sufficient data were obtained from the spacecraft radar tracking test during the Gemini V mission to adequately flight-qualify the radar for the Gemini VI–A flight. Even though the rendezvous operations planned for the first three manned Gemini flights were not all successful, they were extremely valuable to the program since they provided flight experience and indicated areas requiring further analysis, simulation, and training.

On December 15, 1965, the Gemini VI–A crew, using the Gemini VII spacecraft as the target vehicle, completed the first space rendezvous operation. Although this mission did not include a docking, it was successful and after lift-off proceeded almost precisely as planned. On the following mission, the Gemini VIII crew successfully performed the first rendezvous and docking with a Gemini Agena Target Vehicle. Subsequent, more complex,

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rendezvous operations were successfully performed during the Gemini IX–A, X, XI, and XII missions. These successes have provided confidence in the ability to accomplish such operations. However, rendezvous must still be recognized as a highly precise operation that is rather unforgiving of errors which occur during the final approach, details of which will be discussed in this paper.

Review of Rendezvous Operations Development

An explanation of rendezvous can be greatly simplified by a description of the relative-motion concept. Figure 2–1 shows a coordinate system centered on the target vehicle in a circular orbit with the X– and Y–axes in the target orbital plane. The Y–axis rotates with the target vehicle and is positive radially upward; the X–axis is curvilinear and positive opposite the direction of motion. The out-of-plane parameter is the Z–axis, which completes the right-hand coordinate system. The motion of the spacecraft with respect to this reference is illustrated in figure 2–2.

Figure 2–2(a) shows the spacecraft in a lower circular orbit. It should be noted that

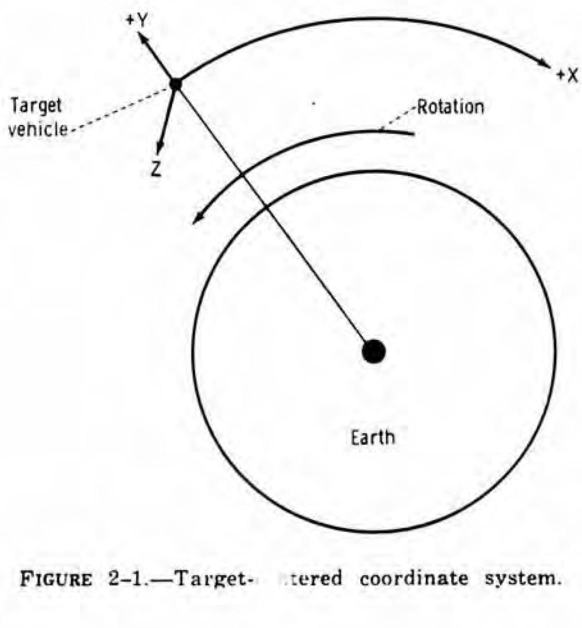


FIGURE 2–1.—Target-centered coordinate system.

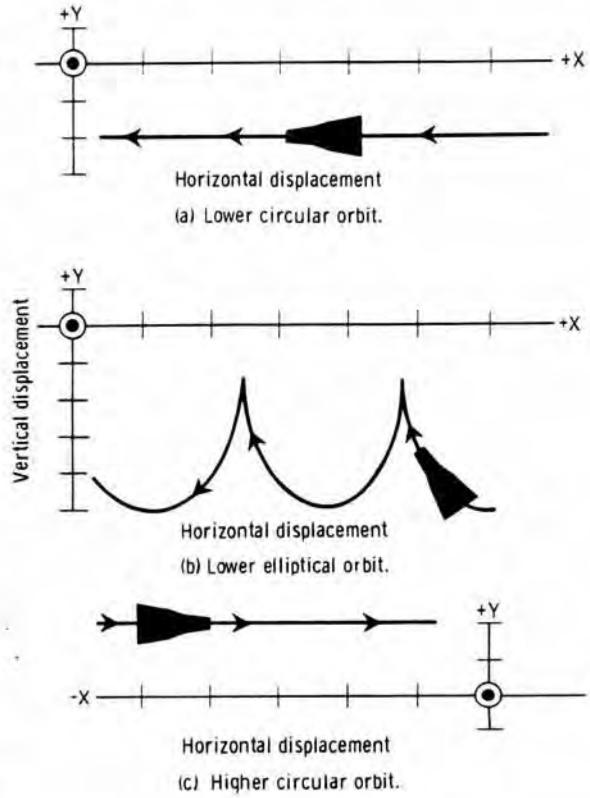


FIGURE 2–2.—Motion relative to a target-centered coordinate system.

the radial displacement Y is constant while the trailing displacement X decreases with time, since the spacecraft in the lower orbit has a higher angular rate. Figure 2–2(b) shows a lower elliptical orbit. As can be expected, this orbit has a catchup rate; however, the radial displacement also changes, with the low points representing perigees, and the high points, apogees. Figure 2–2(c) illustrates a spacecraft in a circular orbit higher than the target orbit. The radial distance is constant, as in the case of the lower circular orbit; however, in this case the trailing displacement changes since the target now has the higher angular rate. The following paragraphs use this coordinate system in describing the Gemini rendezvous operations.

The development of the operational rendezvous missions required extensive analyses as previously described in reference 1. For

Gemini VI, many concepts were evaluated and three were selected as candidates for the Gemini VI mission. The first was the tangential concept which included the tangential approach of the spacecraft to the target vehicle following four orbits of ground-controlled midcourse maneuvers. The second concept had a similar catchup sequence, except that the final midcourse maneuver established a coelliptical approach trajectory, and the spacecraft closed-loop guidance system was then used to establish a collision course. A third concept featured rendezvous at first spacecraft apogee. Following a tangential approach of the spacecraft to the target, the spacecraft would be inserted on a collision course with the target, and the spacecraft closed-loop system would be used to correct insertion dispersions.

After the three concepts had been selected, analyses were performed to determine the concept best suited for the Gemini VI mission. In June 1964, prior to the flight of Gemini II, the coelliptical rendezvous concept was selected for the Gemini VI mission.

Description of Initial Rendezvous Operations

Gemini VI-A, VIII, and X

Figures 2-3 and 2-4 present typical relative trajectory plots of the fourth-orbit rendezvous conducted on Gemini VI-A, VIII, and X. On each mission, the spacecraft was inserted into an orbit essentially coplanar with the target vehicle. The first orbit was left free of rendezvous maneuvers to allow the crew sufficient time to verify satisfactory spacecraft operation. A number of midcourse corrections were performed before completing the rendezvous during the fourth spacecraft orbit near the end of the fourth darkness period. At the first spacecraft perigee, an apogee height-adjust maneuver N_H was performed to correct for in-plane insertion dispersions. At the second apogee, a phase-adjust maneuver N_{C1} was performed to raise the perigee, thus providing the catchup rate required for proper phasing of the terminal-phase initiation near the fourth darkness entry. An out-of-plane correction

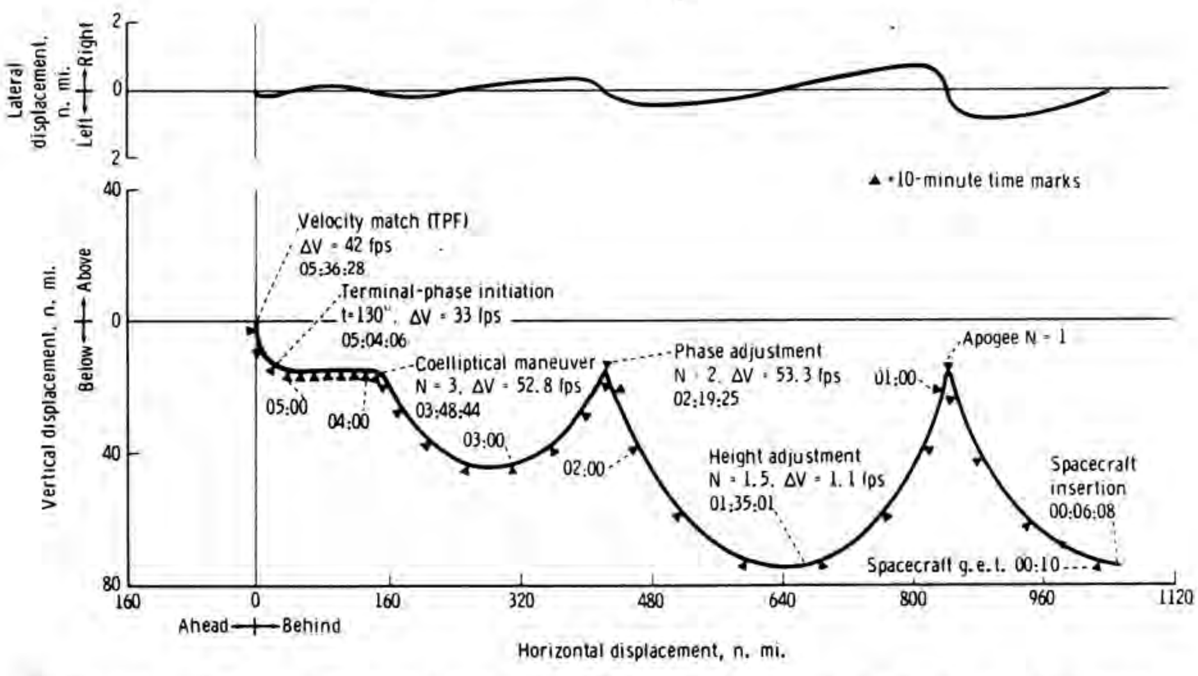


FIGURE 2-3.—Typical relative trajectory of spacecraft from insertion to rendezvous in target-vehicle curvilinear coordinate system. Gemini VI-A, VIII, and X missions.

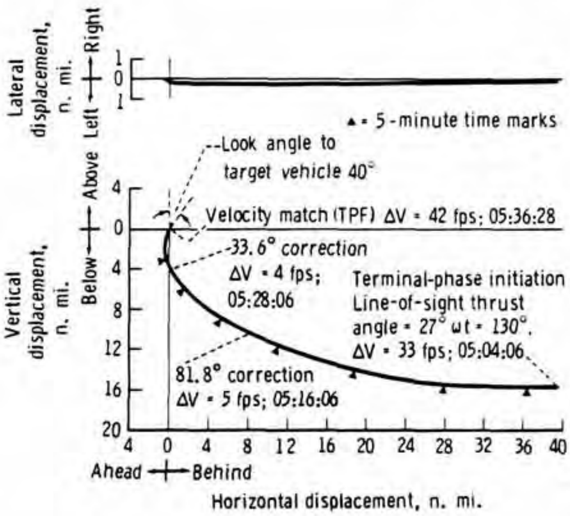


FIGURE 2-4.—Typical relative trajectory of spacecraft from terminal-phase initiation to rendezvous in target-vehicle curvilinear coordinate system. Gemini VI–A, VIII, and X missions.

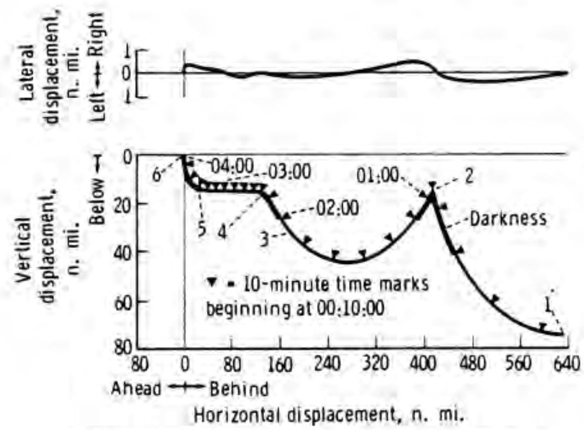
P_c was applied at the nodal crossing after the second apogee to correct out-of-plane insertion dispersions. At the third spacecraft apogee, a coelliptical maneuver N_{SR} was performed to produce a constant altitude differential of 15 nautical miles. The onboard system then provided solutions for the terminal-phase-initiation (TPI) maneuver, which would occur when the line-of-sight elevation angle reached the nominal value of 27°. Two vernier corrections followed at 12-minute intervals. Finally, braking (terminal-phase finalization (TPF)) and line-of-sight rate control were effected by a manual operation based upon radar and visual data.

The transfer trajectory was selected to satisfy several of the mission requirements in the area of onboard procedures. First, in order to provide a backup reference direction for the terminal-phase-initiation maneuver in case of a guidance-system failure, the maneuver had to be performed along the line of sight to the target. The second requirement was a low terminal line-of-sight angular rate and a low closing rate. Finally, the terminal-phase-initiation point had to be below and behind the target vehicle; and the final approach, from below and ahead of the target vehicle, in order to optimize the lighting. These factors were evaluated, and a 130° transfer was selected.

The selection of the nominal coelliptical differential altitude of 15 nautical miles was based upon a tradeoff between two considerations. First, the range to the target at the terminal-phase-initiation point had to be small enough to assure visual acquisition. Second, a large differential altitude was required to minimize the effect of insertion dispersions and catchup maneuver errors on the location of the terminal-phase-initiation point. For example, a differential altitude of 15 nautical miles resulted in a 3-sigma dispersion of ±8 minutes in the timing of the terminal-phase-initiation maneuver. Early error analysis indicated a ±15-minute variation in terminal-phase-initiation timing for a differential altitude of 7 nautical miles. Flight experience demonstrated that the launch vehicle and spacecraft guidance systems accuracies, crew procedures, and ground-tracking accuracy were better than had been expected; as a result, the altitude differential was reduced to 5 and 7 nautical miles in the later rendezvous operations.

Gemini IX–A and XII

A second primary rendezvous technique was utilized on Gemini IX–A and XII (figs. 2–5 and 2–6). This technique resulted in rendezvous in the third spacecraft orbit near the end of the third spacecraft darkness period. A phase-adjust maneuver N_c, was performed at first spacecraft apogee to provide the correct phasing at the second apogee. Approximately three-fourths of an orbit later, the first of a set of two maneuvers was performed: a combination phasing, height-adjust, and out-of-plane correction. The first maneuver N_{cc}, combined with the following coelliptical maneuver, provided a fixed rendezvous time with minimum propellant usage. The out-of-plane portion of the first maneuver established a node at the following coelliptical maneuver point. The coelliptical maneuver N_{SR} eliminated the out-of-plane



1. Spacecraft insertion; 00:06:21
2. Phase adjustment N = 1, ΔV = 53.4 fps; 00:59:39
3. Corrective combination N = 1.75, ΔV = 0.8 fps; 01:57:00
4. Coelliptical maneuver N = 2, ΔV = 52.7 fps; 02:20:32
5. Terminal-phase initiation ΔV = 32.4 fps; 03:27:07, ωt = 130°
6. Velocity match (TPF) ΔV = 41.6 fps; 03:59:52

FIGURE 2-5.—Typical relative trajectory. Gemini IX-A and XII missions.

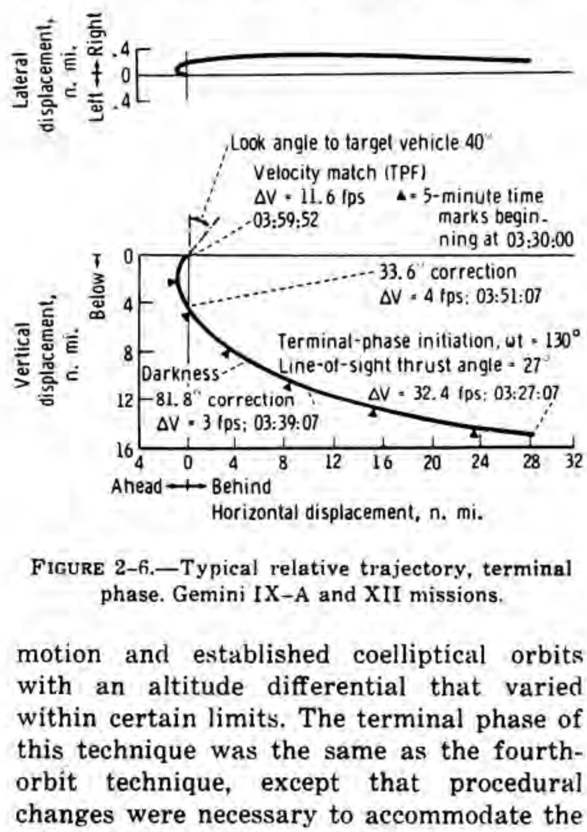


FIGURE 2-6.—Typical relative trajectory, terminal phase. Gemini IX-A and XII missions.

Gemini XI

The third primary rendezvous conducted during the program was the first-orbit technique used for Gemini XI (figs. 2-7 and 2-8). The limited time available to conduct the first-orbit rendezvous prohibited the multi-correction catchup phase and coelliptical approach used on other missions. Instead, a correction was made at spacecraft insertion to remove out-of-plane motion and to adjust motion and established coelliptical orbits with an altitude differential that varied within certain limits. The terminal phase of this technique was the same as the fourth-orbit technique, except that procedural changes were necessary to accommodate the variable altitude differential.

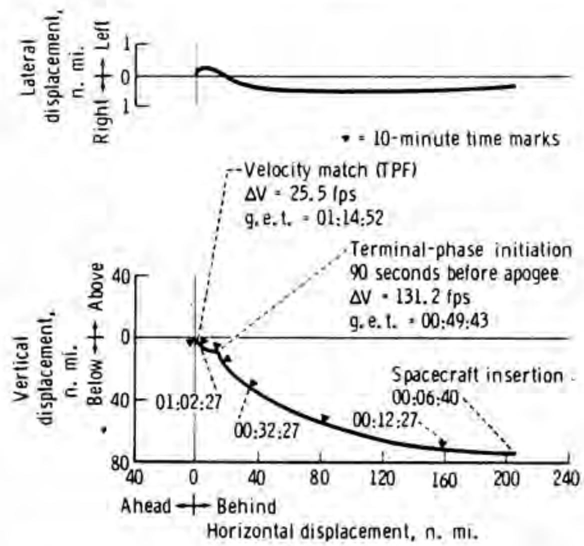


FIGURE 2-7.—Relative trajectory. Gemini XI mission.

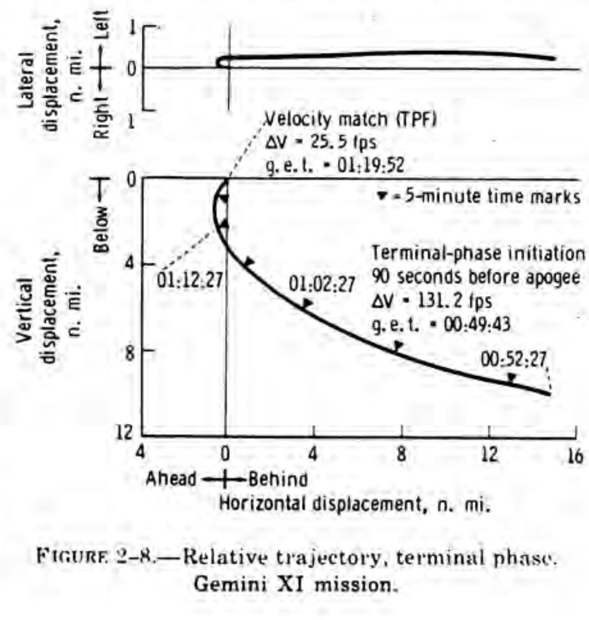


FIGURE 2-8.—Relative trajectory, terminal phase. Gemini XI mission.

apogee height and phasing. This correction was based upon onboard navigation information obtained from the spacecraft guidance system. At 90° after insertion, a second out-of-plane correction, also based upon onboard information, was performed. Terminal-phase initiation occurred just prior to first spacecraft apogee with the spacecraft 10 nautical miles below and 15 nautical miles behind the target vehicle. A 120° transfer was used with two vernier corrections at 12-minute intervals after the terminal-phase initiation. After a manual braking and line-of-sight phase, rendezvous was completed within the first orbit.

Description of Re-Rendezvous and Dual Rendezvous Operations

The first of three re-rendezvous techniques was an optical rendezvous from an equiperiod orbit and was conducted on the Gemini IX–A mission (fig. 2–9). The purpose of this rendezvous was to evaluate the optical rendezvous procedures, and particularly the terminal-phase lighting, required for the dual rendezvous scheduled for Gemini X. An upward radial velocity change was used to separate the spacecraft from the target vehicle into an equiperiod orbit. Approximately one-half orbit after separation, a correction

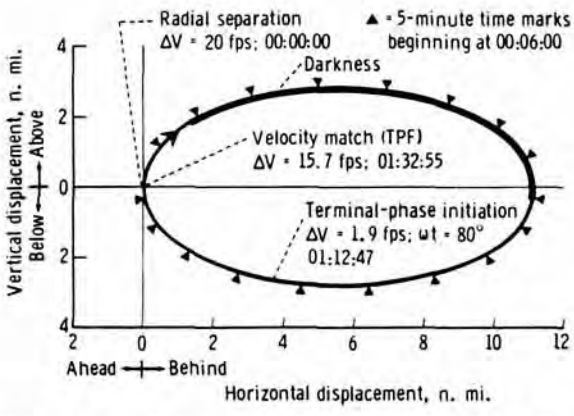


FIGURE 2–9.—Relative trajectory of spacecraft for (equiperiod) re-rendezvous in target vehicle curvilinear coordinate system. Gemini IX–A mission.

was applied based upon the time the line of sight to the target vehicle crossed the local horizontal. The time and the magnitude of the terminal-phase-initiation maneuver were determined from visual angle observations, and an 80° transfer was initiated when the Sun was nearly overhead. Two vernier corrections also based upon visual angle measurements were applied, and rendezvous occurred just prior to sunset. It was a requirement that the spacecraft be in a station-keeping mode prior to entering darkness with a passive target.

A second re-rendezvous technique (figs. 2–10 and 2–11) was developed to evaluate a terminal-phase condition with an Earth background. Two midcourse maneuvers were used to insert the spacecraft into a coelliptical orbit 7.5 nautical miles above the target vehicle. Except for a reversal in approach direction, the terminal phase was identical to that employed on the earlier coelliptical approach from below. Experience gained during this rendezvous indicates that the probability of success would be very low in case of a radar guidance system failure because of the extremely poor target visibility.

During the Gemini XI mission, a third re-rendezvous exercise was performed. This rendezvous was ground controlled except that the terminal braking and line-of-sight control phases were performed by the crew using visual observations (no radar). After the initial separation maneuver, the spacecraft was in a nearly circular orbit at the same altitude as the target vehicle, but with a trailing displacement of approximately 25 nautical miles. Since the relative motion of the vehicles in this configuration was approximately zero, the rendezvous was referred to as a stable-orbit rendezvous (fig. 2–12). A ground-computed maneuver was performed which placed the spacecraft on a trajectory to intercept the target vehicle in 292° of target orbital travel. With 34° of orbital travel remaining, a second and final ground-computed maneuver was applied. The rendezvous was then completed by the flight crew using visual cues. The terminal-phase portion of

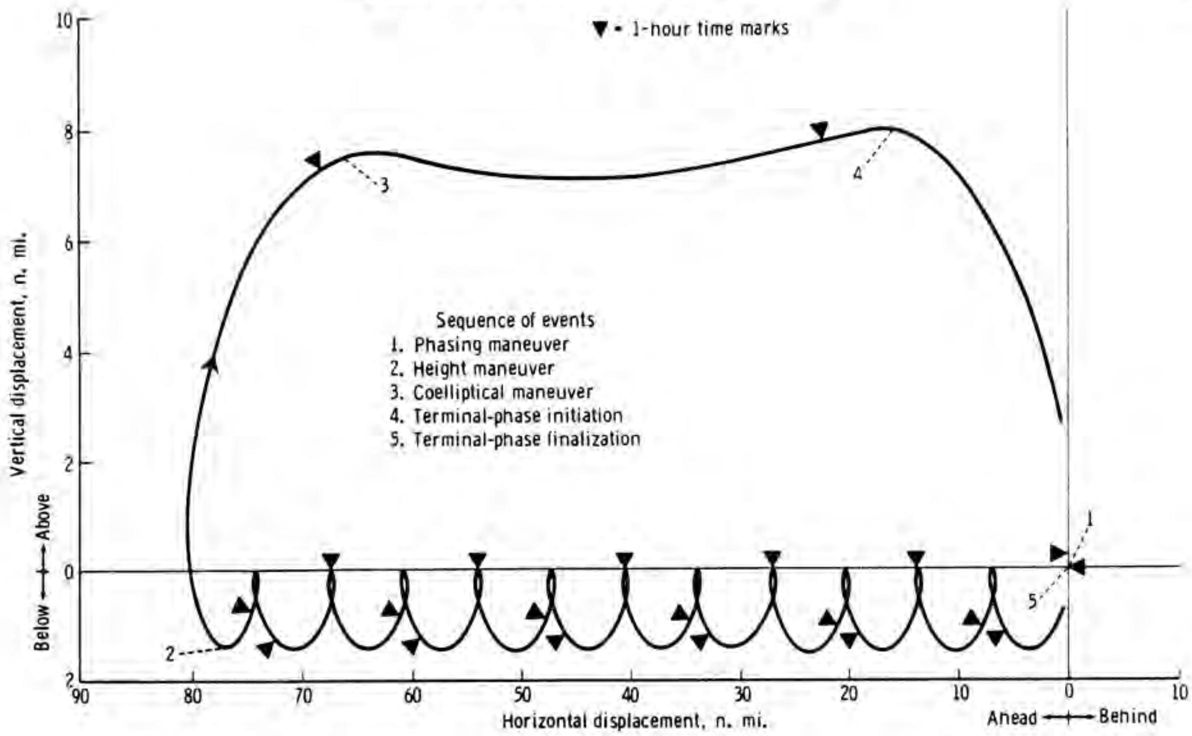


FIGURE 2-10.—Relative trajectory profile for re-rendezvous from above. Gemini IX-A mission.

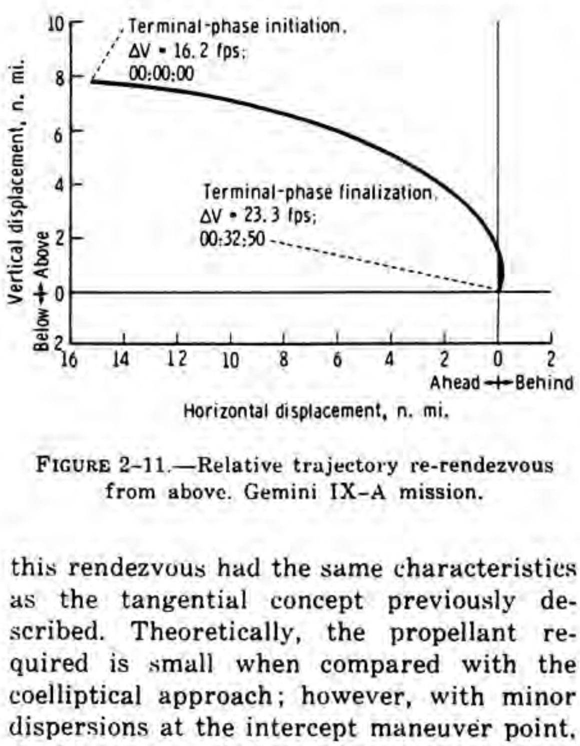


FIGURE 2-11.—Relative trajectory re-rendezvous from above. Gemini IX-A mission.

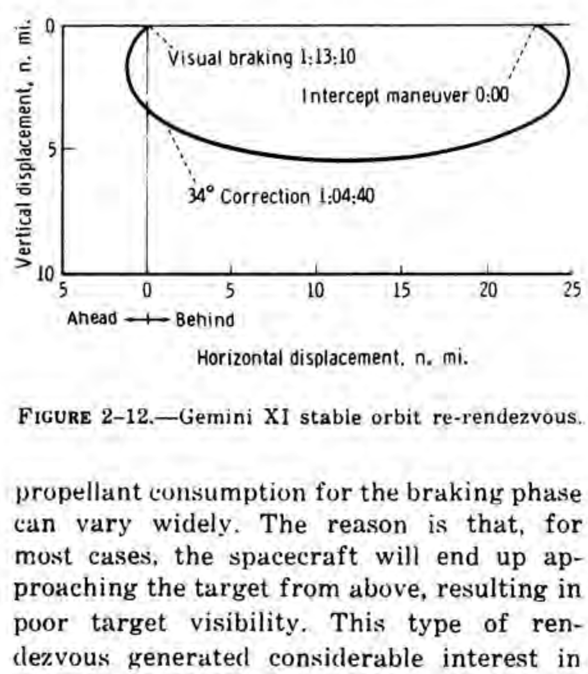


FIGURE 2-12.—Gemini XI stable orbit re-rendezvous.

this rendezvous had the same characteristics as the tangential concept previously described. Theoretically, the propellant required is small when compared with the coelliptical approach; however, with minor dispersions at the intercept maneuver point, the lighting conditions, approach angles, and propellant consumption for the braking phase can vary widely. The reason is that, for most cases, the spacecraft will end up approaching the target from above, resulting in poor target visibility. This type of rendezvous generated considerable interest in its application to certain rendezvous operations, particularly where a highly precise ground-tracking system is used to provide the terminal-phase maneuvers. The commitment to conduct such a rendezvous reflected the confidence that was established during Gemini in the capabilities of the ground-tracking, computation, and control facilities.

In addition to the primary and re-rendezvous missions, a dual rendezvous was performed by the Gemini X crew. The target vehicle launched during the Gemini VIII mission was left in orbit and was the passive target for the dual operation. One problem encountered during the development of the Gemini X mission was obtaining precise state vectors for the passive target vehicle, and making accurate predictions far enough in advance to find acceptable launch windows. Because of the inaccuracies in drag prediction, it was necessary for launch date, lift-off time, and catchup sequence to be flexible. The catchup sequence included a series of maneuvers by the docked Gemini X spacecraft and Gemini X target vehicle for gross catchup, and another series of maneuvers by the undocked spacecraft for fine catchup. The capability for large changes in altitude during the gross catchup sequence allowed an acceptable wide variation in the initial-phase angle. The terminal approach was coelliptical with an altitude differential of 7 nautical miles; the terminal-phase guidance employed was the same as for the optical rendezvous conducted on Gemini IX-A.

Rendezvous Considerations and Flight Results

In developing the rendezvous missions, many factors were considered, primarily launch procedures, system requirements, and crew procedures.

Launch Procedures

Development of the launch procedures required extensive analyses to define methods of controlling out-of-plane displacement, establishing launch-window length, and developing a countdown method.

Selecting a target orbit inclination slightly above the latitude of the launch site makes the out-of-plane displacement relatively small for a long period of time (fig. 2-13). By varying the launch azimuth so that the spacecraft would be inserted parallel to the target-vehicle orbital plane, the out-of-plane displacement of the launch site at the time of launch becomes the maximum out-of-plane displacement between the two orbit planes. The out-of-plane displacement could also be minimized by using the variable launch-azimuth technique with guidance in yaw during second-stage powered flight. This is accomplished by biasing the launch azimuth of the spacecraft so that the launch azimuth is at an optimum angle directed toward the target-vehicle orbital plane (fig. 2-14). As a result, the out-of-plane distance would be reduced prior to the initiation of closed-loop guidance during the second-stage flight. This technique would effectively use the launch-vehicle performance capability to control the out-of-plane displacement. Sufficient performance capability existed in the Gemini Launch Vehicle to control the out-of-plane displacement to within $\pm 0.55°$ (table 2-II). The maximum allowable wedge angle of ± 0.55 was not needed on any of the rendezvous missions. By selecting an inclination of

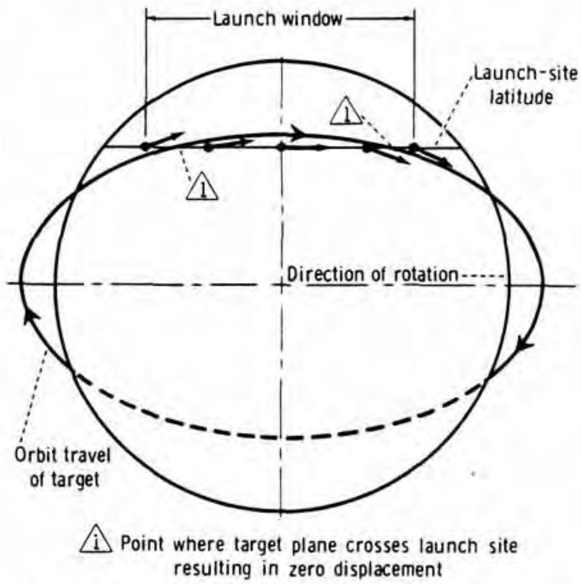


FIGURE 2-13.—Variable azimuth launch technique.

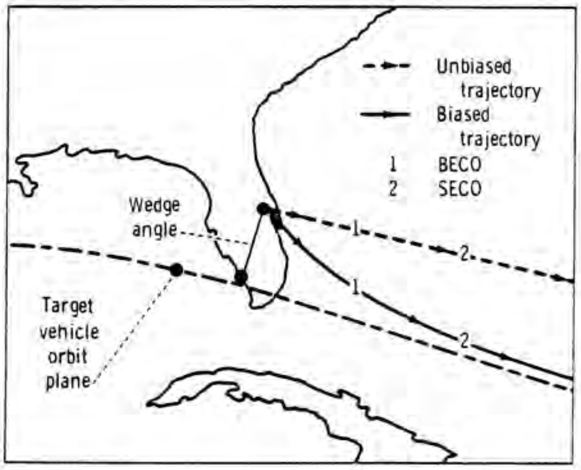


Figure 2-14.—Typical Gemini rendezvous launch. Biased launch azimuth and Stage II yaw steering.

Table 2-II—*Yaw Steering Summary*

Gemini mission	Targeted out-of-plane displacement, deg
VI–A	0.20
VIII	–.21
IX–A	–.50
X	–.077
XI	–.131
XII	–.16

28.87°, 0.53° above the launch-site latitude, and by using a variable launch-azimuth technique, the out-of-plane displacement could be controlled to within 0.53° for 135 minutes.

During the early planning phases of the Gemini Program, a relatively large launch window (table 2–III) was considered mandatory; however, later experience indicated that reliable countdown procedures could be developed, and it is now the general opinion that large launch windows are not required. Since Gemini V, the launches have either been essentially on time, or the launch has been scrubbed. By suitable planning, minor launch delays can be easily absorbed in the count, and if major problems occur, large launch-window lengths are not particularly helpful. An on-time launch capability provides a tremendous potential in planning operational rendezvous missions and indicates

Table 2-III.—*Gemini Launch Performance*

Mission	Launch attempts	Launch date	Launch-time deviation
I	1	Apr. 8, 1964	On time
II	2	Jan. 19, 1965	–4 min
III	1	Mar. 23, 1965	–24 min
IV	1	June 3, 1965	–16 min
V	2	Aug. 21, 1965	On time
VI	1	(a)	—
VI–A	2	Dec. 15, 1965	On time
VII	1	Dec. 4, 1965	On time
VIII	2	Mar. 16, 1966	On time
IX	1	(b)	—
IX–A	2	June 3, 1966	On time
X	1	July 18, 1966	On time
XI	3	Sept. 12, 1966	On time
XII	3	Nov. 11, 1966	On time

[a] Target-vehicle failure.
[b] Target launch-vehicle failure.

that rendezvous operations, booster performance permitting, are operationally feasible at any orbital inclination.

Initial analyses of countdown methods indicated that the highest probability of mission success could be achieved by simultaneously counting down both vehicles. Even though simultaneous countdowns have been used extensively in Gemini, nothing in the results clearly indicates that this is a necessity.

Systems Requirements

A primary consideration in the development of the rendezvous operations was the area of systems requirements. The requirements for the systems design were based upon design-reference missions. As the designs became established, however, the operational missions were developed to exploit the systems capabilities, and, of course, the missions were ultimately limited by the systems capabilities. For example, a desired objective during the Gemini XII mission planning was to complete a rendezvous during the second orbit ($M = 2$). Accomplishing this objective within acceptable dispersions would have required a trajectory correction based on radar range at a point outside the spacecraft radar-range capability. As a result, the second-apogee rendezvous plan was eliminated.

Crew Procedures

Further requirements were imposed to achieve workable crew procedures. The major requirements in this area were the following:

(1) Sufficient time for the crew to complete the necessary activities

(2) Approach trajectories which are reasonably insensitive to insertion dispersion and to errors in midcourse maneuvers

(3) Lighting conditions which are compatible with backup procedures

(4) Low terminal-approach velocities and line-of-sight angular rates

(5) Backup procedures for guidance-systems failures

The requirement to allow sufficient time for crew procedures had an effect on several of the Gemini missions. For example, the first orbits of the Gemini VI–A and VIII missions were free of rendezvous maneuvers, allowing the crew sufficient time to verify the satisfactory operation of all spacecraft systems. The Gemini X primary rendezvous was changed from a third-orbit to a fourth-orbit rendezvous to allow the crew sufficient time to conduct the heavy procedural workload required by the star-horizon onboard orbit determination.

The second procedural requirement, approach trajectories which are reasonably insensitive to insertion dispersion and errors in midcourse maneuvers, was also important in the development of the fourth-orbit rendezvous. An objective was to develop a mission which could effect a near-nominal terminal-approach trajectory notwithstanding insertion dispersions, spacecraft equipment degradation, or ground tracking and computation errors. This objective established the need for the development of backup terminal-phase procedures in the event of a guidance-component failure.

The need for lighting conditions (fig. 2–15) compatible with backup procedures affected all the rendezvous missions. The desired lighting situation for an active target was that the crew (1) see the target by reflected sunlight prior to and at terminal-phase initiation, (2) see the target acquisition lights against a star background during the terminal transfer, and (3) see the target by reflected sunlight for docking after exit from darkness. This lighting situation enabled the crew to maintain target visibility throughout the terminal-rendezvous operations, and established the capability for making inertial line-of-sight angle measurements in the event of a guidance platform failure. The lighting requirement was a factor in selecting the location of the terminal-phase-initiation point, the central angle of the transfer, and the terminal-approach angle. The desirable lighting conditions for rendezvous with an active target were different than for rendezvous with a passive target (fig. 2–16). Since a passive target would not

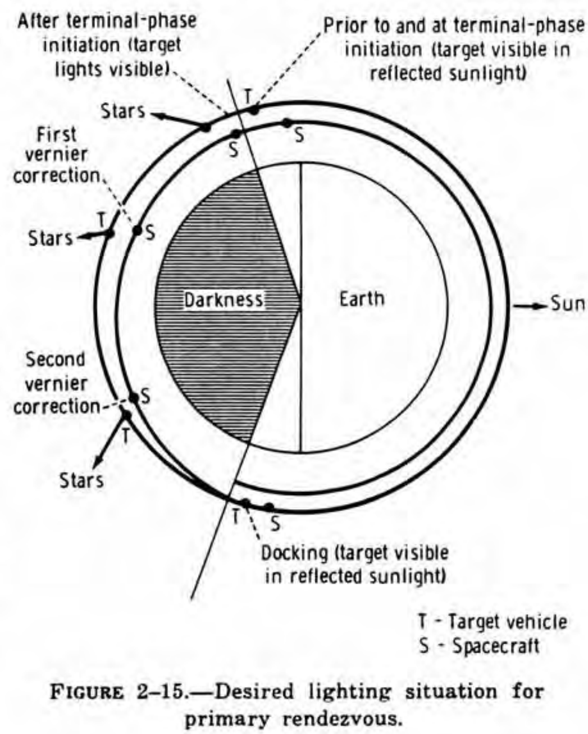


FIGURE 2–15.—Desired lighting situation for primary rendezvous.

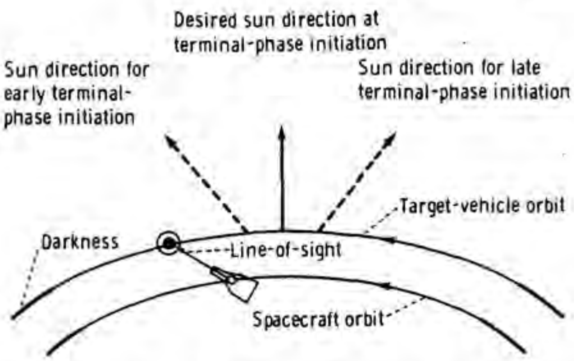


FIGURE 2-16.—Desired lighting situation for passive rendezvous.

be visible in darkness, the terminal-phase portion of the Gemini X dual optical rendezvous was conducted entirely in daylight. The desired terminal-phase initiation occurred near the midpoint of the daylight period. Earlier initiations would have placed the sunline too near the line of sight to the target, thereby obscuring target visibility. Later initiations would not have allowed adequate time in daylight for completing the rendezvous. Gemini experience has shown that lighting is not a major constraint for an active rendezvous provided the spacecraft guidance system does not fail during the terminal approach; but lighting is a major constraint for an optical rendezvous.

The fourth requirement was that the terminal trajectory allow a low terminal-approach velocity and low line-of-sight angular rate. The requirement was important in selecting the trajectory parameters for the coelliptical and the first-orbit rendezvous plans. The 130 transfer utilized on several of the missions was chosen primarily because of the low line-of-sight angular rate near intercept. The biased apogee approach was selected for Gemini XI because the direct tangential approach would have resulted in a high closing velocity.

Throughout the Gemini Program, there was a question of the level of effort to be applied to the development of backup procedures to accommodate guidance-system failures. During the Gemini XI first-orbit rendezvous mission, a problem with the radar system developed just prior to the final terminal-phase midcourse correction. Even though a backup solution for this maneuver was computed and applied, rendezvous could have been accomplished without the correction, since the correction required in this particular instance was small (2 ft/sec). However, on Gemini XII, a failure of a primary guidance-system component required the use of the backup procedures. The radar system failed prior to the terminal-phase-initiation maneuver on this mission, and backup procedures were employed throughout the terminal phase to complete the rendezvous.

The terminal phase of a rendezvous operation involves precision maneuvers and careful control of closing and line-of-sight rates. Table 2–IV compares fuel expenditures encountered during terminal-phase operations with the theoretical minimum. A considerable variation exists between the ratio of actual-to-minimum propellant for various types of terminal-phase conditions, and also for different flights using the same or similar terminal-phase conditions. This variation reflects the critical nature of the task, in that fairly small velocity vector errors can cascade to high propellant consumption or failure to complete the rendezvous. The braking operation is particularly critical. Braking too soon will increase line-of-sight control requirements, and require more time to control the spacecraft during the closing sequence.

An additional comparison of rendezvous performance is shown in table 2–V where the actual terminal-phase vernier corrections are compared with the preflight minimal predicted. This comparison provides an especially good measure of guidance-system performance, since the maneuvers were nominally very small and became large only with degradation of guidance-system performance or with control difficulties.

A number of terminal-phase rendezvous operations were satisfactorily completed during the Gemini Program by using optical

TABLE 2-IV.—*Rendezvous Propellant Usage*

Gemini mission	Type of rendezvous	Conditions at start of terminal phase	Propellant usage, lb		
			Actual	Minimum	Ratio
VI-A	$M=4$	Coelliptic: $\Delta h = 15$ n. mi. $\Delta X = 25$ n. mi.	130	81	1.60
VIII	$M=4$	Coelliptic: $\Delta h = 15$ n. mi. $\Delta X = 25$ n. mi.	160	79	2.02
IX-A	$M=3$	Coelliptic: $\Delta h = 12$ n. mi. $\Delta X = 22$ n. mi.	113	68	1.66
IX-A	Optical	$\Delta h = 2.5$ n. mi. $\Delta X = 3.5$ n. mi.	61	20	3.05
IX-A	From above	$\Delta h = -7.5$ n. mi. $\Delta X = -10$ n. mi.	137	39	3.51
X	$M=4$	Coelliptic: $\Delta h = 15$ n. mi. $\Delta X = 30$ n. mi.	360	84	4.28
X	Optical dual	Coelliptic: $\Delta h = 7$ n. mi. $\Delta X = 12$ n. mi.	180	73	2.46
XI	$M=1$	Spacecraft at apogee of 87/151 orbit: $\Delta h = 10$ n. mi. $\Delta X = 15$ n. mi.	290	191	1.52
XI	Stable orbit	$\Delta h = 0$ n. mi. $\Delta X = 25$ n. mi.	87	31	2.81
XII	$M=3$	Coelliptic: $\Delta h = 10$ n. mi. $\Delta X = 20$ n. mi.	112	55	2.04

TABLE 2-V.—*Vernier Correction Solutions for Primary Rendezvous*

Gemini mission	Actual correction, ft/sec		Nominal correction, ft/sec	
	First	Second	First	Second
VI-A	11	7	1	2
VIII	15	9	1	0
IX-A	1	3	2	3
X	20	23	2	3
XI	6	2	0	2
XII	2	5	2	3

techniques alone (no closed-loop radar-computer operation). Optical rendezvous requires careful control of lighting conditions, and a stabilized reference such as an inertial platform is highly desirable. During simulations, rendezvous have been effected without platform information; however, the probability of success is relatively low.

Concluding Remarks

The rendezvous operations conducted on Gemini have demonstrated that rendezvous is operationally feasible with an active or a passive target. It has also been demonstrated that the operation can be performed using only onboard guidance information after lift-off; using only ground-supplied information; or by using a combination of onboard and ground-supplied information.

Reference

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3. GROUND CONTROL AND MONITORING OF RENDEZVOUS

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Summary

This paper discusses the ground control and monitoring function performed in support of the Gemini rendezvous missions. Included are discussions of the support philosophy adopted for Gemini; the resulting influence upon mission design; and comparisons between predicted and actual flight results.

Introduction

The concepts adopted for the ground support of Gemini were in keeping with the basic mission-design criterion of maximizing the probability of achieving rendezvous. A flexible ground system was developed to permit the flight-control team to react to anomalous situations routinely, while still preserving standardized conditions for the terminal-phase rendezvous. Since the possibility existed for a multitude of anomalous situations, a real-time mission-planning capability was implemented in the Mission Control Center—Houston. This capability consisted of computer-driven displays which permitted the flight controllers to assess current conditions, and to select a maneuver sequence compatible with mission constraints. In effect, the role of the flight controllers was to provide a series of midcourse maneuvers which achieved a particular relative separation and velocity between the spacecraft and the target vehicle. Following the final midcourse maneuver, the role changed more to monitoring the onboard-computed intercept maneuver and the final terminal-phase operations. The following discussion will compare, from a ground-support standpoint, the primary rendezvous missions as well as the re-rendezvous operations which may be conducted during a flight.

Gemini Rendezvous Missions

The ground support of a rendezvous mission was planned so that all information that the flight crew would nominally request, plus additional backup information, would be available at an optimum time in the flight plan. Once the basic mission plan was developed, a large number of final details had to be refined in simulations of the mission with the actual flight-crew personnel. The primary maneuver updates from the Mission Control Center—Houston had to be scheduled at a time that would afford maximum radar tracking history in the mission computers at the Manned Spacecraft Center, Houston. The Gemini rendezvous missions were separated into two distinct mission phases, the midcourse maneuver and the terminal rendezvous. For the midcourse phase, the flight-control team was the primary source for the maneuver computations. The purpose of these maneuvers was to effect a rendezvous between the spacecraft and a point in space that would result in the desired spacecraft displacement and velocity with respect to the target vehicle. To accomplish this, preestablished maneuver points were selected so that the propellant requirements for this mission phase were minimized, and sufficient network tracking was available for maneuver updates. Of course, the first rendezvous mission, Gemini VI-A, had the most uncertain conditions. Consequently, for this mission, a plan was selected which afforded rendezvous in the fourth spacecraft revolution with the following salient features:

(1) The Gemini Launch Vehicle was targeted to provide the desired altitude differential between the target and spacecraft orbits at spacecraft apogee. Also, a dogleg launch trajectory was flown in order to insert the spacecraft into the plane of the target orbit.

(2) The first spacecraft orbit was free of rendezvous activity so the crew could make the necessary systems checks, and the ground controllers could determine the precise spacecraft orbit.

(3) Preestablished maneuver points were selected to account for expected dispersions in lift-off time and spacecraft insertion conditions.

(4) The site chosen to update a maneuver had acquisition so that adequate time remained for the crew to orient the spacecraft to the maneuver attitude.

The midcourse maneuver sequence can be seen in figure 3–1. Tracking during the first revolution indicated that the altitude differential at spacecraft apogee exceeded the acceptable tolerance; thus, the initial midcourse translation was a height adjustment performed at spacecraft perigee near the end of the first revolution. The second midcourse maneuver was a phase adjustment which occurred at the second spacecraft apogee. Out-of-plane errors were removed with a maneuver at the common node following the second apogee. Subsequent radar-tracking information indicated the need for an additional adjustment to the altitude differential at spacecraft apogee. This maneuver was performed at perigee near the end of the second revolution. The final midcourse translation was a coelliptic maneuver performed at the third apogee. The purpose of this maneuver was to place the spacecraft orbit at a constant altitude difference below the target-vehicle orbit. The same basic mission plan was also successfully used on Gemini VIII. For the Gemini IX–A mission, the midcourse-maneuver sequence had the additional requirement to more nearly duplicate the Apollo time line and midcourse phase planned for the lunar rendezvous operations. This led to rendezvous in the third spacecraft revolution with a somewhat different maneuver sequence (fig. 3–2). The phase-adjustment maneuver was performed at the first spacecraft apogee. Since the phasing maneuver was based upon a minimal amount of tracking, a second midcourse maneuver designed to achieve phasing, height, and plane requirements was scheduled in the second revolution. The location of this maneuver was

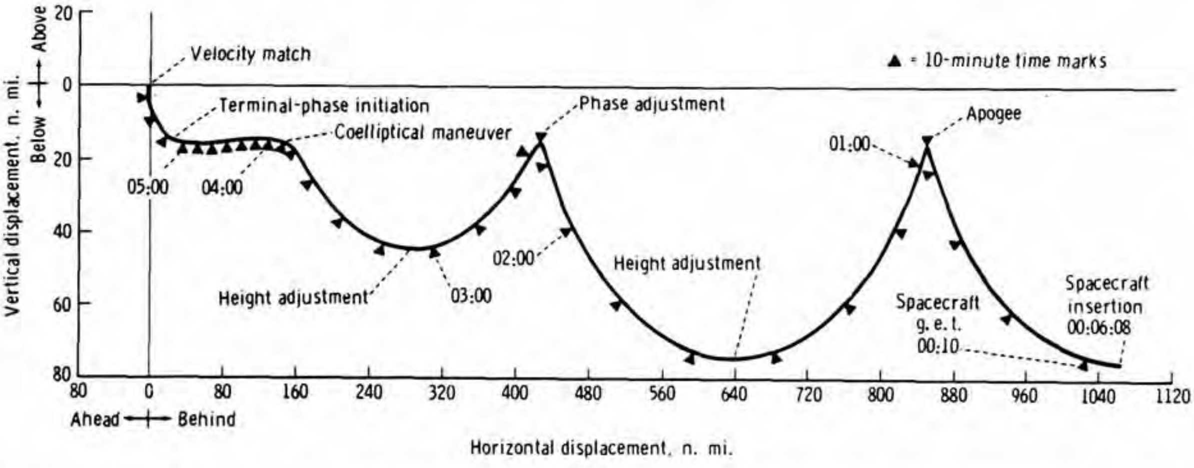

FIGURE 3–1.—Relative trajectory of spacecraft from insertion to three-revolution rendezvous in target-centered curvilinear coordinate system.

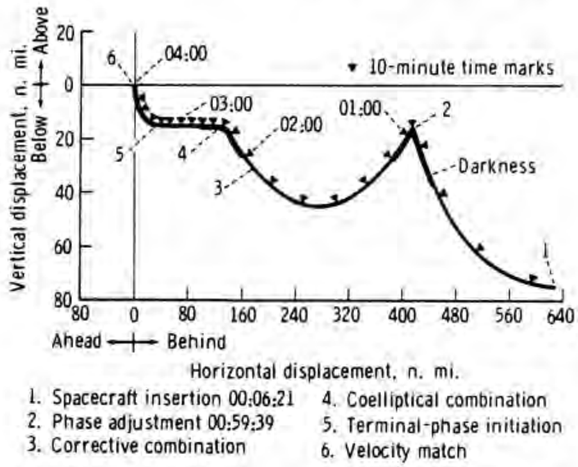

FIGURE 3–2.—Relative trajectory of spacecraft from insertion to two-revolution rendezvous in target-centered curvilinear coordinate system.

TABLE 3–I.—*Dual Rendezvous Planning*

Gemini VIII target-vehicle vector [a]	Gemini X target-vehicle launch time, Greenwich mean time, hr:min:sec	Required apogee, n. mi.
3/19/66	3:40:58	225
3/30/66	3:40:54	245
4/25/66	3:37:30	470
5/16/66	3:37:30	400
6/ 9/66	3:46:30	390
5/24/66	3:41:55	360
6/20/66	3:40:26	420
7/18/66	3:39:46	409

[a] Column shows dates when the passive Gemini VIII target vehicle was in proper position for lift-off of the Gemini X mission to accomplish dual rendezvous.

selected to afford a maximum amount of tracking over the continental U.S. stations. The final maneuver in this sequence provided a constant altitude differential between the two orbits, and also placed the Gemini spacecraft in the plane of the target vehicle.

The initial rendezvous maneuver sequence utilized on Gemini X was identical to that of Gemini VI–A. However, the ground controllers had the additional tasks of evaluating onboard maneuver calculations based upon star measurements and upon the Inertial Guidance System ascent vector; and of giving a go—no-go decision on these solutions based upon retaining acceptable terminal-phase conditions. The flight plan also included a rendezvous between the spacecraft and the passive target vehicle, which had been launched during the Gemini VIII mission and then placed in a higher parking orbit. This plan created an additional complexity, as compared with the earlier rendezvous missions, and necessitated an on-time launch for both the target vehicle and the spacecraft. Table 3–I shows the variation during the 4-month period preceding flight in lift-off time required of the Gemini X target vehicle, as well as the required apogee altitude for dual rendezvous phasing. After the crew completed the initial docking with the Gemini X target vehicle, they initiated midcourse maneuvers (fig. 3–3) to achieve desired conditions for the terminal phase of rendezvous with the passive Gemini VIII target vehicle. The Gemini X target-vehicle propulsion system was used to perform these maneuvers while the spacecraft and target vehicle were docked; the spacecraft propulsion system was used after undocking.

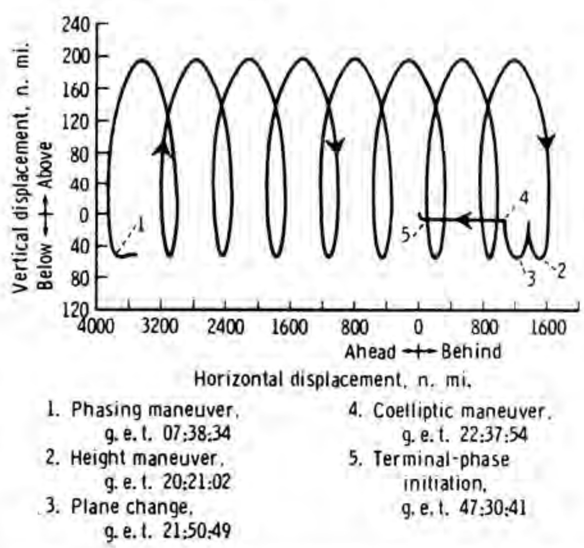

FIGURE 3–3.—Relative trajectory of Gemini X dual rendezvous in target-centered curvilinear coordinate system.

The ground support of the first-orbit rendezvous during the Gemini XI mission (fig. 3–4) was approached in a considerably different manner than during prior rendezvous missions. The only midcourse maneuver scheduled was a plane-change maneuver to account for insertion dispersions. The location of this maneuver was approximately a quarter of a spacecraft revolution after insertion. The major role of the flight controllers for this mission was to evalute the predicted relative conditions at the time of the terminal intercept maneuver, and to give a go—no-go decision for the first-orbit rendezvous. The basis for the go—no-go decision was dependent on the resultant propellant cost for the terminal-phase operations, and on the relative conditions which would preclude the use of onboard backup charts required in the event of degraded systems performance. In addition to providing a go—no-go decision, a contingency maneuver plan was computed in the event that the decision was no-go. This plan was based upon rendezvous in the third revolution.

For the Gemini XII initial rendezvous, the midcourse maneuver sequence was identical to that of the Gemini IX–A mission. The additional complexity involved for this mission included ground evaluation of the onboard-computed plane-change maneuver and the final maneuver to establish the constant altitude differential.

Following the final midcourse maneuver update, the ground provided a backup terminal-phase-initiate maneuver to serve as a comparison between the onboard closed-loop and backup solutions. In addition, supplemental information, such as the variation in altitude differential, was passed to the crew.

Re-Rendezvous Operations

Three re-rendezvous operations were included in the Gemini Program to increase the rendezvous experience. These exercises investigated such factors as variation in lighting and terminal-approach conditions. The equiperiod re-rendezvous of the Gemini IX–A mission was used to study proposed lighting conditions for the dual rendezvous of Gemini X. The second re-rendezvous of Gemini IX–A investigated a terminal approach from ahead and above the target vehicle in support of future Apollo rendezvous operations. The re-rendezvous of Gemini XI was a totally different technique from any previously flown. The spacecraft was given phasing maneuvers from the ground such that no relative phase rate existed between the two vehicles prior to the intercept maneuver. In this configuration, the spacecraft trailed the target vehicle by approximately 25 nautical miles in the same orbit (fig. 3–5). The vehicles remained in this configuration for approximately 12 hours, at which time a ground-computed intercept maneuver was applied, with the final terminal-phase control performed visually by the crew. This technique was flown to compare the propellant cost with that required for long-term, close-range station keeping.

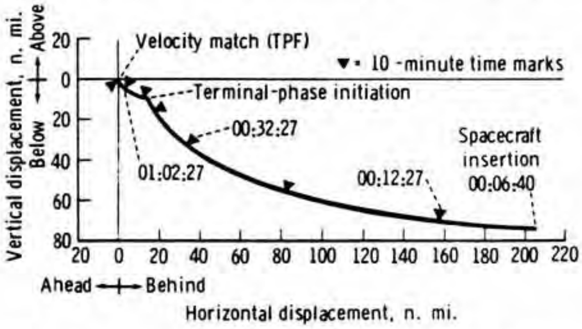

FIGURE 3–4.—Relative trajectory of Gemini XI from insertion to rendezvous in target-vehicle curvilinear coordinate system.

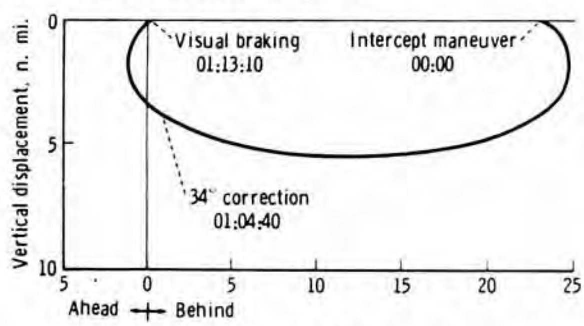

FIGURE 3–5.—Gemini XI stable orbit re-rendezvous.

Flight Results

The effectiveness of the ground-computed midcourse maneuvers can best be evaluated by the propellant required for midcourse maneuvers, and how accurately the conditions for terminal-phase initiation were met. As shown in table 3–II, the lighting conditions obtained were within desired limits; above-nominal midcourse propellant usage was largely due to dispersions in insertion conditions. The variation in altitude differential following the coelliptic maneuver was well within the limits for the use of onboard backup charts on all flights. The ground-computed terminal-phase solutions were consistently in very close agreement with both onboard solutions for all missions.

Gemini Agena Target Vehicle

Prior to spacecraft launch and subsequent rendezvous operations, the Gemini Agena Target Vehicle was monitored and evaluated to insure proper configuration for use as a passive target. Of prime concern, other than total electrical failure, was the verification of insertion into the proper orbit. Any significant error in insertion would require correction by a plane-change maneuver from the target-vehicle propulsion system.

Upon achieving a nominal insertion, complete checkout of vehicle performance and attitude conditioning was accomplished by the target-vehicle flight controllers. This normally consisted of correcting the memory system and configuring the target docking equipment for the rendezvous by real-time commands. The target vehicle was further commanded to an orientation of —90° from the flight path (docking adapter to the north) in order to present a larger target to the spacecraft radar system and to provide a larger target for visual acquisition in sunlight exposure.

From target-vehicle lift-off to spacecraft rendezvous, three major parameters were evaluated to assure a safe target. The propellant-tank differential pressure was of prime concern because a reversal pressure would cause the loss of the target vehicle. The battery temperature was continuously evaluated to predict a rate of change, since the target would be lost if the temperature became excessive. The Attitude Control System pressure was evaluated to assure a nonleak condition which would provide adequate control to preclude vehicle instability and associated unsuitability for docking.

Conclusion

Effective ground support and control has been demonstrated in the successful accomplishment of the rendezvous missions of the Gemini Program. Of key importance in this success was the flexible real-time planning capability which afforded the necessary response to a variety of mission situations.

TABLE 3–II.—*Rendezvous Midcourse Phase Summary*

Gemini mission	Velocity		Variation in terminal-phase initiation time, min [a]	Variation in altitude differential, n. mi.
	Nominal, ft/sec	Actual, ft/sec		
VI–A	117	159	2.1	1.1
VIII	117	150	9.7	1.2
IX–A	126	173	−1.2	.2
X	120	141	−2.2	.9
XI	30	44	.3	—
XII	135	167	5.0	1.3

[a] Positive values indicate late terminal-phase initiations.

4. ONBOARD OPERATIONS FOR RENDEZVOUS

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Introduction

An overall plan for onboard rendezvous operations for the Gemini missions was developed in parallel with the mission plan. The purpose of this plan was to make optimum use of crew time in orbit to maximize the probability of a successful rendezvous. The evolution of the plan began with a preliminary time line of events based upon the known guidance-equipment requirements and upon the estimated crew timing. A preliminary set of flight charts was developed to aid the crew with primary and backup procedures and to establish a backup guidance capability. These charts, which consisted of a few simple graphs and tabulation sheets, enabled the crew to calculate accurate solutions for the terminal maneuvers even with an inoperative guidance-equipment component. As such, the charts significantly contributed to the probability of mission success. Following the development of the charts, an engineering evaluation was conducted on a realistic man-in-the-loop simulation. During this evaluation, the procedures and charts were subjected to the expected equipment errors and trajectory dispersions, and revisions were made as necessary to improve effectiveness. The resulting plan was presented to the flight crew; the charts were evaluated during a period of training on the simulator. The crew spent several weeks training on both the primary procedures and on the various failure modes.

General Rendezvous Operations

The operation of the guidance system for rendezvous was divided into primary, monitoring, and backup procedures. Primary procedures were the crew tasks necessary to define and execute any given maneuver. Monitoring and backup procedures were used to assess the operation of the system and to complete the mission in contingency situations.

Primary Procedures

The spacecraft onboard operations were broadly categorized into insertion corrections, midcourse or catchup-phase corrections, terminal-phase closed-loop corrections, and braking and line-of-sight control. Since these basic operations were common to most missions, each category is described first as it applies to a general rendezvous mission. Then, the rendezvous operations on specific missions are discussed.

Insertion corrections.—An insertion correction based upon onboard navigation information was computed and displayed to the crew by the Insertion Velocity Adjust Routine (IVAR). This correction was designed to achieve the planned apogee altitude and to eliminate the out-of-plane velocity. Although providing a very early opportunity to reduce trajectory dispersions, the correction could possibly include significant navigation errors; therefore, the application of the maneuver was not always advantageous. For missions with relatively long catchup times, it was usually preferable to omit the correction, or to apply only the in-plane component and then use the ground-tracking information that was available later to determine a more accurate correction. Conversely, when an early rendezvous was desired, both components of the correction were applied,

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and a third component was manually computed and applied. This third component was a radial correction based upon a computer readout of downrange travel, and was designed to correct the phasing at first apogee. One of the procedural problems related to the insertion correction was a method for avoiding recontact with the launch vehicle after separation and for applying the Insertion Velocity Adjust Routine maneuver. The problem was resolved by prohibiting a certain band of velocity changes most likely to result in recontact, and by establishing visual contact with the launch vehicle before making retrograde corrections.

Midcourse phase.—The onboard operations required for the catchup phase of a rendezvous mission basically consisted of determining and applying the midcourse maneuvers. Other onboard operations during this phase were routine procedures such as platform alinements and system checks. For most of the rendezvous missions, the midcourse maneuvers were computed by the ground complex and transmitted to the spacecraft. The crew tasks in this case consisted only of achieving the correct attitude and of applying the thrust at the proper time. A typical sequence of catchup maneuvers is shown in figure 4–1.

To demonstrate an onboard navigation capability, the Gemini X mission procedures required the flight crew to compute catchup maneuvers using the onboard orbit determination and prediction capability. The same basic maneuvers were computed as on the earlier four-orbit rendezvous mission, except

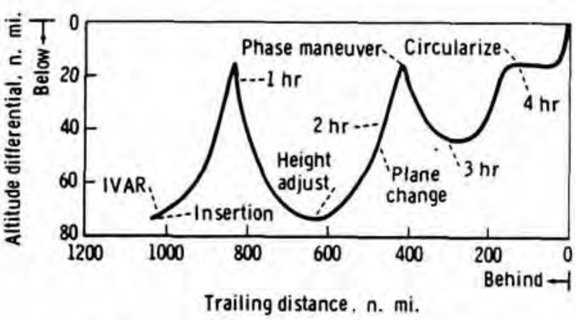

FIGURE 4–1.—Typical midcourse maneuvers.

that the height-adjust maneuver usually performed at second perigee was replaced by an insertion correction. The crew procedures for onboard determination of the midcourse maneuvers involved a sequence of computer and sextant usage. The first maneuver was made at insertion. After this correction, an auxiliary tape memory module containing the mathematical flow for the orbit determination, navigation, and prediction modes of operation was entered into the onboard computer. First, the orbit determination mode was selected and initialized, and a series of star measurements was made and entered into the computer. After processing these data, the computer produced an updated state vector which was used in the orbit prediction mode to predict the spacecraft velocity at the following maneuver point, and the position at the following apogee if no maneuver was made. With the aid of the flight charts, this information was used to predict the desired velocity at the next maneuver point—thus the velocity change. The other maneuvers were determined in a similar manner, and all of the solutions were compared with the corresponding ground computer values. If the differences were within the bounds established before flight, the onboard-determined maneuvers would be applied; if not, the ground-supplied maneuvers would be applied.

Several problems arose in connection with these procedures. For example, a group of star-to-horizon angle measurements from an earlier mission indicated that the apparent altitude of the Earth horizon changed with time, possibly as a result of varying moonlight conditions. These variations were large enough to have a significant effect on the maneuver solutions, and a series of measurements was required to calibrate the horizon. A second problem was the definition of a measurement schedule for orbit determination. The timing, as well as the type and the direction, of the measurements had to be established. Studies revealed that the measurements should be spaced over two darkness periods, and that a variety of directions should be used. The selected schedule consisted of four in-plane and two out-of-plane measurements, but the crew timing requirements and the inaccuracy of the resulting out-of-plane orbit determination led to the decision to use dummy out-of-plane measurements. The effect was that the out-of-plane component of the vector was not updated.

Terminal phase.—The terminal-phase rendezvous operations employed the onboard computer in conjunction with the inertial platform and radar. In the rendezvous mode, the computer gathered radar and platform data, and operated on the data in the sequence outlined in figure 4–2. Initially, data were sampled and stored at a crew-optional fixed-time interval; both 60 and 100 seconds were used. After sufficient data had been stored, an estimate of the total velocity change required for a two-impulse rendezvous transfer was computed and displayed to the crew. The estimate was updated with each succeeding data point for use as an aid in determining the best point to initiate the transfer. The crew initiated the maneuver sequence by depressing the START COMP button on the instrument panel. At this time, the velocity change (in components along the three body axes) for terminal-phase initiation was displayed to the crew, along with the proper attitude for application of the thrust. The maneuver was achieved when the command pilot depressed the maneuver controller until the displayed velocity change counted down to zero. Since equipment and application errors could produce significant dispersions in the resulting transfer trajectory, vernier corrections to the transfer were computed at regular intervals and displayed to the crew. The time of the transfer and the number of vernier corrections were mission-planning options. Generally, based on a trajectory that would result in an intercept in 130° of orbit travel ($\omega t = 130°$), a transfer time of about 30 minutes was selected with two vernier corrections.

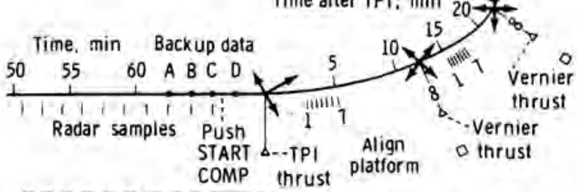

FIGURE 4–2.—Terminal-phase computer sequence.

Braking and line-of-sight control phase.—The braking and line-of-sight control phase which followed the final vernier correction was manually controlled. Simply stated, line-of-sight rate control was achieved by determining the direction of the rate and thrusting normal to the line of sight to null this rate. The direction of the motion could be determined by either of two methods. The first method was to fix the attitude of the target vehicle with respect to a body-fixed reticle. When movement was apparent, thrust was applied radially in the direction of motion. The second method, which could be employed when stars were not visible, made use of the Flight Director Attitude Indicators in an inertial mode. After the command pilot had boresighted on the target, the pilot entered a logic choice into the computer which centered the flight director indicator needles and subsequently deflected them proportionately to spacecraft inertial-attitude changes. The command pilot was then able to hold the attitude that would keep the needles centered, and to observe the target drift with the optical sight. Nulling the target motion was then accomplished in the same manner as the first method.

Monitoring and Backup Techniques

One important crew function during rendezvous was to monitor the performance of the guidance system to assure that the translational maneuvers were accurately computed and applied. Monitoring can be defined as the assessment of guidance-systems operation to the extent necessary for detection and identification of performance degradation in sufficient time for corrective action. During rendezvous, monitoring was accomplished by sampling basic flight data at specified points in the trajectory, and by calculating with the aid of charts and graphs a solution to each maneuver for comparison with the closed-loop and/or ground solution.

Backup charts.—The data used for monitoring and backup are shown in table 4–I. The use of sensor information varied, depending upon the maneuver to be calculated. A typical case was illustrated by the terminal-phase initiation procedure. The spacecraft attitude was maintained in zero roll and boresighted on the target using the optical sight. This alined the X-axis to the target line of sight. The radar and platform data could then be used to calculate velocity increments ΔV along and normal to the target line of sight. The ΔV along the line of sight was obtained in terms of relative range rate $\dot{R}$ by the equation

$$\Delta\dot{R} = \dot{R}_{REQ} - \dot{R}_{ACT}$$

where

$\Delta\dot{R}$ was the increment in velocity along the target line of sight required to transfer to the desired intercept trajectory

$\dot{R}_{REQ}$ was the range rate of the desired trajectory at the point of data sampling immediately prior to terminal-phase initiation, and was defined by target elevation angle and range for the type trajectory desired

$\dot{R}_{ACT}$ was the actual range rate at the point of data sampling immediately prior to terminal-phase initiation

A typical terminal-phase trajectory is one which intercepts in 130° of target orbit travel. Figure 4–3 shows the relationship of $\dot{R}_{REQ}$ at terminal-phase initiation with pitch angle θ and range for this transfer. The relationship is nearly independent of the target orbit; thus, figure 4–3 is valid for altitudes within 20 nautical miles of the nominal.

TABLE 4–I.—*Monitoring Data*

Data	Units	Sensor	Display	
			Prime	Backup
Range	0.01 n. mi.	Radar	Manual data unit	Analog gage
Range rate	ft/sec	Radar	Manual data unit	Analog gage
Pitch angle	0.1°	Inertial measuring unit.	Manual data unit	Flight director attitude indicator, stars
Yaw angle	0.1°	Inertial measuring unit.	Manual data unit	Flight director attitude indicator, stars
Roll angle	0.1°	Inertial measuring unit.	Manual data unit	Flight director attitude indicator, stars
Target boresight	0.1°	Optical sight	Visual	—
		Radar	—	Flight director indicators

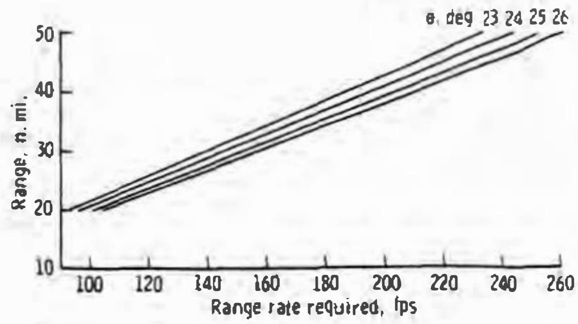

FIGURE 4-3.—Terminal-phase initiation range rate.

The ΔV in-plane, normal to the line-of-sight increment in velocity, defined in terms of line-of-sight angular rate $\dot{\theta}$ and range R by the equation

$$\Delta V_N = (\dot{\theta}_{REQ} - \dot{\theta}_{ACT}) R$$

where

ΔV_N was the in-plane, normal to the line-of-sight increment in velocity required to transfer to the desired intercept trajectory

$\dot{\theta}_{REQ}$ was the in-plane line-of-sight angular rate of the desired trajectory at the point of data sampling immediately prior to terminal-phase initiation, and was defined by target elevation for the trajectory desired

$\dot{\theta}_{ACT}$ was the actual line-of-sight rate at the data sampling point immediately prior to terminal-phase initiation

R was range to the target at the measurement point

Since $\dot{\theta}$ could not be measured directly with sufficient accuracy, an increment in θ over a measured time interval was used.

$$\Delta V_N = \left(\dot{\theta}_{REQ} - \frac{\theta_2 - \theta_1}{\Delta t_{12}} \right) R$$

where

θ_1 and θ_2 were target elevation at the beginning and end of the measuring interval, respectively

Δt_{12} was the measurement time interval

For use in flight, the equations for $\Delta \dot{R}$ and ΔV_N were mechanized graphically (fig. 4-4). This chart was part of the onboard data package for Gemini IX-A. The technique used throughout the Gemini Program was to initiate terminal-phase initiation at a reference target elevation angle. This provided a standardized terminal phase in terms of elevation and approach conditions. Crew procedures approaching terminal-phase initiation were to track the target and observe the increase in elevation angle. Pertinent data were recorded on logging sheets at each interval as samples were taken by the computer for the computation of the closed-loop solution for terminal-phase initiation. The reference elevation angle which keyed the terminal-phase initiation sequence was 21.4° for most rendezvous. As the elevation angle approached 21.4°, certain samples were utilized for the terminal-phase initiation monitoring and backup solutions. The significant data points were labeled A, B, C, and D, and are defined as follows:

A = Data point immediately prior to 21.4° target elevation

B = First data point after 21.4°; first used to calculate the backup solution

C = Next data point after B; used to initiate the closed-loop sequence for terminal-phase initiation

D = Next point after C; provided the final data for the backup solutions for terminal-phase initiation

Figure 4-4 illustrates the sequence for obtaining a backup solution to terminal-phase initiation. Range and pitch angles were recorded each 100 seconds until θ exceeded 21.4°. This angle was designated point B and recorded. After the next sampling point C, the START COMP button was depressed to initiate the closed-loop sequence for terminal-phase initiation. Range, range rate, and pitch angle for the second point beyond B, point D, completed the information needed to calculate the backup solution. The procedures for obtaining the backup solution are as follows:

(1) Boresight on target

(2) Monitor θ, R, and $\dot{R}$ every minute

(3) When $\theta \geq 21.4°$, record data for point B on terminal-phase initiation chart

(4) Push START COMP button after next data point

(5) Record data at point D

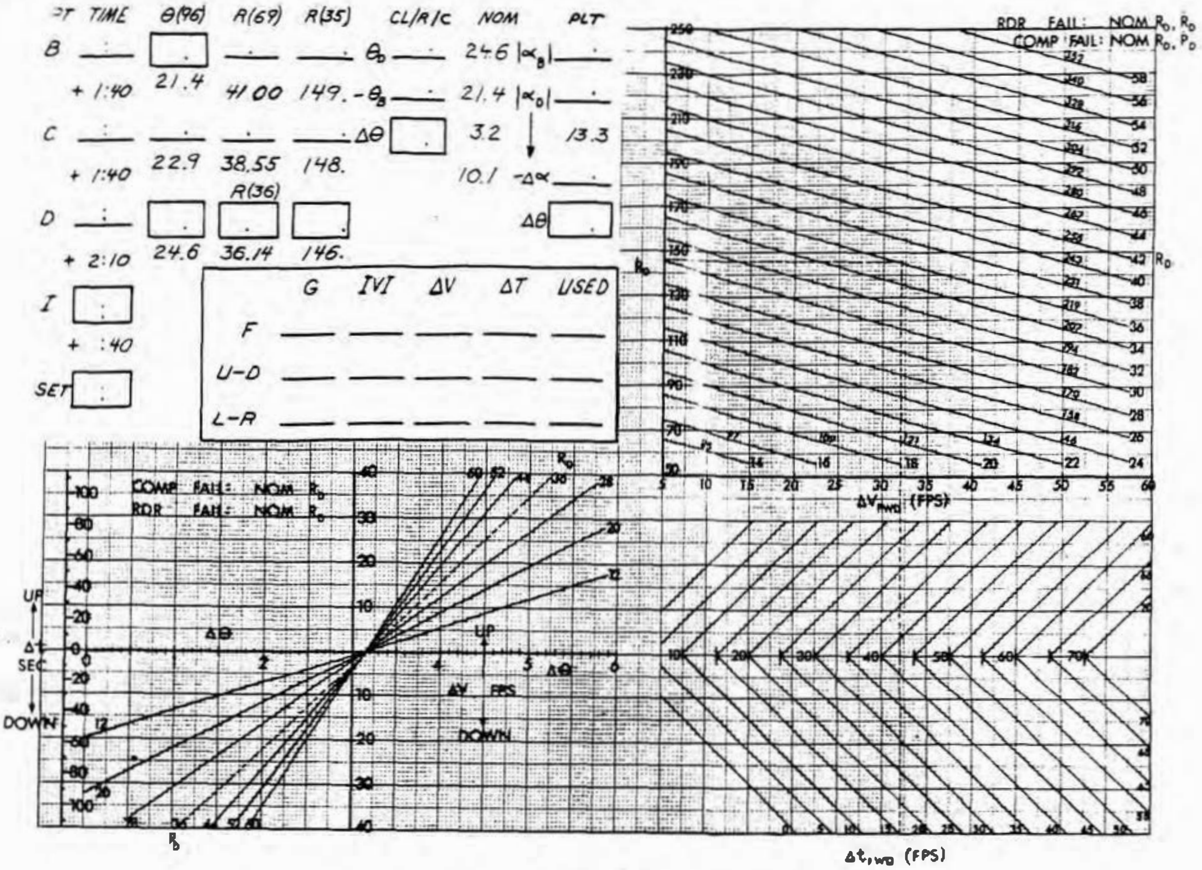

FIGURE 4-4.—Terminal-phase initiation.

(6) Enter terminal-phase initiation chart to calculate ΔR, ΔV_x, and terminal-phase initiation time

(7) Compare ΔR and ΔV_x with closed loop and Manned Space Flight Network

A similar technique was used for midcourse corrections except that measurements were triggered on time after terminal-phase initiation rather than on pitch angle.

Failure modes.—Throughout the Gemini Program, manual techniques were utilized wherever practical to maximize the probability of mission success. Thus, the crew was prepared at all times to continue the mission with degraded or failed systems components. This required frequent reference to monitoring data and substitution of alternate sources when failures occurred. The different situations that could exist for all possible combinations of partial and complete failures were too numerous to permit specific training for each. Therefore, procedures were developed only for total failure of each of the three major guidance system components: radar, computer, and platform. Partial failures were then handled by utilizing whatever valid data were available from the degraded component.

For total failure of any guidance component, the closed-loop solution would no longer be available. In this case, it was necessary to rely on the ground or backup solution obtained by alternate methods. For all failures, procedures were designed to obtain a maneuver solution in components along and normal to the target line of sight. Table 4–II summarizes the sensors used for the significant failures. For radar failures, a redundant source of range information was not avail-

TABLE 4–II.—*Failure Modes*

Failure	Forward/aft, ΔV source	Up/down, ΔV source
None	Closed-loop guidance	Closed-loop guidance
Radar	Manned Space Flight Network or nominal	Manual data unit, θ, $\Delta\theta$
Computer	Analog gage, R, $\dot{R}$	Flight director attitude indicator, θ, $\Delta\theta$
Inertial measuring unit	Manual data unit, R, $\dot{R}$	Sextant nominal, θ, stars $\Delta\theta$

able and only up/down maneuvers could usually be calculated on board. One exception was the first-orbit rendezvous on Gemini XI where a terminal-phase initiation correction along the line of sight could be based on the insertion vector obtainable from the Inertial Guidance System. The computer failure case would not cause loss of information in either axis, but would result in less accurate maneuvers because the readout on the Flight Director Attitude Indicator and radar analog gage was less accurate than from the computer readout.

In training, the platform failure proved the most difficult to resolve because accurate attitude angles could not be obtained late in the terminal phase. Fortunately, this failure was not encountered in flight. On most missions subsequent to Gemini VI–A, a hand-held sextant was provided for determining time of arrival at terminal-phase initiation in case the Inertial Measuring Unit had failed. The time could be determined by noting the time when the angle between the target and horizon lines of sight corresponded to the planned pitch angle at point B. For the platform failure case, the up/down velocity increment for terminal-phase initiation and vernier corrections could be calculated from the change of the target line-of-sight angle as measured against the star background. At the start of an incremental angle measuring interval, the reticle pattern of the optical sight would be fixed against the star background with the target at the top of the reticle. During the measuring interval, the pilot would attempt to maintain the attitude relative to the stars. At the end of the measurement time, noting the position of the target on the reticle provided the delta angle needed for calculating the up/down incremental velocity.

Mission Results

During the Gemini Program, a total of 10 rendezvous was accomplished (table 4–III), providing as broad a spectrum of terminal-phase conditions as possible. On several missions more than one rendezvous was performed. This allowed a rapid development of the rendezvous technology, including problems, tradeoffs, and solutions. The guidance and navigation system proved versatile, as rendezvous plans were shuffled within weeks of launch, and as lessons learned on each mission were incorporated on the next. Since the rendezvous plans and procedures were functions of mission objectives, each type of rendezvous and its characteristics are treated separately in the following paragraphs.

Rendezvous in the Second, Third, and Fourth Orbits

The terminal phase of many of the Gemini mission rendezvous followed a set pattern:

(1) Approach to terminal-phase initiation through a nominally circular catchup orbit, below and behind the target

(2) Time of terminal-phase initiation determined approximately by phasing maneuvers prior to the circular catchup orbit, then fixed precisely by observation of target elevation above the local horizontal

(3) The intercept orbit traveled 130° central angle not including braking

TABLE 4–III.—*Gemini Rendezvous Summary*

Mission	Target	Approach	Separation altitude, n. mi.	Orbit travel, deg
VI–A	Gemini VII spacecraft	Below	15	130
VIII	Gemini VIII target vehicle	Below	15	130
IX–A: Initial rendezvous	Augmented target docking adapter.	Below	12.5	130
No. 1 re–rendezvous		Equiperiod	0	80
No. 2 re–rendezvous		Above	7.5	130
X: Initial rendezvous	Gemini X target vehicle	Below	15	130
Re–rendezvous	Gemini VIII target vehicle	Below	5	80
XI: Initial rendezvous	Gemini XI target vehicle	Below	10	120
Re–rendezvous		Stable orbit	0	292
XII	Gemini XII target vehicle	Below	10	130

(4) Two vernier corrections at fixed times after terminal-phase initiation
(5) An approach from below and slightly ahead of the target through a series of braking maneuvers at fixed ranges along an inertially fixed line

The major variables available for mission planning purposes can be summarized as follows:

(1) Time of terminal-phase initiation
(2) Target elevation angle at terminal-phase initiation
(3) Orbit travel between terminal-phase initiation and terminal-phase finalization
(4) Time between vernier corrections
(5) Braking schedule
(6) Altitude differential between target and spacecraft

The time of terminal-phase initiation was grossly controlled by lift-off time and by phasing maneuvers prior to the circular catchup orbit, with phasing maneuvers determined on the ground. Primary considerations in establishing a time for the terminal-phase initiation were number of phasing orbits desired and sunlight conditions. Three phasing orbits were required for the early flights of Gemini VI–A and VIII. As ground and onboard operations evolved, the number was decreased to two for the later flights, Gemini IX–A and XII. A further decrease in total time to rendezvous required modification of terminal-phase procedures on Gemini XI. Terminal-phase lighting tradeoffs centered around the following:

(1) Target visibility at terminal-phase initiation in reflected sunlight
(2) Availability of stars during braking phase to aid line-of-sight control
(3) Approach to docking in sunlight

These considerations placed the terminal-phase initiation time near sunset with midcourse corrections and line-of-sight control during the night period.

Figure 4–5 depicts the lighting conditions for the typical rendezvous from below the target vehicle. Elevation angles of the target

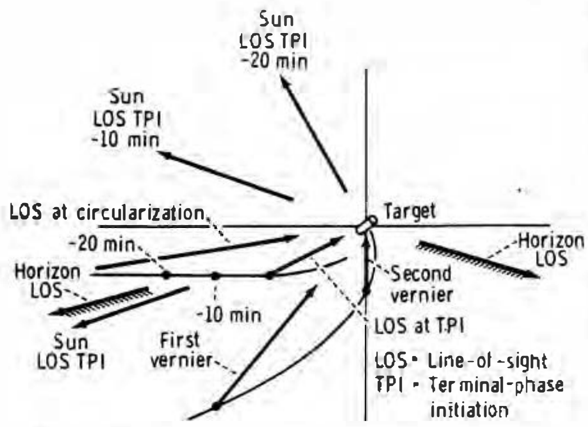

FIGURE 4–5.—Terminal-phase lighting conditions.

vehicle and Sun are shown. With the longitudinal axis of the target vehicle controlled to 90° out of plane, the target vehicle was easily visible in reflected sunlight during the time period when the critical measurements for terminal-phase initiation were made. Thus, the flashing acquisition lights were not relied upon for visual sighting at the longer ranges. As the terminal phase progressed, the Sun elevation and the target line of sight rotated counterclockwise (fig. 4–5). After sunset, motion of the target vehicle in relation to the stars provided confidence in the trajectory status. After the last vernier correction, the star field was also useful for maintaining the collision course. With the terminal-phase initiation near sunset, the spacecraft would pass the last braking gate at a range of 3000 feet at sunrise. The target in perspective, indicated approach angle and closing velocity.

Careful selection of the orbital travel from terminal-phase initiation to terminal-phase finalization and the target elevation at terminal-phase initiation provided an approach that had a line-of-sight angular rate of nearly zero and terminal-phase initiation maneuver along the line of sight. The small line-of-sight drift rate after the last vernier correction assisted the crew in maintaining a simple and efficient collision course which helped to minimize propellant usage. The spacecraft roll axis was boresighted on the target throughout the terminal phase. Selecting a trajectory for which the terminal-phase initiation angle coincided with the target elevation angle allowed the maneuver to be performed nominally along the roll axis with no attitude deviation. Dispersions in the catchup orbit and guidance system errors appeared at terminal-phase initiation as maneuver components normal to the line of sight, and as deviations from the planned forward impulse. Table 4–IV summarizes the terminal-phase initiation and the midcourse maneuvers for

TABLE 4–IV.—*Terminal-Phase Maneuver Summary*

Mission	Closed-loop guidance and applied maneuvers[a]									
	Terminal-phase initiation, fps				1st vernier, fps			2d vernier, fps		
	Nominal, forward	Actual, forward	Up, down	Right, left	Forward, aft	Up, down	Right, left	Forward, aft	Up, down	Right, left
VI–A	32	31	4U	1R	7F	7U	5L	4F	3U	6R
VIII	32	25	3U	8R	12F	6U	1R	4F	7U	3R
IX–A	27	(27)	(1U)	(2R)	2A	2U	3R	3F	2D	0R
		26	8U	4R						
X	32	41	(0U)	(0L)	15A	(14D)	1R	(0F)	25D	5R
			1U	16L		22D		1F		
XII	22	(22)	(0U)	(0R)	(0F)	(2U)	(0R)	(5A)	(1D)	(0R)

[a] Parentheses indicate applied maneuvers when different from closed-loop solutions.

the Gemini IV, VIII, IX–A, X, and XII missions. The times of vernier corrections were selected to be compatible with crew loading and the anticipated accuracy of the guidance system. Vernier corrections 12 and 24 minutes after terminal-phase initiation allowed sufficient time for crew activities, such as system monitoring and platform alinement where necessary, but were close enough to prevent appreciable trajectory divergence.

The relatively low deceleration capability of the Gemini spacecraft (approximately 1 ft/sec^2) dictated that closing velocity be reduced in several stages to enable the crew to devote proper attention to line-of-sight control. Early training simulations indicated that braking to a maximum closing rate of 40 ft/sec at a range of 2.5 nautical miles, and then down to 5 to 10 ft/sec at a range of 0.5 nautical mile, represented a simple and efficient schedule.

The separation altitude selection was a tradeoff between total propellant and sensitivity of time of arrival at terminal-phase initiation to dispersions in the catchup orbit. As previously discussed, there were advantages to certain sunlighting conditions during the terminal phase; and for a given error in the catchup orbit, the dispersion in arrival time decreased as separation altitude increased. However, propellant requirements for the terminal phase increased in proportion to differential altitude. (An altitude differential of 15 nautical miles was selected for Gemini VI–A.) As knowledge of lighting conditions was gained, and as the capability for ground tracking evolved, the altitude differential was varied (table 4–III).

Rendezvous in the First Orbit

The first-orbit rendezvous accomplished during the Gemini XI mission was more demanding of onboard operations than previous rendezvous missions. The previous missions utilized several orbits of ground tracking and computation to eliminate the effects of insertion dispersions on the terminal-approach trajectory. Because of the very short time available for the first-orbit rendezvous mission, the multiorbit midcourse corrections and circular catchup orbit could not be used. As a result, the flight plan included onboard operations capable of absorbing the expected insertion dispersions in a relatively short time. The trajectory plan selected for the first-orbit rendezvous had a terminal approach similar to the approach employed on the coelliptical rendezvous missions. However, it appeared that insertion dispersions would radically affect this approach as shown in figure 4–6. Terminal-phase initiation occurred near the first spacecraft apogee with a 120° central angle of transfer.

In providing a capability for absorbing the insertion dispersions, several procedural methods were required which were not employed on previous missions. At insertion, the horizontal and out-of-plane velocity changes were planned as usual. These corrections, however, did not remove the trailing displacement error at first spacecraft apogee resulting from downrange and flight-path angle errors at insertion. This error could have had a serious effect on the terminal-approach trajectory; to reduce the error, the pilot read (from the computer) the navigated downrange angle traveled at insertion. From this angle, a required value of altitude rate was determined and compared with the

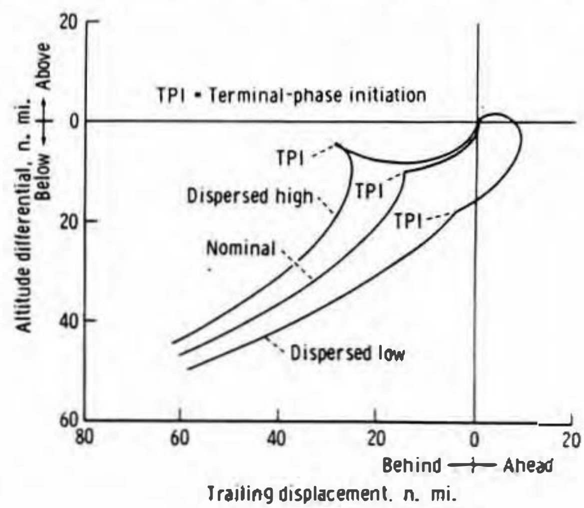

FIGURE 4–6.—First-orbit rendezvous trajectory.

actual altitude rate read from the computer. The velocity difference was applied along the local vertical to achieve an altitude rate resulting in the desired trailing displacement at the terminal-phase initiation point. Although this correction required split-second timing on the part of the crew, it was very effective.

The second onboard-computed maneuver was an out-of-plane correction to be performed 90° after insertion. Since the maneuver at insertion was to eliminate the out-of-plane velocity at that point, the node occurred 90° of orbit travel later. By observing the out-of-plane displacement at insertion, the pilot computed the required maneuver. At the expected time of the node, the correction was applied.

Although the primary procedures for the terminal phase of the first-orbit rendezvous were similar to the procedures for previous rendezvous missions, the effect on the larger terminal-phase dispersions had a significant impact on the design of the backup and the monitoring procedures. The backup procedures utilized measurements of range and line-of-sight angle changes over a fixed time interval. These measurements were used with flight charts to determine the velocity changes and the relative position of the spacecraft at the time of the terminal-phase initiation maneuver. Gemini XI was the first mission to utilize a backup capability for an out-of-plane correction at terminal-phase initiation. The correction reduced the dispersions caused by navigation errors during the earlier corrections.

Two vernier corrections were scheduled at 12-minute intervals during the terminal transfer. The backup computation of these maneuvers was significantly different than for previous missions because the variation from the planned position of the spacecraft at terminal-phase initiation was taken into account. For example, with a radar failure, the earlier charts assumed a planned range in computing the correction instead of using a predicted range based upon the actual spacecraft position at terminal-phase initiation. The use of predicted values provided better accuracy for large dispersions. Table 4–V is a summary of the maneuvers for the first-orbit rendezvous.

Rendezvous From Above the Target Vehicle

A re-rendezvous was conducted on the Gemini IX–A mission to simulate the trajectory of a Lunar Module following abort during powered descent. The trajectory was similar to that utilized on the fourth-orbit rendezvous mission except that the spacecraft approached the target from ahead and above. The procedures for rendezvous from above were very similar to the procedures for a fourth-orbit rendezvous; the only significant differences were in the backup measurements used in the event of a platform failure. Since the spacecraft approached the target from above, there was no star background during the terminal phase. As a result, the hand-held sextant would have been used to make angle measurements with respect to the Earth horizon. These measurements, like those with respect to the star background, required visual acquisition of the target.

A significant lesson was learned from the rendezvous from above; the terminal-phase

TABLE 4–V.—*Gemini XI Rendezvous Maneuvers*

Insertion Velocity Adjust Routine ΔV, fps	Plane change ΔV, fps	Terminal-phase initiation ΔV, fps	1st vernier ΔV, fps	2d vernier ΔV, fps
39 forward	0	140 forward	1 forward	1 forward
5 down	0	27 down	4 up	3 up
1 left	3 left	5 left	4 right	11 right

lighting conditions were more critical than for rendezvous from below. During the early Gemini IX–A mission planning, it was decided that terminal-phase initiation should occur after sunset so that the flashing lights on the target vehicle could be used for visually acquiring the vehicle against the dark Earth background. It was believed that sunset was preferable to an early morning terminal-phase initiation, with acquisition using reflected sunlight (over-the-shoulder lighting) because of the bright Earth background. However, during the Gemini IX–A flight, the nose shroud on the target vehicle (Augmented Target Docking Adapter) did not completely separate, and it was believed that the acquisition lights located in the shroud region might not be visible. The time of terminal-phase initiation was then changed from after darkness to early morning to permit reflected light viewing. Actually, the target was not visible at long range against the bright Earth background, and could not be tracked visually until the range had decreased to 3 nautical miles. If the radar had failed during this exercise, terminal-phase corrections would not have been possible. Furthermore, the rapidly moving terrain background made control of the line of sight more difficult than with a star field or even with a dark Earth. This experience demonstrated the importance of terminal-phase lighting, and pointed out the value of the flashing acquisition lights as a backup to the radar for target tracking. A summary of the terminal-phase maneuvers for the rendezvous from above is shown in table 4-VI.

TABLE 4–VI.—*Terminal-Phase Maneuvers for Rendezvous from Above*

Terminal-phase initiation ΔV, fps	1st vernier ΔV, fps	2d vernier ΔV, fps
19 forward	4 aft	2 forward
4 down	1 up	10 down
2 left	5 left	7 right

Rendezvous With a Passive Target

After the initial rendezvous on Gemini X, an exercise was undertaken to intercept the passive target vehicle that had been in orbit since the Gemini VIII mission. This rendezvous with a completely passive target presented several unique problems, and was more demanding of the crew than any other terminal phase. For the exercise, there was no closed-loop guidance and no radar or acquisition lights; the terminal-phase maneuvers had to be based on backup charts and observation of the target in reflected sunlight. Approximately 27 minutes of favorable lighting time were available in each orbit (from about spacecraft noon until sunset), and the entire terminal phase, including arrival dispersions, braking, and stabilizing position for formation flight through the night period, had to take place within about 108° of orbit travel. Position was maintained after darkness using the docking light on the spacecraft as a source of illumination. The light had a cone angle of about 6° and was effective up to a distance of 300 feet. The short period of visibility indicated that orbit travel between the initiation and the finalization of the terminal phase would have to be reduced considerably from the 130° used on previous rendezvous. An orbit travel of 80° and a differential altitude of 7 nautical miles were selected. The terminal-phase trajectory is shown in figure 4–7. This combination had several advantages in addition to a

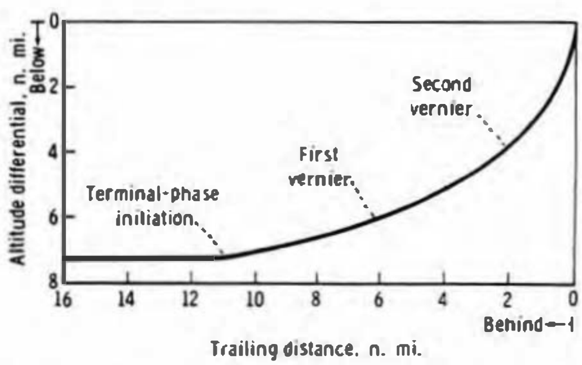

FIGURE 4–7.—Passive target rendezvous trajectory.

short terminal phase. The 80° orbit travel intercept was a relatively high-energy transfer trajectory and, therefore, was less sensitive to initial-condition dispersions and errors in maneuvers. This was particularly significant because no vernier corrections could be calculated along the line of sight without radar information. Second, the reduced differential altitude assisted visual acquisition and, combined with the 80° terminal phase, resulted in closing rates about the same level as the 130° intercept with 15-nautical-mile separation. Thus, similar braking schedules could be used on both rendezvous planned for the mission. The time factor was extremely critical during the braking maneuver; at sunset, all visual contact would suddenly be lost beyond the range of the docking light. Because of the time-critical nature of the exercise, the flight charts included the capability to perform terminal-phase initiation for a range of elevation angles covering a time period of 10 minutes on either side of the nominal. The plan was based upon the nominal elevation angle being used if terminal-phase initiation occurred between visual acquisition and 25 minutes before sunset. A solution was sent from the ground in case visual acquisition occurred too late for an onboard solution.

Stable Orbit Rendezvous

During the Gemini XI flight, a small posigrade separation maneuver was made, followed later by a retrograde maneuver of the same magnitude. The purpose of these ground-computed maneuvers was to establish a trailing position about 25 nautical miles behind the target vehicle and in the same orbit. This location enabled the crew to perform experiments and to sleep while maintaining a position for a simple, economical re-rendezvous. Since the re-rendezvous was initiated from a point in equilibrium relative to the target, the plan was called the Stable Orbit Plan. The maneuver to transfer from the stable orbit to an intercept trajectory was sent from the ground, and was based on the ground track of the spacecraft during the crew sleep period. A terminal-phase trajectory covering 292° was selected, resulting in an elevation time history identical to the familiar 130° transfer. Thus, the backup charts from a previous mission could be used for trajectory monitoring. The radar was not operative during this exercise; therefore, onboard corrections along the line of sight were not possible. However, an up/down vernier correction of zero was calculated, which agreed with the up/down component of the ground solution. The ground-computed maneuver was applied, and braking was accomplished while tracking the target vehicle in reflected sunlight.

Conclusions

The Gemini experience has led to a number of significant conclusions with respect to onboard rendezvous operations.

(1) The extensive participation of the flight crew in rendezvous operations is feasible. They are capable of directing the primary operations of the guidance system and of performing certain phases of the mission without the guidance system. In addition, they can detect and identify system malfunctions and take action to assure the success of the mission.

(2) The crew can monitor the performance of the guidance and navigation system, and determine and accomplish all rendezvous maneuvers with the following basic flight information: (*a*) range to the target, (*b*) range rate, (*c*) body-attitude angles measured from horizontal in-plane references, and (*d*) means for tracking the target (visual or radar).

(3) Flight charts can be developed which provide the crew with the ability to compute solutions for the terminal maneuvers in spite of an inoperative guidance-equipment component. These charts can be made simple to use and can provide accuracies comparable to the primary system.

(4) The onboard operations can be simplified by the proper selection of approach tra-

jectories and lighting conditions. A terminal approach is desirable, which is insensitive to trajectory dispersions and equipment errors. The lighting conditions determine the visibility of the target vehicle and the star background, thus affecting backup procedures.

(5) Visibility through the spacecraft window is an important consideration in terminal-phase rendezvous operations. Visual tracking of the target is a backup to the radar, and the star background is a valuable aid for maintaining a collision course in the braking phase.

(6) A comprehensive program of procedural planning, evaluation, and training is necessary to the success of the mission. Man-in-the-loop simulation is an important part of crew training.

5. OPERATIONAL CHARACTERISTICS OF THE DOCKED CONFIGURATION

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Introduction

In addition to a successful rendezvous between the Gemini spacecraft and the target vehicle, one of the primary objectives of the Gemini Program was to accomplish a docking maneuver to join the two vehicles as a single spacecraft configuration. The next objective was to evaluate the characteristics of the control system on each vehicle in controlling the combined vehicle. A further goal was the use of the Primary Propulsion System of the target vehicle to enlarge the manned spacecraft maneuvering capability. These objectives were all determined feasible, and this paper will describe the implementation of the plan and the achievement of the successful results.

Development of the Docking System

The initial effort in the development of the Gemini docking system was the evaluation of the numerous classical concepts and also of the designs generated during the various studies (fig. 5–1). Each concept raised new questions which had to be studied and resolved. Should the vehicles come together on a collision course or a noncollision course? Should the front end or aft end of the spacecraft be joined to the target vehicle? What differential velocities, mismatch angles, and distances should be considered? How could structural continuity, capable of withstanding orbital maneuvering dynamics, be achieved? How should the propulsion system on the target vehicle be controlled? How could positive separation of the spacecraft from the target vehicle be guaranteed? How could remotely actuated structural attachments be provided on the spacecraft without disturbing the reentry heat-protection configuration?

By systematic evaluation, it was concluded that the docking maneuver must be made with the spacecraft facing the target vehicle, so that the flight crew could adequately control the differential impact velocities and attitudes. This was not the best configuration for orbital maneuvering because of the backward acceleration of the crew, and because the structural arrangement was stress limited in the middle. However, these consid-

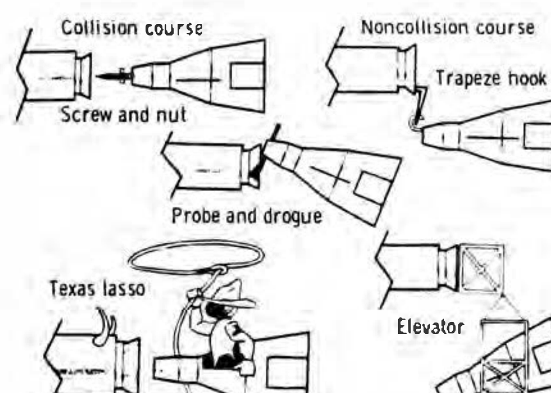

FIGURE 5–1.—Gemini docking concepts.

erations were secondary when compared with the advantage gained by providing a full view of the target vehicle prior to and after docking. With this advantage, impact velocities and attitudes became reasonable values and were determined through simulation exercises. Also, implementation of all target-vehicle control and status display and electrical disconnects was simplified; however, the structural mechanical attachment was somewhat more complicated because of limited bending stiffness.

The evolution from concept to design and the analysis of results from further simulations resulted in the following design criteria: closing velocity of 1.5 ft/sec, angular misalinement of 10°, and centerline displacement of 1 foot with the requirement for multiple docking capability.

Target Docking Adapter

A general arrangement of the selected configuration is shown in figure 5–2. The selected collision-course maneuver was similar to a jet pilot's experience in refueling operations, was the simplest design approach, and was acceptable from a control and safety standpoint. For similar reasons, the probe and drogue design was chosen and a docking bar was installed to provide the indexing

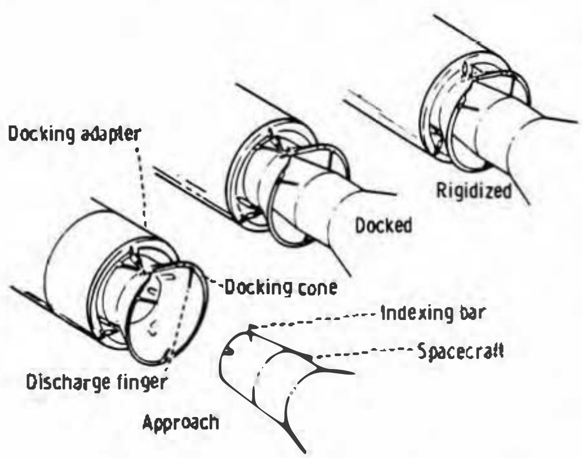

FIGURE 5–2.—Docking and rigidizing sequence.

feature. The electrical and the primary mechanical power devices were installed on the target vehicle because this vehicle was less weight critical than was the spacecraft.

The prime contractor for the spacecraft was selected to manufacture the docking adapter to be mounted on the target vehicle. An interface plane was chosen so that the adapter contained all equipment directly associated with docking. Only electric power, telemetry data, and command system signals crossed the interface. A simple butt joint, consisting of mating skin-former angles and tension bolts, provided easy attachment of the docking adapter to the target vehicle.

The final docking approach (fig. 5–2) was entirely visual, with the target vehicle powered up and stabilized. Visual cues were provided to indicate the status of the target vehicle for nighttime as well as daylight docking. Docking was accomplished when three latches in the target-vehicle docking cone engaged corresponding fittings on the spacecraft. Engagement of the latches completed a circuit that automatically secured the cone against the rigid structure; this was the rigidized mode. Undocking was the reverse of this procedure, with provisions for emergency undocking furnished by pyrotechnic devices which would dislodge the three spacecraft fittings.

Figure 5–3 shows some of the major components of the Target Docking Adapter. Seven dampers were clustered at three locations and damped relative motion in all three axes; they also returned the cone to the ready configuration. A small electric motor provided the power to retract the cone by means of a torsion cable drive to three-gear motors which operated the overcenter bellcrank and linkage devices. Final motion caused the latches to close down on the spacecraft fittings, effecting a rigid connection. Undocking was simply a reversal of this sequence. Some of the other major components were the target-vehicle status display indicators, acquisition lights, and spiral and dipole antennas.

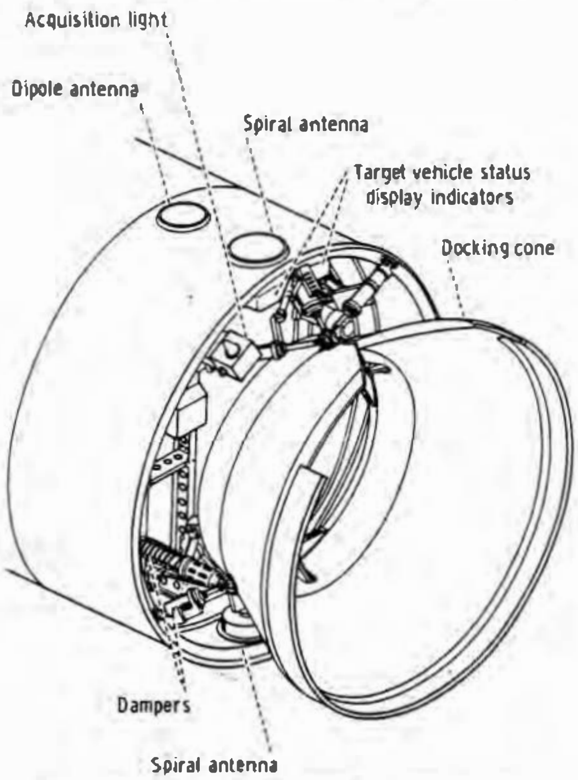

FIGURE 5-3.—Target Docking Adapter assembly.

Characteristics of the Docking System

The basic characteristics of the docking system were determined with a simple 2-degree-of-freedom model (fig. 5-4). By applying the conservation of momentum and energy laws, the energy absorbed by the docking system to provide for an inelastic impact is shown to be

$$T=\frac{1}{1+M_2/M_1}T_0 \tag{1}$$

where

$$T_0=\frac{1}{2}M_2V_0^2$$

and V_0 is the initial relative velocity between vehicles, M_2 is the spacecraft mass, and M_1 is the target-vehicle mass. Roughly, the ratio of masses for spacecraft and target vehicle is 1; therefore, about half of the kinetic energy associated with the relative motion of the vehicles must be absorbed. For a typical

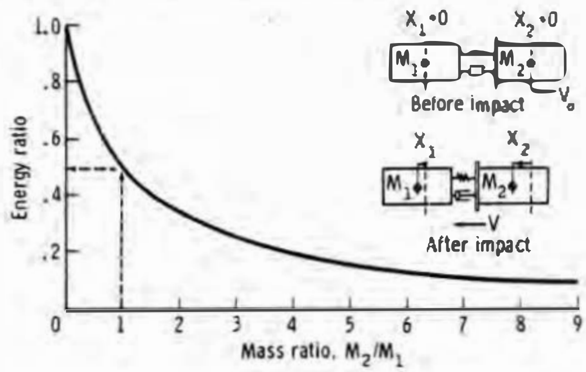

FIGURE 5-4.—Two-degree-of-freedom energy requirements.

closing velocity of 0.5 ft/sec, the system would absorb only about 15 ft-lb of energy.

The 2-degree-of-freedom model also determined the type of shock absorbers that should be used. The following design objectives were utilized: (1) minimum peak load, (2) minimum rebound characteristics, (3) reusability, and (4) maximum reliability. Consequently, the longitudinal members consisted of a spring for reusability and reliability, and of an orifice damper in parallel. The spring and the instroke orifice sizes were matched to produce minimum peak load on the instroke. On the outstroke, the damper fluid was metered through a much smaller orifice which minimized rebound. Since the longitudinal springs were sufficient to return the docking cone to the extended position, springs were not necessary in the lateral members.

After the basic design of the shock absorber had been determined, the analytical study was extended to include all the 8 degrees of freedom of a pitch-plane rigid-body system, consistent with the constraint of the spacecraft being in contact with the target-vehicle docking cone. The 8 degrees of freedom included the following:

(1) Target-vehicle horizontal translation, vertical translation, and pitch

(2) Docking-cone horizontal translation, vertical translation, and pitch

(3) Spacecraft pitch and translation along the surface of the docking cone

Initially, no control-system effects were included. This model permitted detailed investigation of the forces and motions which occurred during free docking.

Figure 5-5 presents a set of typical response parameters plotted against time for the case of the spacecraft impacting the docking cone with a horizontal relative-velocity component of 1.5 ft/sec and a vertical relative-velocity component of 0.5 ft/sec, the design-limit velocities. The initial point of impact at time 0 is near the leading edge of the top inner surface of the docking cone, 26 inches along the docking-cone surface from the latch plane. The motion of the spacecraft leading edge down the cone surface to the latch plane is represented by the curve labeled *D*. The force *F* between vehicles varies from a peak of nearly 300 pounds for this case, to a small grazing value after about 0.4 second. The figure also shows the inertial angular rates produced by *F* for each vehicle; these rates were initially zero. At about 1.5 seconds the spacecraft reaches the base of the docking cone, and the mathematical model no longer applies. The impact essentially has 2-degree-of-freedom characteristics after this point. The damper strokes are not shown on the figure but are available from the program. The maximum single-point contact load between the vehicles was determined to be approximately 800 pounds, and occurred when the spacecraft impacted on the bottom side of the docking cone approximately 1 foot from the latch plane.

Figure 5-6 shows the effect of having the stabilization systems of both vehicles on during docking. This case has the same initial conditions as the previous case when the stabilization systems were off. The main difference in vehicle response between the two cases is that the spacecraft attitude rate is now reduced to the 0.2 deg/sec deadband level instead of maintaining the 3.5 deg/sec level shown in figure 5-5. The target vehicle, on the other hand, acquires a slightly higher attitude rate with the systems on. The higher rate occurs because the spacecraft system is the more powerful and, in stabilizing the spacecraft, it overpowers the stabilization attempts of the target-vehicle system. Consequently, by the time the spacecraft reaches the latch plane, larger angular eccentricities between the vehicles result with the stabilization systems on rather than off and assuming the same errors at initial contact. This becomes less important when the ease with which the pilot can control initial errors in

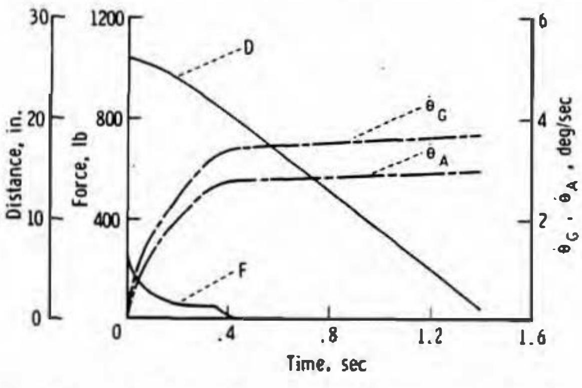

FIGURE 5-5.—Typical response with stabilization systems off. Initial conditions: horizontal velocity = 1.5 ft/sec; vertical velocity = 0.5 ft/sec; *D* = distance traveled by spacecraft leading edge along the docking cone; $\dot{\theta}_G$ = spacecraft inertial angular rate; $\dot{\theta}_A$ = target-vehicle inertial angular rate; *F* = force between the spacecraft and target vehicle.

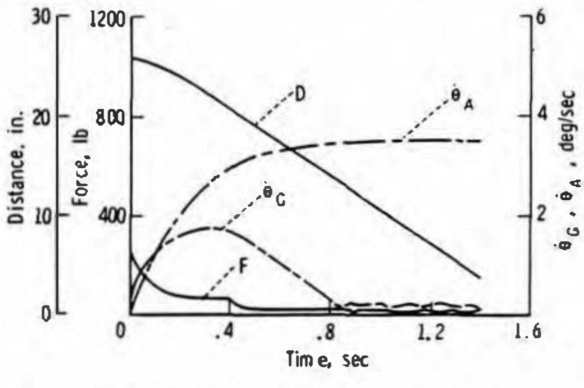

FIGURE 5-6.—Typical response with stabilization systems on. Initial conditions: horizontal velocity = 1.5 ft/sec; vertical velocity = 0.5 ft/sec; *D* = distance traveled by spacecraft leading edge along the docking cone; $\dot{\theta}_G$ = spacecraft inertial angular rate; $\dot{\theta}_A$ = target-vehicle inertial angular rate; *F* = force between the spacecraft and target vehicle.

the stabilized mode is compared with the unstabilized mode. Simulator training showed better pilot control when docking in the spacecraft rate-damping mode (the stabilized case) than in the direct mode (the unstabilized case).

While the 8-degree-of-freedom study was being made, a docking test was conducted with a ¼-scale dynamic model. The objectives were to confirm the design of the docking system by providing the following information:

(1) Stability of the shock-absorbing modes

(2) Maximum loads in shock-absorbing system components

(3) Time histories of the accelerations of each vehicle in all rigid-body 6 degrees of freedom

(4) Angular and linear misalinement limiting values for latching the two vehicles

(5) Adequacy of the proposed spring and damper characteristics of the shock-absorbing system

(6) Adequacy of the mathematical model used in the analytical studies

Each vehicle was represented by a ¼-scale model with a rigid-body mass and moment-of-inertia simulation. Other scale factors used in designing the models are listed in table 5–I.

TABLE 5–I.—*Docking Model Scale Factors*

Parameter	Scale factor, model/prototype
Assigned:	
Length	1/4
Time	1/4
Mass	1/100
Derived:	
Velocity	1/1
Acceleration	4/1
Spring rate	4/25
Kinetic friction	1/25
Preload force	1/25
Moment of inertia	1/1600
Angular velocity	4/1
Angular acceleration	16/1
Velocity-squared damp constant.	1/25

The kinematics of the model's shock-attenuation system closely duplicated the kinematics of the full-scale system, and the springs and dampers were dynamically scaled. The docking-cone surface was coated with the same dry-film lubricant planned for use on the full-scale system; similarly, the leading edge of the Rendezvous and Recovery Section of the spacecraft model was covered with a layer of fiber glass.

Each model was supported at the center of gravity by a low-friction gimbal device suspended by a 30-foot cable from a zero spring-rate mechanism. The device provided each model with the rigid-body 6 degrees of freedom required for simulating the orbital condition.

The tests confirmed the docking-system design in every aspect. The 8-degree-of-freedom analytical model was verified. This was desirable before the equations of motion were extended to include the stabilization systems of the vehicles, since a model test with active stabilization systems was not practical. The test indicated that angular eccentricities between the vehicles of about 5° at the latch plane would permit automatic latch.

The final development test of the docking system was a full-scale test using a Target Docking Adapter and a spacecraft Rendezvous and Recovery Section of the normal production configurations. The test setup was similar to the ¼-scale test except that zero spring-rate suspension mechanisms were not used. Each vehicle was suspended as a simple pendulum 57 feet in length, the maximum working height available. Also, the Target Docking Adapter contained an operational rigidizing mechanism which automatically actuated when all three docking-cone latches engaged the spacecraft. All systems performed satisfactorily during the test and favorably agreed with previous analytical and ¼-scale-model studies.

Design Considerations for Maneuvering the Docked Vehicle

During maneuvers, the critical loading condition on the docked vehicle was the bending moment at the spacecraft/target-vehicle latch joint. Two separate conditions produced design-limit loads. The first was the target-vehicle Primary Propulsion System engine performing a hard-over gimbal motion and remaining in the hard-over position. This malfunction produced the maximum bending moment at the latch joint, 117 500 inch-pounds. The bending moment, combined with the associated axial load of 11 000 pounds due to engine thrust, defined the design-limit load for the compression load paths of the docking-adapter structure and also for some stringer structure in the spacecraft Rendezvous and Recovery Section.

The second design condition resulted from terminating the Primary Propulsion System thrust at various times after initiation of the hard-over movement, and then determining the thrust termination time that yielded maximum bending at the latch joint with thrust completely terminated. The maximum bending moment (97 000 inch-pounds) with no accompanying axial load defined the design-limit load for the tension linkages in the mooring structure.

Using the test setup shown in figure 5–7, the Target Docking Adapter and the spacecraft Rendezvous and Recovery Section were qualified for ultimate load levels corresponding to the limit loads previously described. Instead of the usual 1.36 factor of safety for defining ultimate loads from limit loads, a factor of 1.5 was employed to account for the possible use of heavier spacecraft later in the Gemini Program.

A bending moment was applied in increments from 10 percent to ultimate about the horizontal axis, so that the bottom docking latch was placed in tension; no axial load was applied. The loading qualified the tension linkages in the docking-adapter mooring structure.

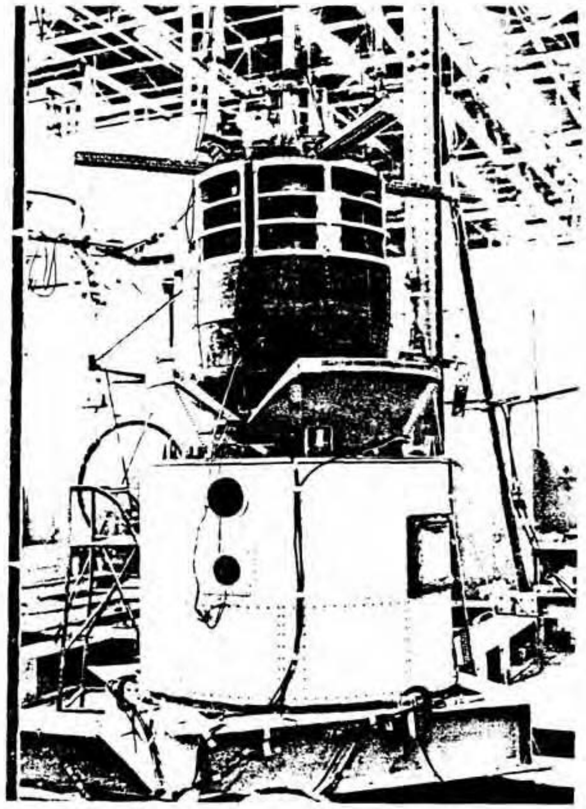

FIGURE 5–7.—Maneuvering loads qualification test.

Starting from zero loading, limit axial and shear loads were applied. Limit bending moment was applied, in increments of 10 percent, about the horizontal axis to place the bottom docking fitting in compression. The axial and shear loads were then increased to ultimate levels. Finally, the bending moment was increased to failure. Failure, in the form of buckling of two stringers adjacent to the bottom docking fitting on the spacecraft Rendezvous and Recovery Section, occurred at 227 percent of limit bending moment. This loading qualified the compression load paths of the Target Docking Adapter and the Rendezvous and Recovery Section.

Considering that the Gemini spacecraft would be a rather awkward payload for an Agena, it was reasonable to expect that the original Agena control system might be unsatisfactory. Based upon an initial estimate of 5 cycles per second for the first body bending frequency of the moored configuration, stability studies indicated that an inadequate gain margin existed in this mode. The Agena autopilot system was modified by adding a 5-cycle-per-second attenuation filter to the electrical compensation networks. Later estimates, however, indicated that the actual first bending frequency was considerably lower than the estimated 5 cycles per second and was closer to 3 cycles per second. This seriously affected the performance of the newly designed control system.

As shown in figure 5–8, the new control system failed to provide a minimum desirable gain margin of 6-dB and 25° phase margin in the dominant rigid-body mode for the applicable damping values of the first bending mode of the system. As computed here, gain margin is 10 times the common logarithm of the ratio of the upper critical gain to the lower; that is, a ratio of 4 gives 6 dB. The upper critical gain corresponded to instability of the first bending mode, and the lower gain corresponded to rigid-body instability. The dashed portions of the figure are extrapolated values obtained from the actual damping regime that was studied. To improve the gain margin available, the control system was modified by altering the configuration of the lead-lag network to accommodate the 3-cycle-per-second first bending frequency. The gain margins were significantly increased.

To determine the structural dynamic characteristics of the docked configuration, a ground vibration test was conducted using the test setup shown in figure 5–9. The spacecraft was moored to a Target Docking Adapter bolted to a target-vehicle forward auxiliary rack that was cantilevered from the laboratory floor. Data from this cantilevered configuration were then related to the actual spacecraft target-vehicle free-free configuration, which could not be conveniently simulated in the laboratory. Various axial load and docking-adapter bending-moment conditions were simulated to correspond with inputs from the target-vehicle Primary Propulsion System. The data of primary importance were those needed in the Primary Propulsion System stability study—minimum first bending-mode frequency and damping, and maximum cross-axis coupling. The minimum first free-free bending-mode frequency was determined to be 3.3 cycles per second. The damping ratio (C/C_c) of the first mode varied considerably with test conditions from

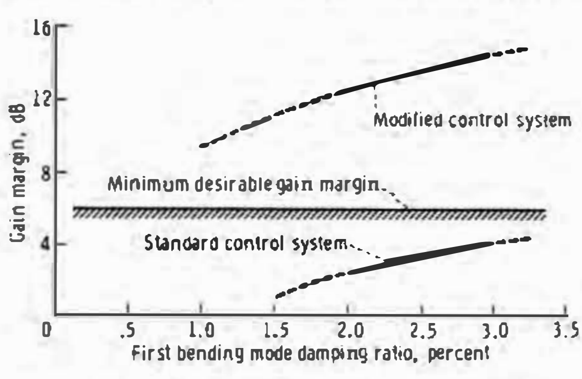

FIGURE 5–8.—Primary Propulsion System stability study.

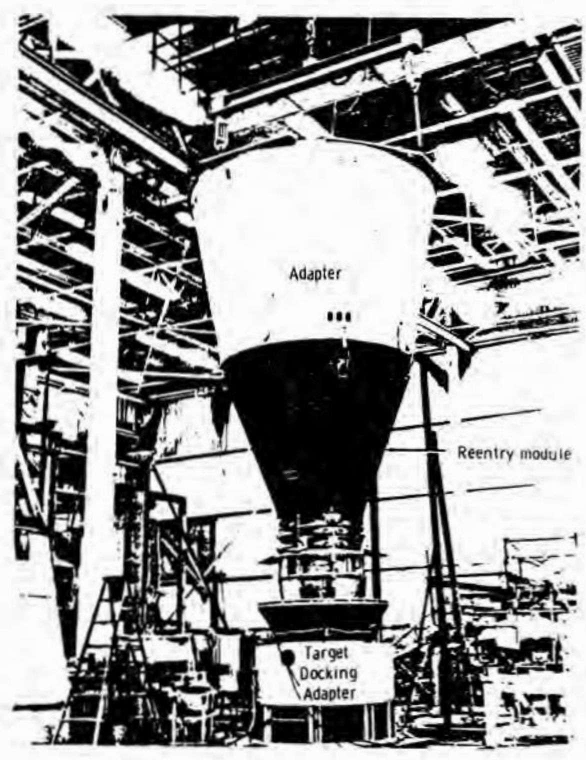

FIGURE 5–9.—Moored configuration ground vibration test.

a minimum of nearly 3 percent to a maximum of almost 5 percent. A minimum damping ratio of 2.34 percent was used in the study to account for possible high-temperature effects on the docking-adapter dampers. The cross-axis response in the test configuration was frequently 50 percent of the in-axis response, indicating that spring coupling coefficients of 3 to 6 percent should be included in the stability study equations of motion. Inclusion of the spring coupling effect in the study showed it to be only slightly destabilizing; this effect is included in figure 5–8.

Inflight Bending-Mode Test

When it became apparent that the original Agena control system was going to perform marginally during the docked Primary Propulsion System firings, a simple test was devised to determine inflight values of the first bending-mode frequency, damping, and cross-axis coupling. Determination of these parameters under actual flight conditions would have increased the confidence in the gain margins for this system (fig. 5–8). When the decision was made to replace the standard control system with a modified system, the inflight bending-mode test was retained in the flight plan as a final check on the docked configuration structural parameters.

The test was performed during the Gemini X mission. After the spacecraft and target vehicle were docked and rigidized, the command pilot fired a pair of spacecraft pitch-plane attitude thrusters for 3 seconds; this was immediately followed by a 3-second firing of the opposing pair of pitch-plane thrusters. The procedure produced three separate sets of vibrational motions for the first bending mode of the vehicles. Each set contained about 10 cycles. The same procedure was repeated in the yaw plane of the docked vehicles. Accelerometers having full-scale values of 0.02g were located in the spacecraft adapter section to sense the vibrations. The accelerometer signals were transmitted through the spacecraft telemetry system to a ground network station. The network station relayed the signals, in real time, to the Manned Spacecraft Center where the data were evaluated prior to the first firing of the target-vehicle Primary Propulsion System.

Table 5–II compares the inflight test data with corresponding data from the cantilever ground test. The first bending-mode frequency was 4 cycles per second and was about 10 percent higher than the frequency indicated from the ground test at corresponding amplitudes of vibration. Due to the thrusters firing, the moored vehicle was bent through an angle of 1 minute at the docking-adapter latch. The observed damping ratios varied from approximately 4.5 to 6.5 percent and were considerably higher than the ground-test value of about 3 percent. The differences could have been caused by low temperatures that sharply increased the contribution of the dampers to the total damping of the first bending mode. The temperature of the dampers was unknown. Cross-axis coupling was evident and was approximately the same level as indicated in the ground test. Since all measured values of frequency and damping were higher than the predicted values, and cross coupling was equal to the predicted values, the configuration was considered safe for maneuvers using the target-vehicle Primary Propulsion System.

TABLE 5–II.—*Comparison of Inflight Data With Ground-Test Data*

Test	Frequency, cps	Damping ratio, percent	Spring-coupling coefficient, percent
Ground	3.3	3 (Ambient temperature)	3 to 6
Inflight	4.0	4.5 to 6.5 (Temperature unknown)	3 to 6

Target-Docking Simulations and Training

The next evaluation of the target-docking systems was simulator training by the flight crews to develop proficiency for the docking and docked maneuvering phases of the actual flight. The first training phase was performed on the Translational and Docking Simulator which provided a full-scale simulation of close-in formation flying and docking maneuvers.

Differences in orbit-plane positions between the two vehicles were provided by lateral translation of the spacecraft mockup. A displacement of 22 feet either side of the center position was available. Differences in orbit altitude were represented by the vertical movement of the target-vehicle mockup with a total displacement capability of 33 feet. Closing or opening rates were simulated by moving the target vehicle toward or away from the spacecraft along a 125-foot horizontal track. Docking, latching, and rigidizing were accomplished with hardware similar to that to be used on the flight vehicle. Relative attitudes of both vehicles were provided by the ability of the spacecraft to move in all three axes: 45° to either side in yaw, 45° to either side in roll, and 40° down and 50° up in pitch.

The realism of the docking simulator was successfully demonstrated by comparing the conditions observed through the window of the trainer with those observed during the actual flights. The simulated closing and docking sequence started from a position slightly left of and below the target vehicle. The command pilot first maneuvered the spacecraft to aline the two vehicles, then translated forward with a relative velocity of approximately 1 ft/sec. The docking cone and docking bar adjusted for small alinement errors at impact and the docking cone absorbed the impact loads. After impact oscillations were damped, the spacecraft and target-vehicle mockups were rigidized and prepared for combined maneuvers.

Another part of the docking training was crew recognition of the status and safety of the systems in the target vehicle, and of the mooring system of the Target Docking Adapter. Visual observation of the target-vehicle status display (fig. 5-10), located above the docking cone, provided this information. Figure 5-10 shows a normal system condition as observed before docking. Green DOCK and PWR lights indicate that the mooring system is satisfactory for docking. The target-vehicle systems are verified by the green MAIN light, indicating that the hydraulic system pressure and the differential pressure between fuel and oxidizer are normal; by the green SEC HI and SEC LO lights, indicating that the Secondary Propulsion System is in a satisfactory condition; and by the green ATT light indicating that the target-vehicle cold-gas attitude system is activated. Upon docking, the green DOCK light is deenergized; when the vehicles are rigidized a green RIGID light is observed.

The second training phase was directed toward utilizing the target-vehicle systems, principally for attitude and translational maneuvers of the combined vehicles. This training was performed on the Gemini Mission Simulator at the Manned Spacecraft Center. The flight-crew control of the target vehicle and of the mooring system was through the encoder and docking-adapter controls, as illustrated on the spacecraft instrument display in figure 5-11. The docking-adapter controls on the center control panel

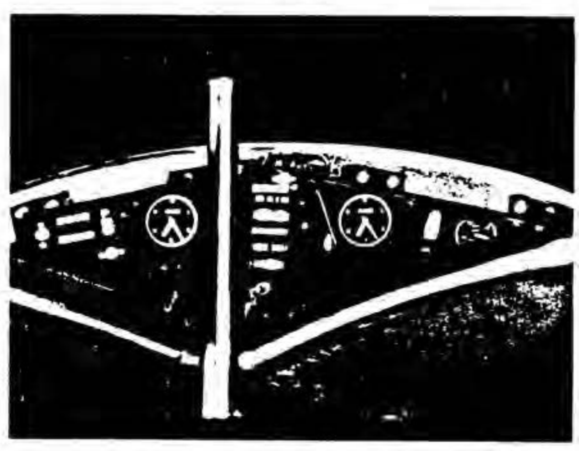

FIGURE 5-10.—Target-vehicle status display panel.

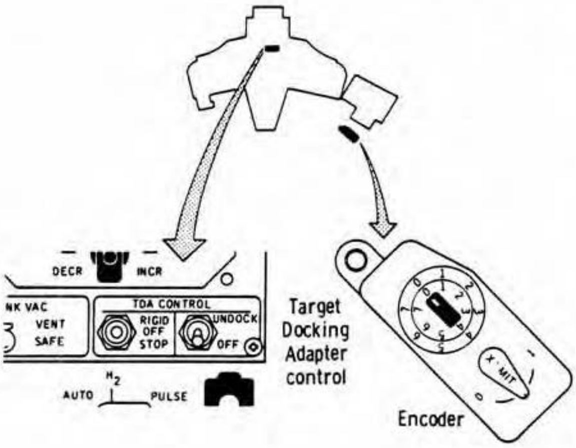

FIGURE 5-11.—Spacecraft instrument display.

were utilized for backup to the automatic rigidizing sequence and encoder-commanded unrigidizing signal. The crew used the encoder (located below the right-switch/circuit-breaker panel) to send commands to the target-vehicle propulsion, guidance, and electrical power systems. Approximately 100 commands could be sent to the target vehicle, and the sequence of the commands was significant; consequently, this phase of training was a major task.

Table 5-III shows an example of the sequence of commands required to perform a posigrade maneuver with the Primary Propulsion System. Before this sequence could be initiated, the spacecraft had to be configured for the maneuver. The spacecraft and target vehicle were then maneuvered to the proper heading; the Attitude Control System was adjusted for a Primary Propulsion System firing and for the desired velocity input; and the engine was activated. Sixteen seconds after the command to fire the Primary Propulsion System, the Secondary Propulsion System fired to establish the proper ullage configuration. The Primary Propulsion System initiate would not occur until 84 seconds after the PPS ON command, with automatic

TABLE 5-III.—*Posigrade Maneuver With the Primary Propulsion System*

Spacecraft command no.	Command title	Function
	Time = translation minus 30 min	
361	Geocentric rate normal	Establish proper heading for posigrade maneuver
310	Roll horizon sensor to yaw	
	Inertial Reference Package ON	
321	Horizon sensor to yaw	
	In phase	
460	Attitude Control System gain low	Establish necessary attitude control for Primary Propulsion System firing
310	Roll horizon sensor to yaw	
	Inertial Reference Package ON	
370	Attitude Control System pressure low	
450	Attitude deadband narrow	
271	Power relay reset	
	Time = translation minus 3 min	
041	Record data	Final system commands to lockout Target Docking Adapter, and prepare status display panel
471	Attitude Control System gain high, docked	
371	Attitude Control System pressure high	
271	Power relay reset	
201	Agena status display on bright	

TABLE 5-III.—*Posigrade Maneuver With the Primary Propulsion System*—Concluded

Spacecraft command no.	Command title	Function
Time = translation time		
501	Primary Propulsion System ON	
Time = translation plus 16 sec		Secondary Propulsion System ON occurs
Time = translation plus 84 sec		Primary Propulsion System initiate occurs
When inertial velocity indicator zeros: ENGINE, STOP		Primary Propulsion System shutdown, backup to automatic shutdown
Time = end of translation plus 2 sec		
500 450 370 451 271	Primary Propulsion System cutoff Attitude Control System gain low Attitude Control System pressure low Attitude Control System deadband wide Power relay reset	Disable the Primary Propulsion System and reset attitude control for nonthrusting operation

shutdown occurring after the desired velocity was achieved. A backup to the engine shutdown was performed by the flight crew by placing the engine switch to STOP. After shutdown the Primary Propulsion System was deactivated and the Attitude Control System was transferred to a nonthrusting configuration.

Crew training for the rendezvous and docking portions of the Gemini X, XI, and XII missions consumed an average of 89 hours per mission. This time would be approximately doubled if it included the docked maneuvering simulation training at Kennedy Space Center.

Docking and Undocking Flight Experience

Actual flight experience with docking and undocking of the spacecraft and target vehicle demonstrated that the design was sound, that testing had been adequate, and that crew training had provided a high degree of proficiency. Gemini VIII was the first mission in which a Gemini Agena Target Vehicle was placed in orbit. After a successful rendezvous and final station keeping, the following events occurred. The spacecraft was maneuvered to a position directly in line with the Target Docking Adapter at a distance of approximately 3 feet. The spacecraft attitude control system was in the rate command mode. After the command pilot had inspected the status panel, the docking cone, and the latches, he initiated the final approach by firing the aft-firing maneuver engines. Contact occurred with less than 2 inches of linear displacement, and with very little angular misalinement at a velocity of about 3/4 ft/sec. Onboard sequence pictures of the event show a smooth operation with no evident reaction by the target vehicle. The latches appeared to engage immediately, followed by cone retraction and illumination of the rigid light. The Target Docking Adapter data indicate accelerations less than 1g peak-to-peak in the horizontal and vertical axes, and less than ½g in the longitudinal axis. About ½ hour later, a spacecraft attitude-control problem caused an unscheduled emergency undocking. Although the combined vehicle rates at this time were 3 deg/sec in pitch, 2.5 deg/sec in yaw, and 5 deg/sec in roll, the undocking was smooth and orderly.

With one minor exception, all docking and undocking operations during the Gemini X, XI, and XII missions were equally smooth and uneventful. The exception was the second docking during Gemini XII. Flight-crew observations, onboard sequence pictures, and telemetry data indicate that the following probably occurred during this docking. Final approach of the spacecraft to the Target Docking Adapter was at a low velocity, and the point of contact was somewhat low. These factors caused the bottom docking latch to engage; however, the relative motion between the two vehicles stopped and the upper two latches did not engage. Sensing this, the command pilot immediately fired the aft-firing engines; but because the two vehicles were in contact, the thrust was insufficient to complete the dock. After about 40 seconds of unsuccessful maneuvers, a pitchup maneuver coupled with forward-firing engines caused successful separation. This condition had been encountered during tests and it was recognized that it could occur in flight; however, tests demonstrated that maneuvers, such as successfully employed in this case, would either separate the vehicles or would complete the dock, and no design changes were made.

An unexplained anomaly occurred after the second undocking maneuver during the Gemini XI mission. The undocking was accomplished by direct hardline signal from the spacecraft. Postseparation telemetry data indicated that the latches of the Target Docking Adapter had not reset; this was confirmed by crew observation. The crew recycled the unrigidized sequence using a radiofrequency command, and proper resetting followed. No further difficulties occurred but the hardline command was not used for the remaining undockings on this flight.

On all missions, while in the docked configuration, attitude control was excellent when using the various modes provided by both vehicles. Spacecraft rate command was used for random maneuvers when relatively fast operation was desired; very precise, but slow, cardinal-heading changes were made using the target-vehicle gyrocompassing maneuver. Spacecraft fixed-attitude control modes, such as platform or platform with orbital rate, provided good general control of the vehicles. However, for very precise pointing of the docked vehicles such as was required during photography, the target-vehicle Attitude Control System in the inertial mode was far superior to anything obtainable from the spacecraft systems. Because of the constant need to conserve spacecraft propellants for later phases of the missions, the target-vehicle control system was used whenever possible.

One of the most exciting aspects of the entire Gemini Program, and the primary reason for rendezvous and docking, was the capability to utilize the target-vehicle propulsion systems to greatly increase the maneuvering potential of the manned vehicle. This capability was not exercised on Gemini VIII because of the spacecraft control problem. However, Gemini X made very good use of this capability. First, as previously stated, an inflight test was performed to assure that the dynamic characteristics of the docked configuration would permit safe use of the target-vehicle Primary Propulsion System. Three Primary Propulsion System maneuvers and three Secondary Propulsion System maneuvers were performed on Gemini X. The maneuvers were all part of the highly successful and spectacular dual rendezvous of the docked vehicles with the Gemini VIII passive target vehicle which had been in orbit 4 months. Table 5-IV outlines the purposes of these maneuvers, the increased velocities realized, and the resulting orbital changes. It should be noted that the actual velocities gained during the Gemini X firings were greater than the command values. The error was caused by a characteristic of the target-vehicle velocity meter that allowed velocity errors to build up when the meter was activated for relatively long periods (4 minutes) of time prior to a firing. On subsequent flights, the velocity meter was activated only 20 seconds prior to a firing and was set with a positive null torque instead of a negative value.

The modified lead/lag stabilizing networks of the target vehicle were first utilized in the Gemini VIII mission. Larger-than-expected initial yaw-attitude transients were noted during the undocked Primary Propulsion System firings. The transients, in conjunction with the slow response of the autopilot, were directly related to the offset angle between the vehicle center of gravity and the geometric alinement axes measured from the engine gimbal point. Relatively large vehicle displacements and rates were required to position the engine so that the thrust vector would pass through the center of gravity. The vehicle excursions represent the normal control-loop linear response in the presence of center-of-gravity offsets. A typical attitude response is presented in figure 5–12. The target vehicle for the Gemini VIII mission had particularly large yaw center-of-gravity offsets because running light batteries were added to assist in-orbit visual sighting by the flight crew. In-plane and out-of-plane velocity errors resulted from attitude transients caused by Primary Propulsion System firing and from affected orbital maneuvering accuracies.

On missions subsequent to Gemini VIII, the center-of-gravity offset problem was minimized by adding ballasts on the target vehicle to locate the center of gravity at the approximate intersection of the lateral geometric alinement axes. Offsets were reduced to within alinement and center-of-gravity location uncertainties of the system. From target-vehicle insertion firing data, the magnitude of the heading errors resulting from alinement uncertainties could be approximated to provide inflight programing correc-

TABLE 5–IV.—*Docked Maneuvers During Gemini X*

Maneuver	Engine	Initiation of maneuver ground elapsed time, hr:min:sec	Length of firing, sec	Desired velocity, ft/sec	Actual velocity, ft/sec	Resulting orbit apogee perigee, n. mi.
Phase adjust, N_{CR1}	Primary Propulsion System	7:38:34	13	420.0	423.6	412/158.5
Height adjust, N_{CH2}	Primary Propulsion System	20:20:13	[illegible]	340.0	346.2	205.8/158.4
Circularization, N_{SR}	Primary Propulsion System	22:37:06	[illegible]	75.7	82.2	208.7/203.9
Phase adjust, N_{C1}	Secondary Propulsion System	22:45:36	10	7.7	9.7	209.9/205.0
Plane change, N_{PC}	Secondary Propulsion System	41:04:26	18	14.8	16.0	209.9/205.0
Phase adjust, N_{C1}	Secondary Propulsion System	41:35:50	[illegible]	3.3	4.4	208.5/203.5

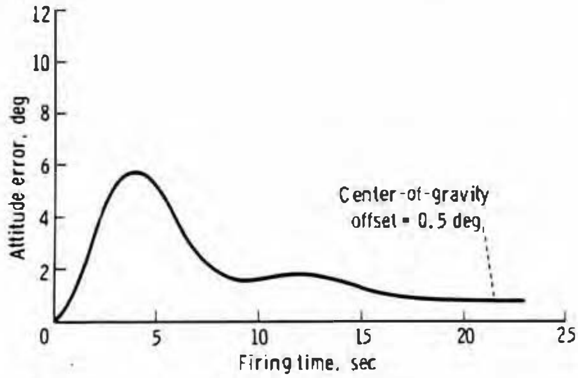

FIGURE 5-12.—Typical docked attitude response during firing.

tions for subsequent firings. Vehicle dynamic performance and the stabilizing influence which the modified lead/lag compensation network had upon the first body bending mode were as predicted in early stability studies. Except for the slow response, the maneuvers were satisfactory in all respects. The crew reported that the experience of accelerating backward produced no discomfort, and described the maneuvers as very thrilling. Table 5-V shows the three Primary Propulsion System maneuvers that were performed during Gemini XI to achieve the high-altitude apogee of 742 nautical miles. It should be noted that the modified velocity meter procedures resulted in very accurate velocities on this flight.

Onboard sequence pictures of the long firing to achieve the high altitude confirmed the crew description of visual effects of firing the Primary Propulsion System. The engine start was characterized by sparks, a yellow glow, and considerable visible flame. As full engine operation was reached, visible light was almost completely extinguished. Upon termination of the firing, the engine tailoff produced a display as spectacular as the ignition phase.

Concluding Remarks

From the experience in the Gemini Program relative to the operational characteristics of the docked configuration, several significant conclusions are apparent.

(1) The maneuvering and subsequent docking of spacecraft in orbit is practical and, when a proper design exists, is a relatively easy task.

(2) The joining of manned vehicles to unmanned craft containing large propulsion units can provide large maneuver capability where launch payload constraints prevent a combined launch.

(3) The development of docking and docked maneuvers of the Gemini spacecraft and the Gemini Agena Target Vehicle was in many respects a remarkable example of engineering success. It was a venture into an entirely new area of operation. No prior technology was applicable. It had all the impediments and interfaces of a combined effort by several large prime contractors, their subcontractors, and several Government agencies. Yet, most of the potential problems were eliminated in the drafting room, a few were discovered and corrected during test, and some were removed at the conference table. The efforts were culminated during the flight operations when all design parameters were easily met and problems were few.

TABLE 5-V.—*Docked Maneuvers Using Primary Propulsion System During Gemini XI*

Effect of maneuver	Initiation of maneuver ground elapsed time, hr:min:sec	Length of firing, sec	Desired velocity, ft/sec	Actual velocity, ft/sec	Resulting orbit apogee/perigee, n. mi.
Plane change	4:28:48	3	110.0	109.8	164.2/154.6
Raise apogee	40:30:15	25	920.0	919.6	741.5/156.3
Lower apogee	43:52:55	22.5	920.0	919.47	164.2/154.6

6. OPERATIONS WITH TETHERED SPACE VEHICLES

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Introduction

Basically, two modes of tethered space-vehicle operations were explored in the Gemini Program. One mode of operation consisted of intentionally inducing an angular velocity in the tethered system by translational thrusting with the spacecraft propulsion system. The other mode involved tethered, drifting flight during which the effect of gravity gradient on the motion of the system was of interest. These two modes of tethered-vehicle operation will be individually discussed.

Rotating Tethered Vehicles

The tether evaluation in the rotational mode was accomplished during the Gemini XI mission. This exercise was to evaluate the basic feasibility of rotating tethered-vehicle operations as the operations might apply to generating artificial gravity or to station keeping. The exercise consisted of connecting the spacecraft and target vehicle with a 100-foot Dacron tether, and then using the translational thrusting capability of the spacecraft propulsion system to induce a mutual rotation. The result of this mutual rotation was that the vehicles essentially maintained a constant separation at the ends of the tether. Figure 6-1 is an illustration of the spacecraft/target-vehicle tethered configuration.

Analytical Studies

The analytical studies made in support of the rotating tethered-vehicle exercise consisted of two distinct phases. The first phase was a general exploration of the properties of tethered-vehicle dynamics. The second phase consisted of an analysis of the specific spacecraft target-vehicle tethered configuration of the Gemini XI and XII missions. Primarily, the analytical studies were made using a 12-degree-of-freedom digital computer program. This program numerically integrated the equations of motion of two rigid bodies, each having 6 degrees of freedom and connected by an elastic tether. The program allowed the bodies to have arbitrary mass properties, and the tether attachment points to be arbitrarily specified. The tether was mathematically described as a massless spring obeying a linear force-elongation relationship, and as exhibiting a linear dashpot-type damping property. Since a model for the dynamic behavior of the tether was not included in the analysis, tether motions were not predictable from these studies. In this particular analysis, it was assumed that the only significant external forces on the system were control forces exerted by the spacecraft control system. This assumption eliminated gravity forces which were shown

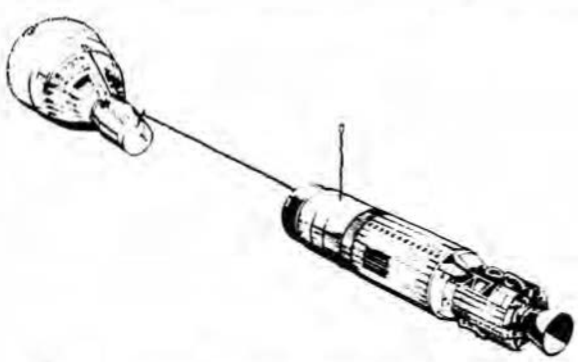

FIGURE 6-1.—Gemini spacecraft/target-vehicle tethered configuration.

to have negligible effect on short-term tether operations such as spinup and despin maneuvers. These studies predicted the dynamic behavior of tethered-system response to initial conditions and to simple, digitally simulated, control-system inputs; however, there was need for a study to reflect the interaction of man with the tethered system.

To supplement the digital studies, a 12-degree-of-freedom, real-time, man-in-the-loop simulation of the tether problem was implemented. This simulation was used to study the effects of pilot real-time inputs into the motion of a tethered-vehicle system by means of an attitude and translational control system. Information about the dynamic behavior of the tethered system was obtained from manual attempts to spin up the system, to control oscillations, and to despin the system.

Properties of tethered-vehicle dynamics.—The first study phase resulted in the establishment of the basic feasibility of the tethered-vehicle exercise. Two rigid bodies connected by a single elastic tether were found to have no alarming dynamic characteristics. The tethered system, however, was found to exhibit oscillational motions that were very complex and peculiar but which could be controlled to some extent with the spacecraft attitude-control system. The most interesting results of the first phase of the study were that tether damping was not very effective for reducing the attitude oscillations of a rotating tethered system, and that tether damping was quite effective in eliminating a slack/taut tether oscillational condition. These two properties of tethered-system motion are illustrated in figures 6–2 and 6–3.

Figure 6–2 illustrates two spinup starts which were identical, except that damping was present in the tether in one case, and no damping was present in the other case. The figure also presents a time history of tension in the tether, and the yaw angle of the spacecraft relative to the target vehicle. It can be seen that while the tension in the tether was strongly affected by damping, the attitude oscillation was relatively insensitive to tether damping.

Figure 6–3 illustrates the effectiveness of tether damping in eliminating a slack/taut tether mode of oscillation. This run started with an initially slack tether that quickly became taut, causing the slack/taut tether oscillation. A time history of the distance between tether attachment points is provided. Since the unstretched tether length was 100

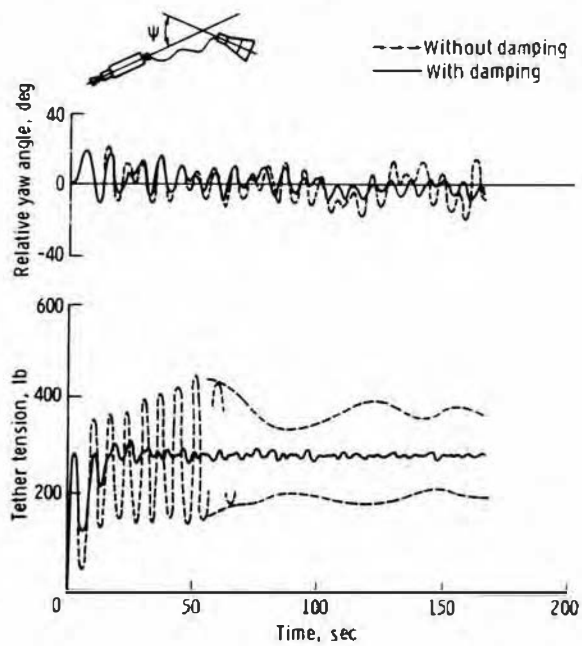

FIGURE 6–2.—Effect of tether damping on the attitude oscillations of tethered systems.

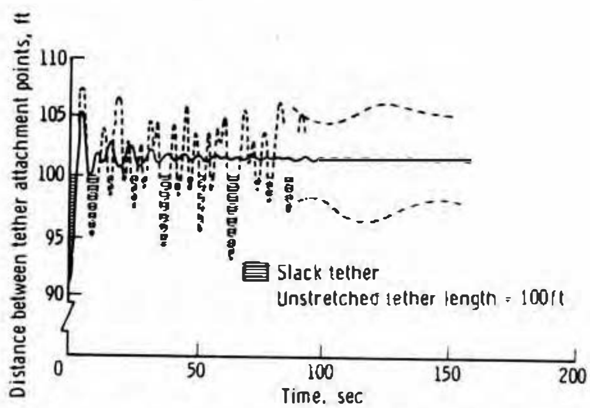

FIGURE 6–3.—Effect of tether damping on slack/taut oscillations.

feet in this run, any time the distance between the tether attachment points was less than 100 feet the tether was slack. It is apparent from figure 6–3 that with no tether damping, the slack taut condition continued throughout the run; but with tether damping, the slack/taut condition was quickly controlled and resulted in a constantly taut tether condition.

Spacecraft target-vehicle tethered configuration.—The second phase of the analytical study involved choosing a specific configuration for the spacecraft/target-vehicle tethered system. The selection of a specific configuration primarily involved the hardware and operational aspects. This freedom of choice was possible because the first phase study verified that a rotating tether-system operation was feasible and safe; besides, at this point in time, any possible configuration could be thoroughly studied. The tether length was specified as 100 feet as a compromise between maintaining safe separation of the spacecraft and the target vehicle and for minimizing fuel usage to obtain a given angular rate for the system. The tether size and material were dictated by an early program objective of producing significant artificial gravity effects (high tether loads). The tether spring rate of 600 pounds per foot was intentionally high so the tether could be broken by impact loading as a backup means of jettisoning the tether and the target vehicle if the primary jettisoning procedure should fail. Dacron webbing with a breaking strength of 6000 pounds was chosen as the tether material. The tether attachment points on the two vehicles were determined on the basis of minimum hardware implication on the Gemini Program. Attaching the tether to the spacecraft docking bar also provided a convenient scheme for jettisoning the tether. After it was decided that large artificial gravity effects would not be attempted in the Gemini Program, an 800-pound break link was installed in the tether to lower the requirements on the spacecraft propulsion system for impact breaking of the tether. The final tethered-vehicle configuration was then studied analytically to determine specific dynamic behavior.

Operational Aspects

The operational procedure for spinning up the tethered spacecraft/target-vehicle system consisted of backing the spacecraft away from the target vehicle until the tether was almost taut, then firing the translational thrusters to provide thrust on the spacecraft normal to the line between the vehicles. This imparting of angular momentum to the tethered system generally resulted in a net change in velocity of the center of mass of the system, and subsequently changed the orbit of the vehicles. This effect would not have been present if the system spinup had been accomplished with a pure couple; however, due to the passiveness of the target vehicle in the exercise, the spinup moment on the system had to be supplied solely by the spacecraft translation-control system.

The first complication associated with the operational implementation of the spinup tether exercise involved the fact that the spacecraft lateral translation thrusters had a significant component of thrust in the forward longitudinal direction. As a result, an attempt to spin up the system by firing only the lateral thrusters resulted in a significant closing rate between the vehicles. This closing rate produced an appreciable period of tether slackness, culminating in an extensive slack taut tether oscillatory mode. The alternatives to this spinup procedure were to orient the spacecraft so that its lateral thrust vector was, in fact, normal to the line between the vehicles, or to simultaneously thrust aft and laterally, thus holding the tether in tension during the spinup maneuver. Both methods had merit, depending upon the degree of spin rate desired for the system. Since the lateral and aft firing technique was applicable in all cases and was operationally simple, it was chosen as the operational technique for spinup of the system. For long-duration spinups, the aft thrusting could be terminated eventually, because the tether would remain taut during the remainder of the spinup due to the motion of the system.

During the spinup procedure, attitude control was required to maintain accurate thrusting to establish a desired spin plane. After the spinup was accomplished, neither the safety nor success of the exercise required further attitude control. Because tether damping did not prove to be an effective means of damping attitude oscillations, active attitude control was required when it became desirable to rapidly reduce spacecraft oscillations. It was found through simulation that the spacecraft control system could effectively reduce the attitude oscillations of the spacecraft; also, when the target vehicle was oscillating, those oscillations would ultimately be propagated through the tether to the spacecraft.

It was evident from the analyses that a differential rolling motion of the spacecraft relative to the target vehicle would probably be excited during the spinup maneuver. This mode of oscillation would be difficult to control with the spacecraft attitude-control system. Probably more difficult to control would be a rolling motion in which the target vehicle and the spacecraft were rolling together. Stopping this latter mode would require inducing a relative roll oscillation so that the tether could be used as a torsional spring which, although weak, would exert a roll moment on the passive target vehicle. Since mild rolling motions would not jeopardize the tether exercise, there was no reason for undue alarm.

From a safety-of-operation standpoint, establishment of a despin procedure was necessary. Such a procedure would enhance the probability of successful jettisoning of the tether at the termination of the exercise. The despin maneuver was essentially the inverse of the spinup maneuver. One procedure for despinning was to locate the spin plane of the system, either visually or with body-rate information available in the spacecraft, and then apply thrust in the spin plane and opposite the direction of spin. An alternative despin procedure involved applying thrust to reduce the line-of-sight rate to zero by visual observation of the spacecraft/target-vehicle line-of-sight motion. The despin maneuver invariably left the target vehicle with residual angular rates when the tether eventually became slack; however, this could be controlled by activating the target-vehicle control system in the despin procedure. An interesting phenomenon was discovered during the operational studies of the despin maneuver. Due to the location of the spacecraft attitude-control thrusters, and to the fact that attitude control of the spacecraft caused translation (the attitude-control moments not being couples), it was possible to automatically despin the rotating tethered system. By activating the rate-command attitude-control mode in the spacecraft and by commanding zero attitude rates, the attitude-control system would attempt to drive the spacecraft body rates to zero and produce a net translational thrust which slowly, but surely, would despin the system.

Crew Training

The crew training in preparation for the spinup tethered-vehicle exercise was primarily familiarization through simulation practice. To provide a realistic simulation of the interaction of two vehicles tethered together, a real-time simulation of the tethered-vehicle system was implemented.

The simulation facility consisted of a high-fidelity crew-station mockup, a planetarium-type projection visual display, and a hybrid-computer complex. The equations of motion describing two unconstrained rigid bodies (6 degrees of freedom per body) connected by a massless elastic cable were solved in real time on the hybrid-computer complex. This mathematical model included the off-symmetrical tether attachment points on the spacecraft and target vehicle, as well as the actual inertia properties of the vehicles. Best estimates of the tether-spring constant and damping characteristics were used for the training simulations. Included in the solution of the governing equations of motion was a simulation of the spacecraft attitude and translational control system. This simulation allowed real-time astronaut control inputs to properly effect the motions of the tethered vehicles. All basic flight instrumentation, as well as engineering parameters, were displayed in real time in the crew station.

The visual presentation consisted of a planetarium-type gimbaled Earth-scene horizon and star-field projection. The visual presentation of the target vehicle consisted of two spots of light from dual-target projectors. The two spots represented the ends of the target vehicle. This presentation allowed a visual recognition of maneuvering relative to the target vehicle, as well as observation of the attitude oscillations of the target vehicle. In flight, the tether would supply a visual cue concerning the separation distance between the two vehicles; however, in simulation, visual representation of the tether was not possible and the cue was supplied by a display in the crew station.

The training simulations usually began with the spacecraft undocked, but close to the target vehicle. The astronaut was then required to translate away from the target vehicle to a tether-extended position where the spinup maneuver would be initiated. After the system achieved the desired spin rate, the astronaut was free to observe the subsequent motions and obtain a feel for the behavior of the tethered system. Attitude control could be attempted in a direct, pulse, or rate-command mode of attitude control. Typical training exercises consisted of intentionally inducing large attitude oscillations in the spacecraft by means of the attitude-control system, and subsequently reapplying control moments to reduce these oscillations. Following these maneuvers, the astronaut could finish the exercise by practicing the despin procedure. Practice in breaking the tether with impact loading was also possible, since tether tension levels resulting from various maneuvers were displayed to the astronaut.

In addition to the crew training usage of the tether simulation, valuable engineering knowledge was gained concerning the general behavior of the tethered systems as well as of the specific configuration selected for Gemini. It was possible to observe in real time the response of a tethered system to very complex forcing functions (that is, inputs by a pilot). Although not directly associated with the flight maneuvers, the functions nevertheless yielded insight into the system behavior. The simulation allowed the design engineer to personally intervene in the scientific solution of the tether motion by way of a control system. The simulation was used to determine system response to control thrusters stuck in the ON position. Before the Gemini XI mission, the simulation was used to determine the effects of a degraded thruster prior to and in support of the actual spinup. Fuel usage for the spinup procedures was also determined in this training simulator.

Flight Results

During the Gemini XI mission, a total lateral thrusting of approximately 13 seconds was applied to the tethered system and resulted in a system spin rate of approximately 0.9 degree per second. Slack taut tether oscillations were induced during the spin following the termination of aft thrusting. This was due primarily to the fact that the tether tension associated with the low spin rate was smaller than the tether tension induced by thrusting aft; hence, at termination of aft thrusting, the tether simply catapulted the vehicles toward one another. After approximately 1½ orbits of the Earth, the spinup operation was terminated with a despin type of maneuver and the tether was jettisoned.

The results of the rotating tethered-vehicle maneuvers during the Gemini XI mission were essentially as anticipated. By comparing the motion pictures of the maneuver taken during the mission with the observations in the training simulation, it is evident that the simulation was quite accurate in predicting the general behavior of the tethered system. The flight crew found that the active damping of oscillations with the spacecraft attitude-control system was easier in flight than in the training simulation. This effect was probably due to the degraded sensory information available to the astronaut in the simulation as compared with the actual flight. It was observed that cable slack/taut oscillations damped out more rapidly in flight than in the simulation. This discrepancy was traced to a conservative value for the tether damping constant which corresponded to a room-temperature tether rather than a cold tether which would have a higher damping constant. As anticipated by analysis, the differential roll motion between the vehicles did, in fact, occur and was approximately to the extent predicted.

An interesting event occurred during the deployment of the tether. Near the end of deployment, a cable-dynamics phenomenon known as the skip-rope effect became significant. This behavior, although obviously possible, had not been predicted by the tether analyses employed in the design of the tether maneuver, since the studies did not include tether degrees of freedom. After the skip-rope mode of oscillation subsided, the spinup maneuver was successfully conducted with no evidence of significant cable-dynamics effects, thus confirming the analytical assumption that cable dynamics were not significant in the rotational behavior of this particular tethered system.

Gravity Gradient

The gravity-gradient tether exercise was accomplished during the Gemini XII mission to study the feasibility of using gravity-gradient effects in the stabilization of manned spacecraft. The exercise consisted of tethering the orbiting vehicles together, then arranging the vehicles one above the other at the ends of the extended tether (that is, along a local vertical). By imparting the proper relative velocities to the vehicles in this arrangement, the vehicles would proceed into a constantly taut tether configuration and the tethered system would be captured by the gravity gradient. This captured behavior would be manifested by oscillation of the system about the local vertical.

Analytical Studies

Analytical studies of the gravity-gradient tether exercise ranged from simple feasibility studies to fairly sophisticated analyses. While the operational feasibility of gravity-stabilized satellites was well established, the stability of two rigid bodies tethered together in orbit was questionable. Therefore, analytical studies were first aimed at exploring the basic behavior of a tethered system in a gravity field, and then at establishing the operational aspects of obtaining a gravity-gradient-stabilized tethered system.

The first feasibility studies were conducted using a mathematical model that consisted of two point masses (each with 3 degrees of freedom) subject to an inverse-square central force field. The two point masses were assumed to be connected by an elastic tether which satisfied a linear force-elongation relationship. The equations describing this system were numerically integrated in a digital computer program to yield time histories of the significant parameters in the analysis. This phase of the analytical study established that at least two point masses could be tethered together and gravity gradient stabilized. This study, of course, had applicability to the actual situation since it could be argued that two rigid bodies connected with a tether of sufficient length would exhibit particle-like behavior. Since there was no effective damping mechanism in the proposed tethered system, and since the gravity-gradient exercise could continue over but a few orbits, the success of the exercise was strictly a matter of giving the tethered system the proper initial conditions. This being the case, the first phase of the study consisted of determining the response of the tethered system to various combinations of initial conditions.

The initial conditions for a perfect start were established; these included a slightly taut tether, and a relative velocity of about 0.138 ft/sec for a 100-foot tethered spacecraft/target-vehicle combination. The perfect start, of course, also included an initial alinement along a local vertical and an approximately circular orbit for the system. Response to the perfect start consisted of continued alinement of the two point masses along the local vertical and of a constantly taut tether. Perturbations to this perfect start involved off-nominal relative velocities which were not compatible with continued motion along the local vertical, or an initially slack tether with or without range rate between the bodies. The tethered point masses were found to be reasonably tolerant of off-nominal starting conditions. For small perturbations, the solutions to the motions of the tethered point masses were in agreement with linearized dumbbell-satellite theory. This point-mass analysis was eventually modified to include an oblate earth as the attracting force on the point masses. This change was found to have negligible effect on the behavior of the tethered system. From the first phase of study, it was concluded that gravity-gradient stabilization could possibly be obtained with the spacecraft and target vehicle in the tethered configuration. Figure 6–4 illustrates typical results obtained from the point-mass analysis on the sensitivity of the system motion to initial relative velocity between the point masses.

The second phase of the analytical studies was conducted using a mathematical model consisting of two rigid bodies in planar motion subject to an inverse-square central force field, and connected by an elastic tether. The equations of motion describing this mathematical model were integrated numerically in a digital computer program to provide time histories of significant parameters. This phase of the study was implemented to answer questions concerning the rigid-body attitude response of the spacecraft and the target vehicle during the gravity-gradient exercise, and to confirm the validity of the conclusions drawn from the point mass analysis. From the results of this rigid-body study, it was found that (1) there was good agreement between the rigid body and the particle analysis concerning capture limits and tolerance to starting perturbations; and (2) there could be considerable rigid-body rotation of the target vehicle and the spacecraft during the gravity-gradient exercise. Figure 6–5 illustrates a typical time history provided by the planar rigid-body analysis. Of importance was the determination that the capture sensitivity of the system was not significantly related to the rigid-body-attitude initial conditions. This fact was certainly welcome from the operational standpoint of setting up a captured system. On the other hand, the large rigid-body excursions of the vehicles would have an operational implication on such things as observation of the total system motion during the gravity-gradient exercise. While this rigid-body study provided valuable information, there were still a few questions concerning the rigid-body response of the vehicles and the stability of the system with all degrees of freedom present.

To answer these questions, a final study phase was implemented. The final phase con-

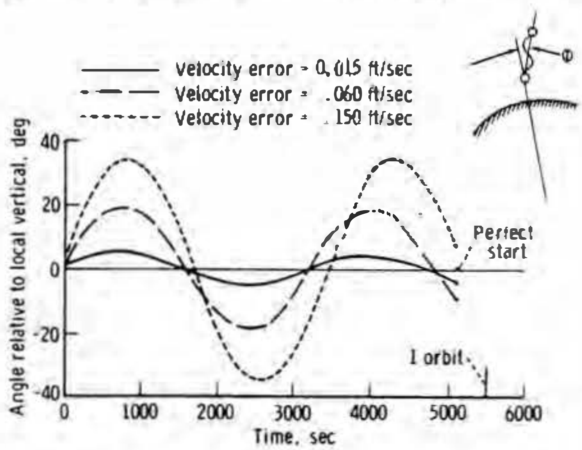

FIGURE 6–4.—Effect of off-nominal relative velocity on motion of gravity-gradient tethered system.

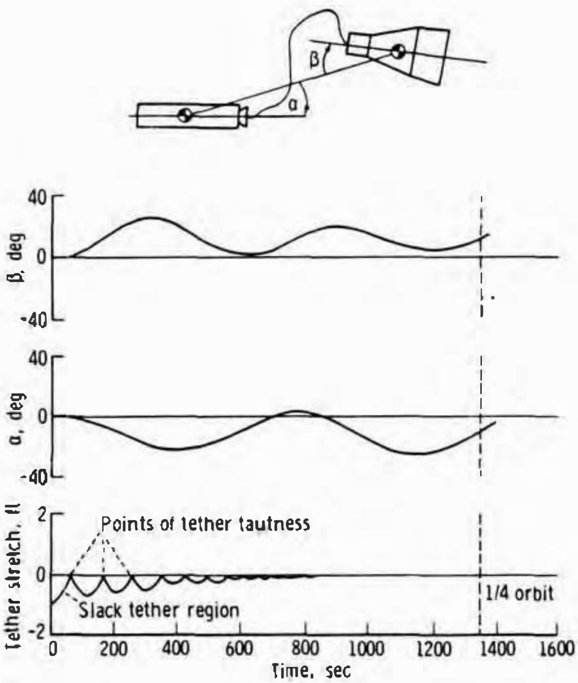

FIGURE 6-5.—Effects on rigid-body attitude response during gravity-gradient motion due to initial tether slackness of 1 foot.

sisted of solving the equations of motion describing two rigid bodies (each with 6 degrees of freedom) in an inverse-square central force field and connected by a linear elastic tether. This study confirmed the applicability of the lesser analyses that had been performed, in that good comparisons of capture limits and response to perturbations were obtained. As expected, the results of the final study indicated that a captured system would still be likely to have large rigid-body-attitude excursions; however, of even more significance, was the finding that there were no unforeseen instabilities in the behavior of the proposed gravity-gradient exercise. This final phase of study was primarily concerned with the spacecraft/target-vehicle configuration which would be used in the mission.

This concluded the analytical study phase of the tethered-vehicle gravity-gradient experiment. With the theoretical validation of the exercise completed, the problem then was to devise an operational technique to provide the proper initial conditions for the tethered system.

Operational Aspects

The objective of the gravity-gradient-stabilized tethered-vehicle exercise was to orient the vehicles one above the other (along a local vertical), and to provide proper starting conditions so that the subsequent motion would, at worst, be a limited amplitude oscillation of the system about a local vertical, and, at best, a continued perfect orientation along a local vertical. The proper starting conditions consisted of a slightly slack tether and a relative velocity of 0.138 ft/sec. Although it was relatively easy to position one vehicle directly over the other with a slightly slack tether, it was much more difficult to obtain a relative velocity of 0.138 ft/sec between the vehicles. A deviation of more than 0.23 ft/sec from the perfect relative velocity would mean that the gravity-gradient torque on the system could no longer contain the oscillations of the system around the local vertical; the system would then cartwheel, or be spun up.

The problem of obtaining the correct relative velocity between the spacecraft and the target vehicle was approached as follows. The perfect initial relative velocity corresponded to that relative velocity which would exist between the separated bodies if they were both attached to the same radius vector from the center of the Earth and rotating at orbital rate. It was decided to make use of this fact in the starting procedure. The capability existed on board the spacecraft to provide information to the flight crew from which the longitudinal axis of the vehicle could be made to coincide at all times with the local vertical direction. By positioning the spacecraft directly above the target vehicle with the longitudinal axis of the spacecraft maintained continuously along a local vertical, deviations from the perfect relative-velocity conditions would be manifested as drift of the target vehicle relative to the spacecraft. This drift could be detected quantitatively by the flight crew using the optical sight, and could be converted to an equivalent drift rate. From the drift rate. the deviation in relative velocity from the perfect start could be determined; hence, an appropriate velocity correction could be applied with the spacecraft translational thrusters. A perfect relative-velocity start would result in a zero-drift rate of the target vehicle relative to the spacecraft, as long as the longitudinal axis of the spacecraft was continuously along a local vertical. Figure 6-6 shows a flight chart from which the flight crew could take quantitative drift measurements (as angular drift in the optical sight) over a measured period of time and find the equivalent drift rate in the form of a relative-velocity correction. The flight chart indicates the expected maximum oscillation of the system from a local vertical for a given error in relative velocity. After the flight crew had ascertained that an acceptable initialization had been accomplished, the flight plan required that all thrusting be terminated and the drifting system observed to determine the success of the initialization. While a perfect starting condition dictated a very slightly taut tether, it was operationally more feasible to start the system with a definitely slack tether, and a zero-closure rate. This was due to the minimal perturbation to, and

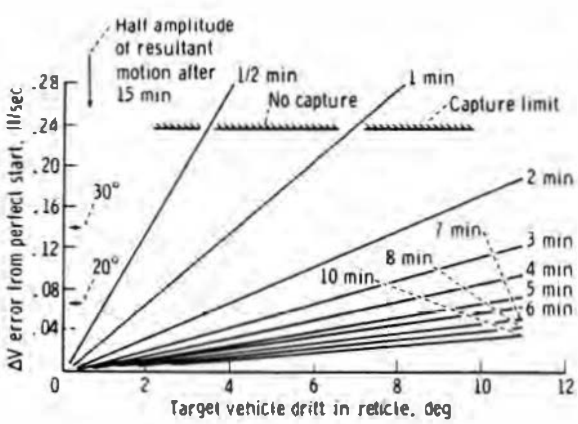

FIGURE 6-6.—Starting procedure chart for Gemini XII gravity-gradient tether exercise.

rapid recovery of the system from, an initially slack tether. The gravity-gradient effects would soon draw the tether taut (this being the stable configurations for the tethered system) for the remainder of the operation. The penalty paid for an initially slack tether was an increase in the angle of oscillation of the system relative to a local vertical.

Crew Training

Crew training for the gravity-gradient tether exercise consisted of briefings and simulator exercises. The significant flight-control task involved measuring the drift of the target vehicle in the optical sight, then applying the proper translational thrust to correct the relative velocity of the vehicles. The training was accomplished in the Gemini Mission Simulator, which had the capability to start a flight simulation run with the spacecraft docked with the target vehicle. The simulation exercise could then proceed with the undocking, followed by a maneuver to reach a position approximately 100 feet above the target vehicle. From this position, the use of the flight chart for the gravity-gradient starting procedure could be practiced. The mission simulator did not include tether dynamics or a visual simulation of the tether. This deficiency did not greatly hinder training for the gravity-gradient exercise, since the cable was not supposed to be taut during the starting procedure. The significant task to be practiced in training was to maintain a local vertical with the aid of the spacecraft instrumentation, and to detect and remove target-vehicle drift rates relative to the spacecraft.

Flight Results

There were three orbits allotted to the gravity-gradient tether exercise on the Gemini XII mission. Approximately half of this orbit time was used in establishing the starting conditions for the exercise. The remainder of the allotted time was spent observing the subsequent motion of the system.

The initialization of the system consisted of various translational and attitude thrusting maneuvers by the spacecraft, and an active stabilization of the target vehicle using the target-vehicle control system. After the flight crew had ascertained that acceptable initial conditions had been achieved, the crew deactivated the target-vehicle control system and terminated all spacecraft thrusting. The resulting motion was one of limited amplitude oscillations relative to local vertical. It was evident that the system was indeed captured by the gravity gradient. After initial perturbations, the tether became constantly taut, and the attitude oscillations of the spacecraft were of sufficiently limited amplitude that the crew were able to view the target vehicle almost continuously. Under these conditions, the target vehicle was never observed to rise toward the horizon by more than approximately 60° from local vertical.

The initialization of the gravity-gradient exercise was greatly hampered because some of the control thrusters on the spacecraft were malfunctioning. Attitude control had degraded to the extent that the preflight planned procedure for setting up the gravity-gradient exercise could not be accomplished. Despite this handicap, the crew was able to devise a backup procedure consisting of judicious use of remaining thrust capability to provide initial conditions for a successful gravity-gradient capture.

The simulation training for the gravity-gradient exercise was adjudged by the crew to present a more difficult problem than the actual flight situation. The crew concluded that, with a properly functioning control system, the gravity-gradient-capture initial conditions could have been accomplished with relative ease and certainty.

MAN'S ACTIVITIES IN SPACE

7. LIFE SUPPORT SYSTEMS FOR EXTRAVEHICULAR ACTIVITY

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Introduction

The Gemini Program has provided the U.S. Space Program with the initial steps in the study of manned extravehicular activity. Extravehicular activity was planned for 6 of the 10 manned Gemini flights and was actually performed during 5 flights. One prerequisite for attempting extravehicular operations was a reliable life-support system to provide the extravehicular pilot with a habitable environment while outside the protective confines of the spacecraft. The life-support system consisted basically of a space suit, a portable environmental control system, and an umbilical link with the spacecraft. This paper will trace the development of the suits, the environmental control system, the umbilical, and the related components from the original concepts through the modifications imposed by specific missions.

Testing

All elements of the extravehicular life-support systems were subjected to comprehensive unmanned and manned testing. Unmanned testing was performed individually on the space suits, the portable environmental control systems, and the umbilicals, and most manned testing concentrated on end-to-end tests. These manned tests included operation with the flight spacecraft for final verification of satisfactory performance.

The unmanned tests included humidity, vibration, explosive decompression, acceleration, oxygen compatibility, exposure to simulated space environment, temperature cycling, and shock. In some instances, tests were performed on a single life-support system element to fulfill some special requirement. For example, the space suits were tested for their ability to retain integrity during seat ejection tests.

The manned test series was performed at the Manned Spacecraft Center and at the spacecraft contractor facilities. Qualification tests for demonstrating the adequacy of metabolic heat rejection under induced workloads up to 2400 Btu hr were performed in high-altitude and space simulation chambers. Operation of the self-contained oxygen supplies of the Gemini IV and Gemini VIII through XII chest packs was verified as a suitable emergency mode should the extravehicular crewman lose the spacecraft oxygen supply. The crews practiced the various steps required to return to the spacecraft Environmental Control System in a decompressed cabin environment. This type of testing was performed in a vacuum chamber equipped with operational life-support system gear and a boilerplate Gemini spacecraft.

Space Suits

During an extravehicular mission the space suit becomes, in effect, a small, close-fitting pressure vessel which has to maintain a structurally sound pressure environment and provide the pilot with metabolic oxygen and thermal control. The space suit must also provide the body-joint mobility necessary for the pilot to perform the assigned extravehicular tasks.

The basic Gemini space suit was a multilayer fabric system generally consisting of a comfort liner, a gas bladder, a structural restraint, and an outer protective cover. To permit easy donning and doffing of the suit and components, quick disconnects were located at the wrists for glove connections, at the neck for helmet connections, and at the waist for ventilation-gas connections. Suit entry and body waste management were provided by a structurally redundant pressure-sealing zipper. Internal to the suit, a gas distribution system directed a flow of oxygen to the helmet area for metabolic use and thermal control, and over the limbs and body for thermal control.

Accessories provided on the suit included handkerchiefs, pencils, survival knife, scissors, neck dam, wrist dams, parachute harness, and stowage pockets for the flight-data books and charts (fig. 7–1). Equipment added to the space suit for extravehicular missions included: (1) extravehicular coverlayer, (2) pressure thermal gloves, (3) visor temperature-control coating, and (4) sun visor.

Gemini IV Mission

The Gemini IV mission objectives included short-duration extravehicular activity and evaluation of the basic extravehicular equipment. The basic (G3C Series) Gemini suit was adapted for extravehicular use (fig. 7–2) by incorporating the following:

(1) The extravehicular coverlayer consisted of nylon felt material for micrometeoroid protection, seven layers of aluminized

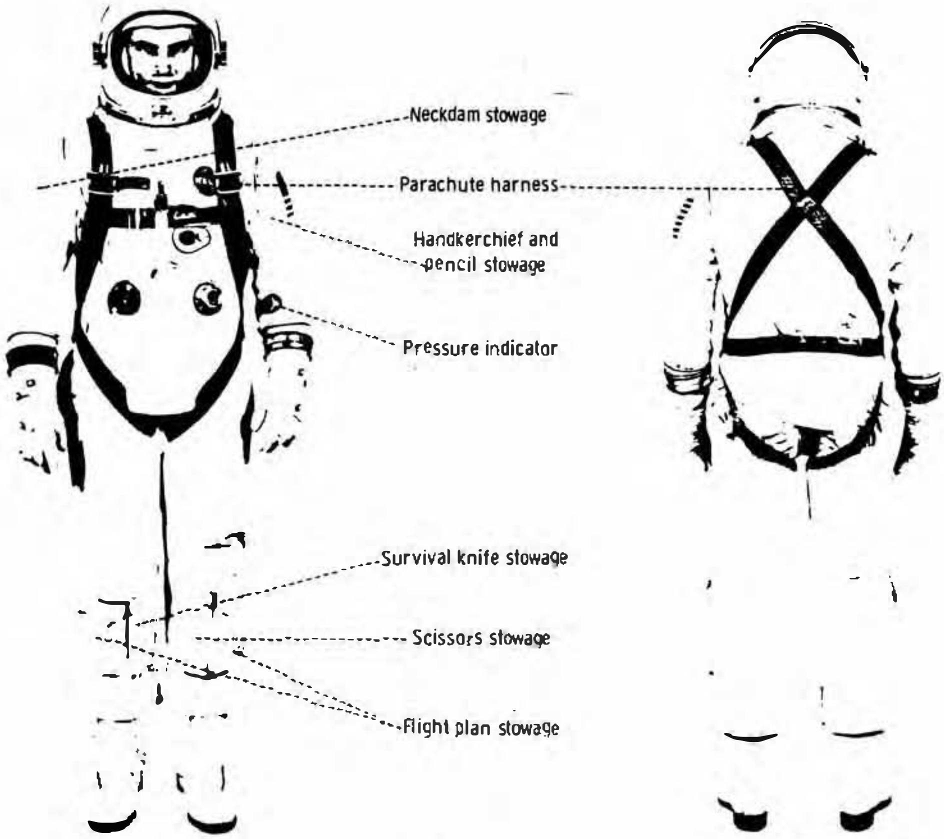

FIGURE 7–1.—Gemini G4C extravehicular space suit.

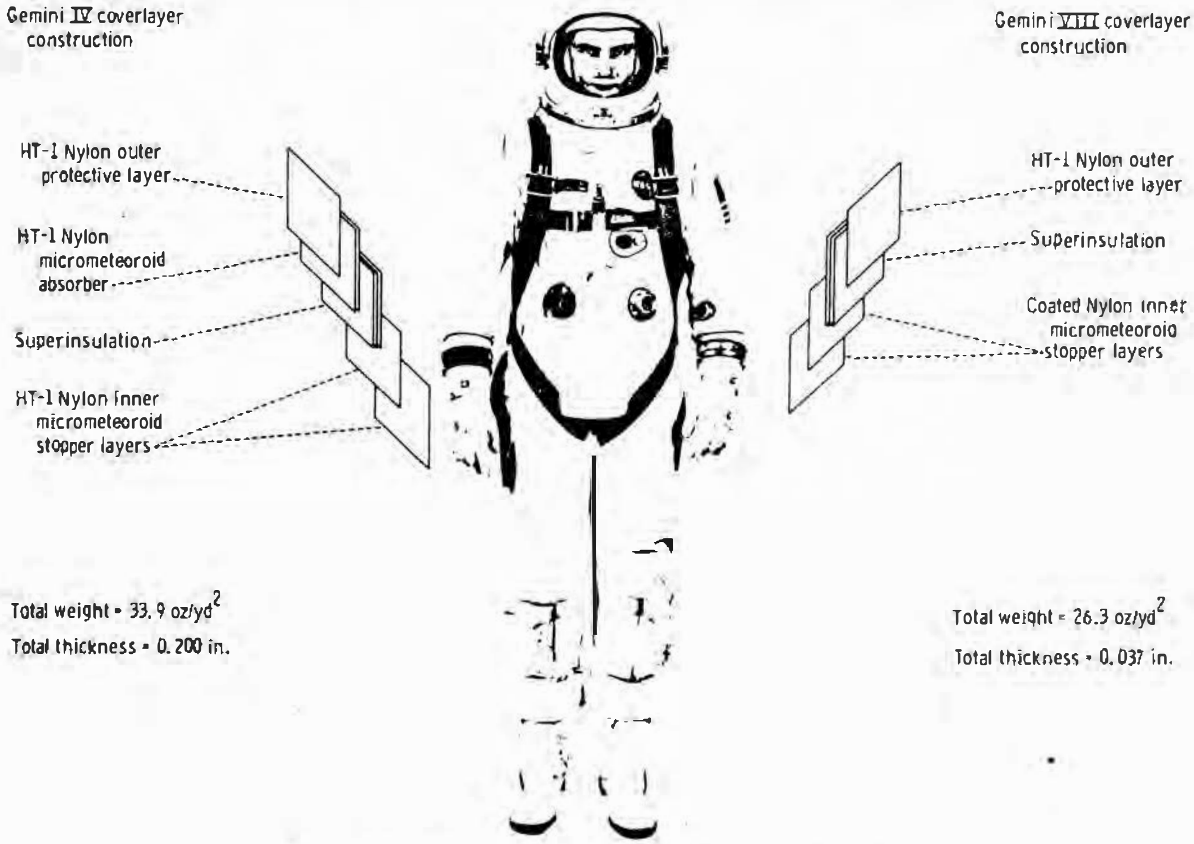

FIGURE 7-2.—Gemini IV and VIII extravehicular space suit.

Mylar superinsulation, and an outer covering of high-temperature nylon cloth.

(2) The extravehicular visor was a two-lens assembly with the outer lens providing visible and infrared energy attenuation, and the inner lens providing impact protection and thermal control.

(3) Thermal overgloves were provided for protection from conductive heat transfer.

During the Gemini IV mission, no difficulties were experienced with any of the spacesuit equipment. The mission demonstrated the following:

(1) The adequacy of the micrometeoroid and thermal protection of the coverlayer

(2) The acceptability of the visible light attenuation of the sun visor

(3) The adequacy of the thermal-control coating on the impact visor to maintain the pressure-visor surface temperature at the proper level

(4) The adequacy of the pressurized suit mobility to permit the pilot to egress and ingress the spacecraft

(5) The need for reduced coverlayer bulk to improve unpressurized suit comfort

Gemini VIII Mission

The space suit (fig. 7-2) used for the Gemini VIII mission was basically the same as the suit provided for the Gemini IV mission, with the following exceptions:

(1) The micrometeoroid protective layer was improved to provide significant reductions in coverlayer bulk (fig. 7-2).

(2) The thermal protection for the gloves, previously a part of the overglove, was incorporated into the basic pressure glove to

provide integrated thermal-conduction protection.

The Gemini VIII extravehicular equipment was not evaluated in flight due to early termination of the mission.

Gemini IX–A Mission

The Gemini IX–A mission imposed some very difficult requirements upon the space-suit assembly. To use the Astronaut Maneuvering Unit in conjunction with the space suit, it was necessary to redesign the lower portion of the extravehicular coverlayer to protect the pilot from the high-temperature (1300° F) impingement by the thruster plume of the Astronaut Maneuvering Unit. The suit was modified as follows:

(1) To afford protection from the high-temperature plume, the extravehicular coverlayer in the leg areas included a stainless-steel fabric outer covering to provide thermal energy distribution and erosion protection. A high-temperature superinsulation was used below the outer cover; the superinsulation consisted of alternate layers of double aluminized film and lightweight fiber glass.

(2) To further protect the visor from impact damage, the plexiglass pressure visor was replaced with a coated polycarbonate pressure visor. This modification also permitted the use of a single-lens sun visor.

Due to fogging of the pressure visor during the latter portion of the extravehicular activity, the Astronaut Maneuvering Unit experiment was not completed; consequently, the plume protection provided for the legs could not be evaluated. However, the mission indicated the need for an inflight application of antifog solution to preclude visor fogging.

Gemini X, XI, and XII Missions

The space suits for the Gemini X, XI, and XII missions were generally of the same configuration as the suits provided for the Gemini VIII and IX–A missions. The specific experiments and operations of each flight required only minor modifications to the suits. These missions continued to expose man to the extravehicular environment, and each exposure offered areas for improvement of the space-suit equipment.

Environmental Control Systems

Two different portable environmental control systems were developed for use in Gemini extravehicular activity. These included the open-loop system used on Gemini IV and the semi-open-loop system used for Gemini VIII through XII. The basic functions of both systems were identical: (1) to provide metabolic oxygen within the suit, (2) to provide the necessary controls to maintain suit pressure at the proper level, (3) to provide ventilation gas for carbon-dioxide washout, (4) to provide a means of removing the thermal load generated by the extravehicular pilot, and (5) to provide an emergency oxygen supply to assure pilot safety in case of loss of the primary oxygen supply. The Gemini IV Ventilation Control Module System was composed of a Ventilation Control Module, two multiple gas connectors, a 25-foot umbilical, and a restraint system.

The Gemini VIII through XII Extravehicular Life-Support System consisted of a chest pack, two multiple gas connectors, two hoses connecting the multiple gas connectors to the inlet and outlet ports of the chest pack, and a restraint system. In addition, an umbilical was an integral part of the system when operating from the spacecraft supply systems. For Gemini VIII, IX–A, and XII, a 25-foot umbilical and an electrical cable were utilized. For Gemini X and XI, a 50-foot and a 30-foot umbilical, respectively, performed the combined function of the electrical cable and 25-foot umbilical.

Ventilation Control Module System

The Ventilation Control Module (fig. 7–3), flown on Gemini IV, was mounted on the pilot's chest by Velcro straps attached to the

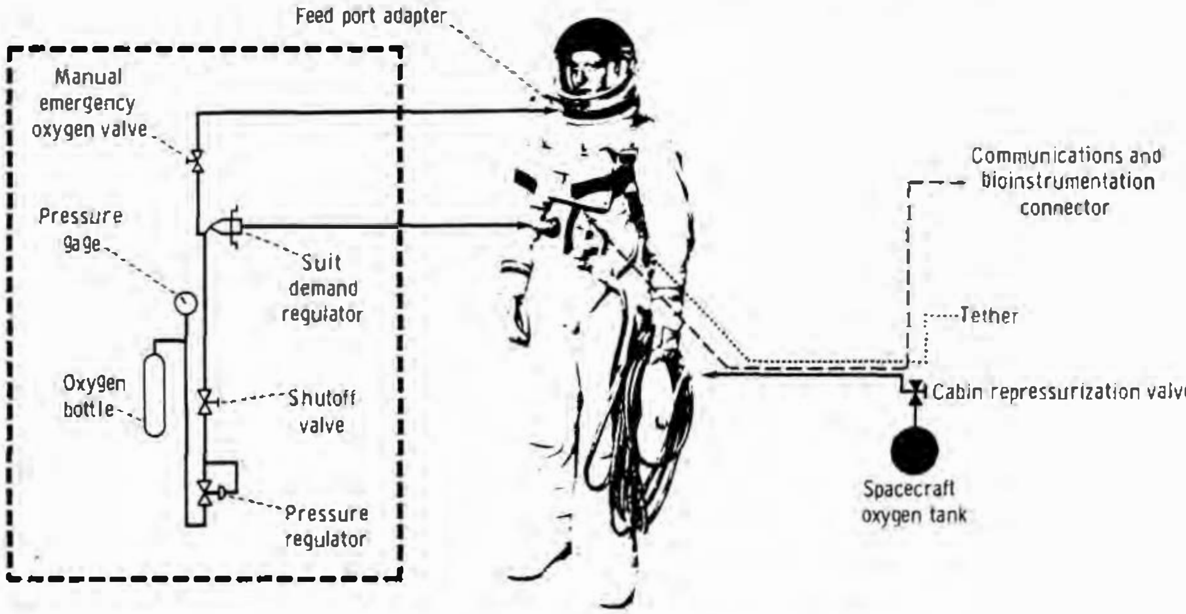

FIGURE 7-3.—Gemini IV Extravehicular Life-Support System.

parachute harness, and was connected to the suit-ventilation outlet fitting through a multiple gas connector. The Ventilation Control Module was an open-loop system; the gas was not recirculated through the system. In operation, oxygen flow of approximately 9 lb/hr was supplied to the suit to provide ventilation and for oronasal carbon-dioxide washout for metabolic rates not greater than 1000 Btu hr. The oxygen was supplied from the primary spacecraft oxygen supply through a 25-foot umbilical and a flow restrictor. The exhaust flow from the suit was controlled by a demand regulator so that suit pressure was maintained at approximately 4 psia. The emergency oxygen supply in the Ventilation Control Module was capable of supplying oxygen for 7.5 to 9 minutes. The pilot could have activated an emergency oxygen valve to initiate oxygen flow directly into the helmet by means of an adapter installed in the helmet feed port. If a leak had developed in the suit, a makeup flow of oxygen, sufficient to maintain suit pressure, would have been initiated automatically from the emergency supply.

Extravehicular Life-Support System Chest Pack

The Extravehicular Life-Support System chest pack (fig. 7-4) was flown on the Gemini VIII through XII missions. This system was designed to provide greater heat-rejection capability than the Gemini IV system, while requiring no more oxygen makeup flow from the spacecraft. The chest pack was secured by Velcro straps attached to the parachute harness, and was connected to the suit ventilation inlet and outlet fittings through two multiple gas connectors. The chest pack was a semi-open-loop system; approximately 75 percent of the ventilation gas was recirculated through the system (fig. 7-5). The chest pack was designed to accommodate average metabolic rates of 1400 Btu hr with peaks of 2000 Btu hr. Tests showed that the system was capable of higher heat loads, provided the higher loads were not imposed at startup. Normally, oxygen was supplied at approximately 90 psig from the spacecraft through a quick-disconnect fitting attached to the cabin repressurization valve; however, the Extravehicular Support Package and the Astronaut Maneuvering Unit backpacks car-

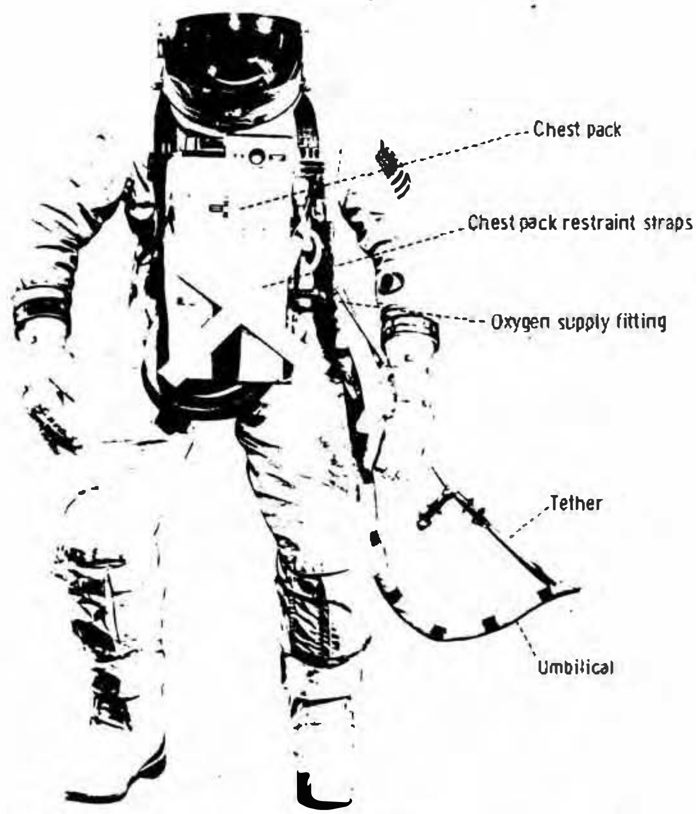

FIGURE 7-4.—Gemini VIII through XII life-support system.

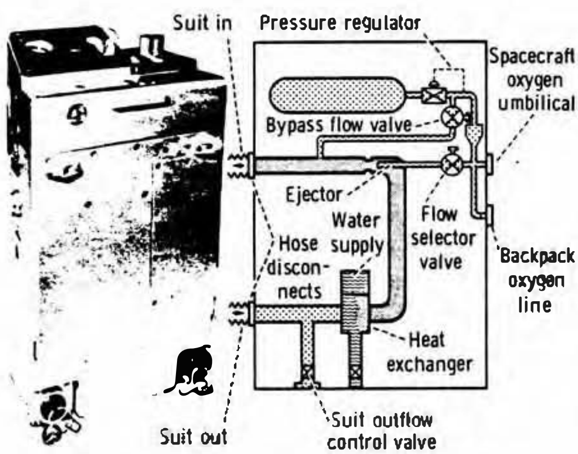

FIGURE 7-5.—Gemini VIII through XII Extravehicular Life-Support System.

ried a self-contained oxygen supply for chest-pack use, which would permit the extravehicular pilot to maneuver detached from the spacecraft oxygen system. The primary oxygen was supplied through a three-position flow-selector valve to an ejector where the 90 psig gas expanded to 4 psia. The gas expansion drove the recirculated secondary vent gas through the heat exchanger of the chest pack. The flow-selector valve permitted the pilot to select a medium or high flow (18 to 22 acfm) depending on cooling requirements. In case of blockage in the ejector, or if additional cooling or carbon-dioxide washout were required, the primary oxygen flow could be bypassed around the ejector through a valve. Suit pressure was maintained at a nominal 3.7 psig by a poppet-type outflow valve. An acceptable carbon-dioxide level was maintained by dumping overboard through the outflow valve an amount of vent gas equal to the amount of primary oxygen introduced to the system through the ejector.

If a leak in the suit loop had developed and caused the suit pressure to drop below 3.4 psig, makeup primary oxygen would have been automatically metered to the system through a demand regulator to maintain suit pressure.

The majority of the cooling for the Extravehicular Life-Support System was provided by the recirculating ventilation gas from the suit passing through an evaporative heat exchanger. In the condenser portion of the heat exchanger, the gas was cooled to approximately 45° F by the evaporation of stored water. Since the gas from the suit was about 85° F with a relative humidity of 85 percent (nominal), this cooling removed the water vapor by condensation. The condensate was then wicked to the evaporative portion of the heat exchanger to provide additional evaporative water. This type of boiling-condensation-reboiling technique is called bootstrapping.

If the normal oxygen flow to the chest pack had been interrupted, decreasing pressure in the umbilical would have automatically actuated a 30-minute emergency supply of oxygen. A visual and audio warning system on the chest pack indicated when oxygen was being used from the emergency supply. Visual and audio warning also denoted decreasing suit pressure. A special regulator acted to maintain suit pressure above 3.3 psi in the event of a suit leak, and the supply to this regulator was arranged such that makeup flow could be drawn from the spacecraft, the self-contained emergency supply, or simultaneously from both sources. Additional warning devices were available if the Astronaut Maneuvering Unit had been used.

Mission Results and Implications

The Gemini IV extravehicular activity lasted 36 minutes, and the pilot reported good thermal control except during high work periods such as ingress. Ingress into the spacecraft and closure of the hatch were difficult tasks, and caused the pilot to become overheated. The Ventilation Control Module System operated within the specified limits; however, high metabolic heat loads could not be sustained because of the inherent limited rate of heat rejection.

The semi-open-loop system was flown on Gemini VIII; however, because of the early termination of the mission, extravehicular activity was not conducted. Gemini IX–A was the first mission to evaluate the performance of the semi-open-loop Extravehicular Life-Support System. Due to the formation of fog on the visor and the resulting reduced visibility, the planned extravehicular activity was not completed. Higher-than-expected workloads were evident throughout the 2 hour 7 minute extravehicular period. The chest pack was designed for a nominal metabolic rate of 1400 Btu/hr and a maximum of 2000 Btu/hr for short periods. Medical data, crew comments, and metabolic simulations all indicated that much higher workloads were experienced. Tests after the mission showed that visor fogging occurred at metabolic rates above 2400 Btu/hr, although no fogging occurred at lower rates. The high rates, in effect, overpowered the capabilities of the evaporator-condenser. Even in medium flow the cooling capability for physiological comfort was adequate, but the evaporator-condenser could not overcome the thermal load sufficiently to prevent fogging. Visor fogging was further induced by high respiration rates (30 to 40 breaths per minute) which humidified 55 to 75 percent of the total gas flow to the helmet to near saturation. This high humidity raised the dewpoint enough so that visor fogging occurred even at normal operating temperatures. The pilot commented that the only time he became uncomfortably warm was during ingress. From this statement and from postflight examination of the evaporator-condenser, it was evident that the evaporator-condenser performance was degraded due to dryout at some period during the extravehicular activity. That period probably occurred very close to ingress.

The Gemini X extravehicular activity was terminated early because of spacecraft problems unrelated to the Extravehicular Life-Support System. Comments by the pilot and the biomedical data gathered during the 39-minute extravehicular activity indicated that the Extravehicular Life-Support System operated completely within specifications.

The Gemini XI extravehicular activity was prematurely concluded after 33 minutes. The pilot stated that the Extravehicular Life-Support System provided adequate cooling; however, the pilot stated that he was fatigued after a relatively brief period of activity outside the spacecraft. Because of a problem in securing the sun visor during the preparations for the extravehicular activity, the pilot experienced high workloads and profuse perspiration. After egress, difficulties involved in the pilot's attempts to attach the extravehicular camera and the spacecraft/target-vehicle tether resulted in high respiration rates and rapid fatigue. It is believed that the chest pack was saturated with warm, moist gas before proper evaporator-condenser operation could reduce the temperatures resulting from the problems before egress.

During the 2 hours 8 minutes of Gemini XII extravehicular activity, the Extravehicular Life-Support System operated completely within specifications. The problem of excess workload was resolved by the use of improved restraints for body positioning and frequent rest periods. This mission proved that at workloads within the design limits, the Extravehicular Life-Support System would function normally, and would provide a comfortable suit environment.

In summary, the Ventilation Control Module System operated satisfactorily within the design capabilities. Other than the possible depletion of heat-exchanger water at the end of Gemini IX-A extravehicular activity, the Extravehicular Life-Support System performed exceptionally well. It is evident, however, that future systems of this type will require increased cooling and metabolic heat-rejection capabilities. Crew comments have also indicated the desirability of eliminating bulky packages from the chest area, and of reducing the volume of self-contained life-support systems. Umbilicals from the spacecraft permit the use of smaller life-support packages, and the use of umbilical systems should be considered for future extravehicular applications.

Umbilicals

Several types of umbilicals have been used in accomplishing the Gemini extravehicular activities. These include the 25-foot umbilical used on Gemini IV, IX-A, and XII; the 50-foot umbilical used on Gemini X; and the 30-foot umbilical used on Gemini XI. Except for the Gemini IV umbilical, which interfaced directly with the space suit, all umbilicals were designed to interface with the Extravehicular Life-Support System chest pack.

The 25-foot umbilical (fig. 7-6) used for Gemini IX-A and XII supplied gaseous oxygen, either directly to the space suit or through the Extravehicular Life-Support System. The 50-foot and 30-foot umbilicals (fig. 7-7) supplied gaseous oxygen only through the Extravehicular Life-Support System and supplied gaseous nitrogen to the Hand Held Maneuvering Unit. The gaseous oxygen was supplied from the spacecraft primary supply at a nominal flow rate of 8 to 9 lb/hr at 90 psia and 65° F. The gaseous nitrogen was supplied from tanks in the spacecraft adapter section (at the inlet to the Hand Held Maneuvering Unit) at a nominal flow rate of 2 lb/min at 75 psia and 0° F.

During the standup extravehicular activity, short hose extensions connected the pilot's space suit to the spacecraft Environmental Control System. In this closed-loop operation, no interface with the Extravehicular Life-Support System was required, and the normal spacecraft ventilation flow rates were provided.

All of the umbilicals were of similar materials and of the same basic design. Each umbilical consisted of wire-reinforced, sili-

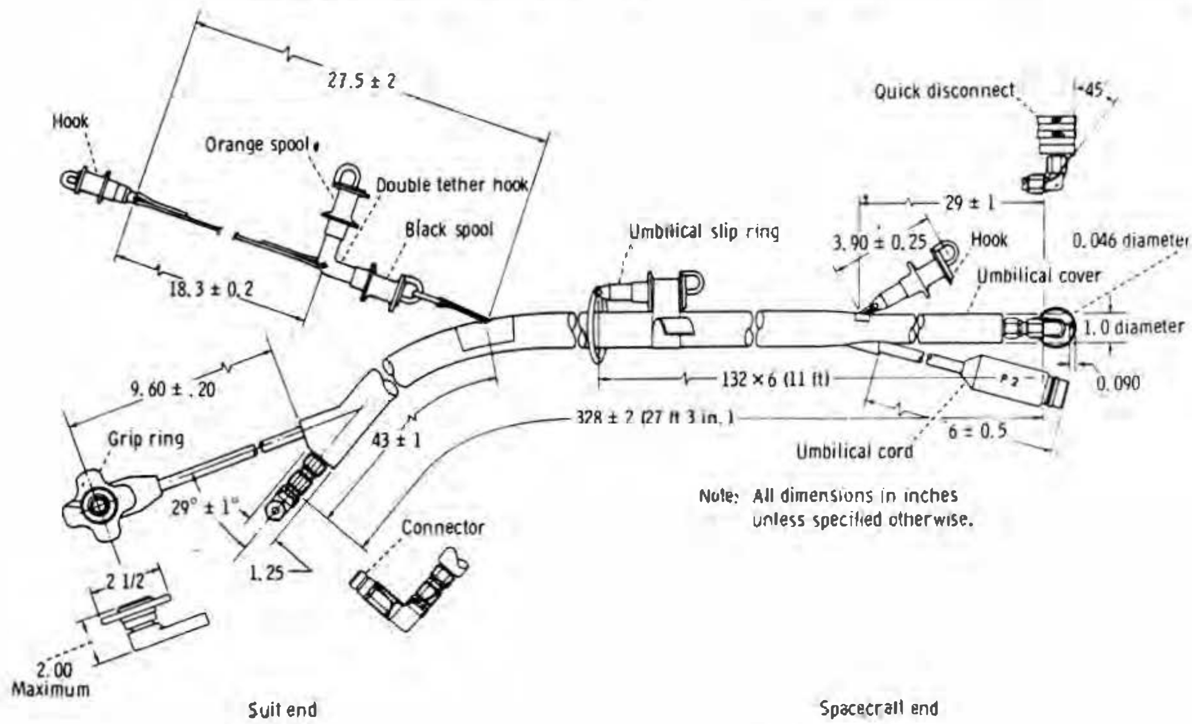

FIGURE 7-6.—Extravehicular Life-Support System, 25-foot umbilical.

cone rubber-lined hose; a 1000-pound test nylon structural tether; and wiring for voice communication, electrical power, and measurements of heart and respiration rates. For the 25-foot umbilical, the oxygen hose was 3/16-inch inside diameter. For the 50-foot and 30-foot umbilicals, the oxygen hose was ¼-inch inside diameter and the nitrogen hose 3/16 inch.

The umbilicals utilized multilayers of Mylar superinsulation for thermal protection. The temperature of gaseous oxygen supplied to the Extravehicular Life-Support System had to be maintained above —15 F to prevent freezing in the ejector. Because of the proximity of the cold nitrogen line to the oxygen line, thermal control was more critical for the 50-foot and 30-foot umbilicals than for the 25-foot umbilical.

The umbilicals were covered with nylon fabric, and chafing protection was provided where required, particularly in the area where the umbilical emerged from the cabin and contacted the hatch sill. The structural tethers were designed so that during the worst conditions of stretch under applied load, no strain was imposed on the oxygen and nitrogen hoses, or on the electrical wiring and connections. In all umbilical designs, the load was transmitted to the spacecraft through a tether attachment point located on the egress handle just inside the cabin. The loads were applied through the parachute harness of the extravehicular pilot. The 25-foot umbilical was attached by a hook to the upper part of the parachute harness; the 50-foot and 30-foot umbilicals were attached to the parachute harness at the pilot's hip.

The extensive test program for the 25-foot umbilical contributed to the development of the 50-foot and 30-foot umbilicals. The materials and the design experience gained from the development of the 25-foot umbilical were used extensively in the fabrication of the longer umbilicals. Based upon the previous experience, the test program was reduced to pressure-temperature performance, leak tests, electromagnetic interference, and

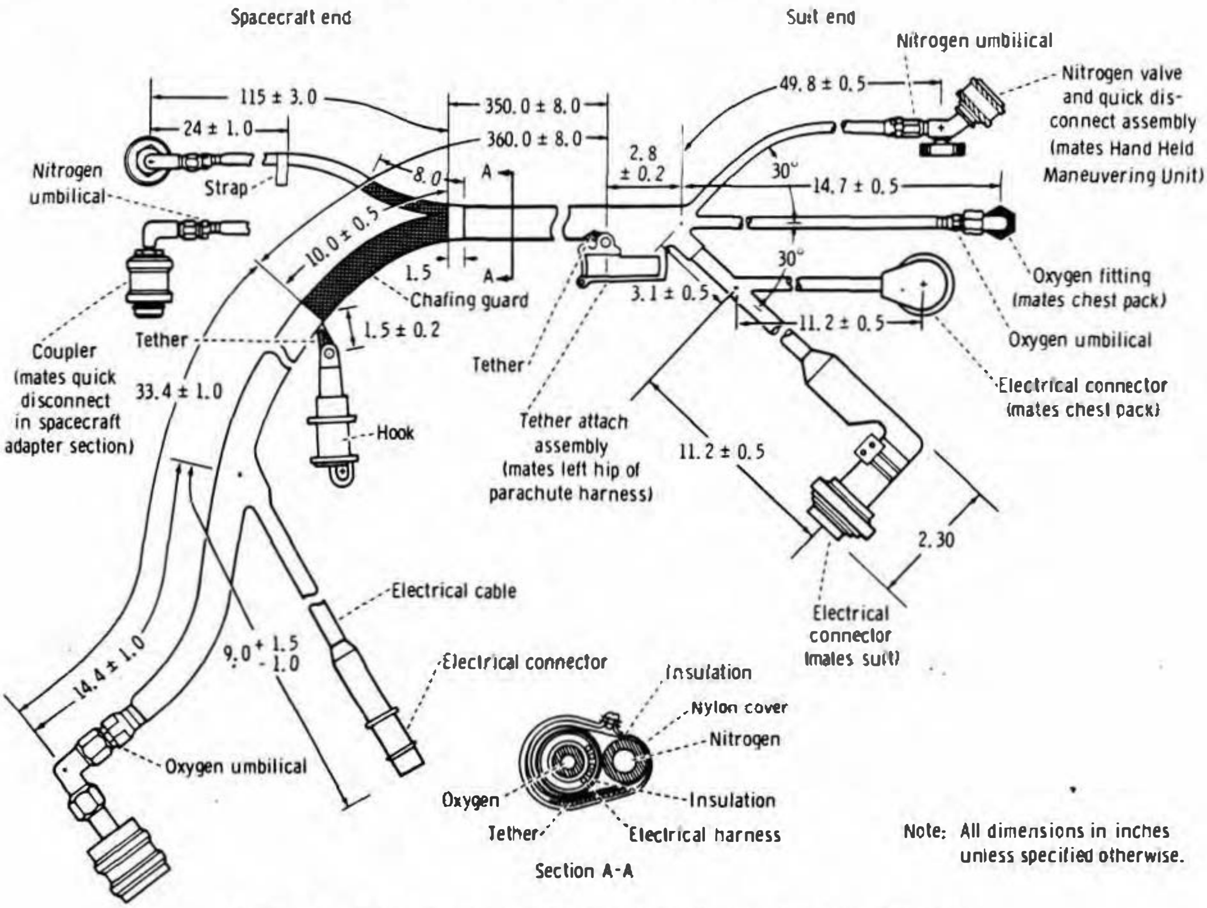

FIGURE 7–7.—Extravehicular Life-Support System, 30-foot umbilical. The 50-foot umbilical is similar.

static and dynamic structural tests. As in the case of the 25-foot umbilical, extensive unmanned altitude-chamber tests were conducted, as well as several manned chamber tests for end-to-end confirmation of the umbilical and the interface with other equipment.

The Gemini Program has shown that extravehicular activity with umbilicals is a useful, operational mode. The umbilical produced no unfavorable torques or forces on the extravehicular pilot; in fact, the pilot was hardly aware of the umbilical. Because of the length and bulk, some difficulty was experienced with the 50-foot umbilical during ingress. Therefore, any umbilical should be kept as small as practicable. Assuming that future spacecraft will be larger than the Gemini spacecraft, umbilical size may not be a problem; however, excessive length would still be undesirable. The donning of the umbilicals proved quite easy and allowed a complete system checkout prior to the extravehicular activity. Incorporation of the propulsion system supply proved satisfactory; this has many possible future uses, such as a power supply for tools.

The umbilical concept is particularly applicable to near-vehicle operations or operations in close quarters where the bulk of a self-contained life-support pack would be undesirable. Umbilical-based life-support systems would be less useful for operations that involved approaching a tumbling vehicle. However, the ease of development and the successful utilization of umbilicals during the

Gemini Program indicate a promising approach to extravehicular activity for future space programs.

Conclusion

The success of the Gemini XII extravehicular activity was largely due to the assimilation of information from preceding flights into a comprehensive program for system testing and flight-crew training. The input to this program from the NASA/Industry Life-Support System Team aided in the generation of extravehicular tasks within a planned time, mobility, and workload envelope.

8. BODY POSITIONING AND RESTRAINTS DURING EXTRAVEHICULAR ACTIVITY

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Summary

One of the foremost conclusions obtained from the experience with extravehicular activity during the Gemini Program was that man's capability to perform work was drastically reduced without the proper restraint provisions. However, with the proper restraint provisions his capability was quite comparable to his 1g capability.

Introduction

This paper describes the body positioning and restraint problems encountered during extravehicular activity in the Gemini Program, and the types of restraint equipment which were used.

The requirement for body restraints during extravehicular activity was indicated on Gemini IV. After depletion of the propellant in his maneuvering unit, the pilot evaluated the umbilical as an aid in body positioning and in moving through space. It was concluded that the umbilical was reliable only as an aid in moving to its origin, and that handholds would be required for other extravehicular maneuvers. The significance of the requirement was emphasized when body-restraint problems contributed to the premature termination of the Gemini IX–A and Gemini XI extravehicular activities. The Gemini XII mission verified that, with adequate restraint provisions, man can perform a great variety of tasks, some of considerable complexity. On Gemini XII, 44 pieces of equipment were provided for extravehicular body restraint in contrast to the 9 pieces of body-restraint equipment provided for Gemini IX–A extravehicular activity.

Control of Body Position

Foot Restraints

The first major work task attempted during Gemini extravehicular activity was the checkout and donning of the Astronaut Maneuvering Unit on Gemini IX–A. The original restraint provisions for this task were two handbars and a horizontal footbar. Velcro on the footbar was intended to mate with Velcro on the pilot's boots; however, the need for additional body restraint for this task was demonstrated in the zero-g airplane (fig. 8–1). A pair of foot restraints was added to the horizontal footbar, and on subsequent flights in the zero-g airplane, checkout of the Astronaut Maneuvering Unit was easily accomplished (fig. 8–2). The pilot would force his feet into the restraints, and the frictional force would contain his feet, allowing him to have both hands free for working.

However, during the Gemini IX–A extravehicular activity, the pilot was not able to maintain body position using only foot restraints. The attempts at two-handed tasks, primarily the tether connections, were exceedingly difficult because every few seconds the pilot had to stop working and use his hands to regain proper body position. The foot restraints were even less satisfactory when unstowing the Astronaut Maneuvering

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FIGURE 8-1.—Donning of Astronaut Maneuvering Unit without foot restraints.

FIGURE 8-2.—Donning of Astronaut Maneuvering Unit using foot restraints.

Unit controller arms. When the pilot bent forward and applied a downward force to the controller arm, he created a moment which forced his feet out of the restraints. The inadequacy of the foot restraints caused the pilot to exert a continuous high workload to maintain control of body position, in addition to the work involved in performing the tasks. Heat and perspiration were produced at a rate exceeding the removal capability of the life-support system, and fog began accumulating on the space-suit visor. This fogging progressed until the pilot's vision was almost totally blocked, forcing him to abandon his attempts to don and use the Astronaut Maneuvering Unit.

As a result of this experience during Gemini IX–A, new requirements for foot restraints were developed and the investigation of underwater simulation of zero-g was initiated. Numerous equipment modifications were also incorporated to simplify the extravehicular activity tasks on subsequent missions.

Analysis of the Gemini IX–A body-restraint problem resulted in the following criteria for design of new foot restraints: motion must be restrained in all 6 degrees of freedom, and restraint of the feet must involve no mechanical devices. Molded fiberglass foot restraints incorporating these features were designed for the Gemini XI and XII spacecraft. The restraints were custom fitted to the pilot for each flight, and were mounted on a platform attached to the inside surface of the spacecraft adapter equipment section (fig. 8–3). During the zero-g airplane training, the Gemini XI and XII flight crews used and evaluated the foot restraints and found them completely adequate for all tasks envisioned. The Gemini XII flight crew also trained with the restraints in the underwater zero-g simulation facility with the same results.

Underwater Zero-Gravity Simulation

The initial evaluation of the underwater zero-g simulation was conducted by the

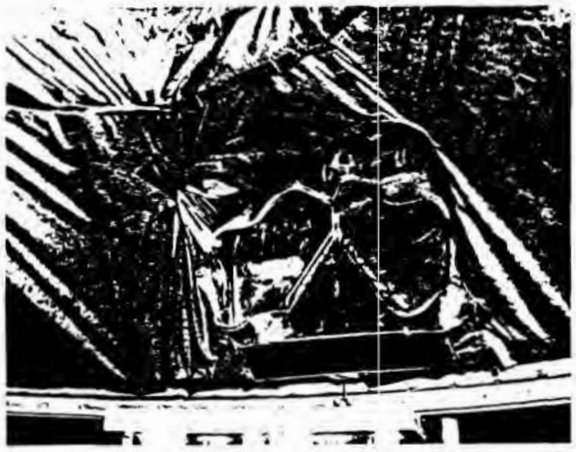

FIGURE 8-3.—Foot restraints used during Gemini XII extravehicular activity.

Gemini IX–A pilot shortly after the mission. The configuration of the mockup equipment was similar to that of the Gemini IX–A spacecraft, and the pilot repeated the Astronaut Maneuvering Unit checkout and donning procedures previously attempted in flight. The pilot concluded that the underwater zero-g simulation very nearly duplicated the actual weightless condition and the accompanying problems experienced in flight. The extravehicular tasks planned for Gemini X, XI, and XII were then performed in the underwater zero-g simulation, and recommendations were made concerning the required restraints and the feasibility of proposed tasks. Underwater simulation of zero-g has great applicability to extravehicular activities, particularly to the problems of body positioning and restraints.

Handholds and Tether Devices

Several restraint problems were encountered during Gemini X extravehicular activity, but performance of the planned tasks was not seriously affected. The pilot had difficulty controlling his body position while using the edge of the target-vehicle docking cone as a handrail to move to the area of the Experiment S010 Agena Micrometeorite Collection package. Attachment of the umbilical nitrogen fitting was also a difficult task because the handrail provided for restraint did not properly deploy. The tasks were accomplished with one hand, while the other hand was used for restraint.

For the Gemini XI mission, the tether for the spacecraft/target-vehicle tether evaluation was assembled and stowed so that the pilot could attach it to the spacecraft docking bar with one hand. With the other hand, the pilot could use one of the three handholds on the back surface of the docking cone for maintaining his position. However, the pilot had trained to have both hands free, and he had been able to wrap his legs around the spacecraft nose and wedge his legs into the docking cone. The pilot could force himself into position by arm force using the handholds provided. In the zero-g airplane, the task was so easy that the pilot was able to move from the hatch, force himself into the restrained position, and make the complete tether hookup in a single parabola (about 30 seconds). In flight, however, the restraint technique proved extremely difficult, and the pilot expended a great deal of energy during the 6 minutes that were required to move from the hatch and make the tether hookup. This was the major factor in his inability to continue the flight plan for the extravehicular activity. As in the case of the Gemini IX–A pilot, the prime expenditure of energy by the Gemini XI pilot was the continuous struggle to maintain body position in order to perform the required tasks. Apparently, the frictional forces exerted by the pilot in wedging his legs into the docking cone were not sufficient to overcome the tendency of the pressurized suit to expand and push him out of the docking cone.

As a result of this experience, it was decided that the Gemini XII flight crew would include underwater zero-g simulation in the training for extravehicular activity. As a result of the problems encountered during Gemini extravehicular activities, the extravehicular objective for Gemini XII was changed to an evaluation of body restraints instead of the evaluation of the Astronaut Maneuvering Unit. The objective of the restraint evaluation was to determine what type of restraints were required for representative extravehicular tasks.

Restraint Equipment

The use of restraint devices for extravehicular activity on the Gemini Program is summarized in table 8–I. Descriptions of these devices and results of their use follow.

Rectangular Handrail

Two rectangular handrails (fig. 8–4) were installed along the spacecraft adapter section to assist the extravehicular pilot in moving from the cockpit to the adapter equipment section where various tasks were to be performed; for example, donning the Astronaut Maneuvering Unit. The handrails were flush with the spacecraft surface at launch, and were 1.5 inches above the spacecraft surface when deployed. The aft handrail deployed automatically when the spacecraft separated from the launch vehicle. The forward handrail was manually deployed by the extravehicular pilot.

The Gemini IX–A and XII pilots used the handrails to travel the 8 feet from the cockpit to the aft end of the spacecraft. The limited suit mobility and interference by the life-support system chest pack required the pilots to traverse the handrail by moving the hands one after the other to the side, rather than hand over hand. The Gemini X pilot used the handrail to travel from the hatch to the end of the adapter retrograde section and return, and then as a handhold while making and breaking the nitrogen connection on the 50-foot umbilical. Comments by the pilots indicated that the configuration of this handrail was the best for travel between two points on the spacecraft surface. A rectangular, rather than a cylindrical, cross section

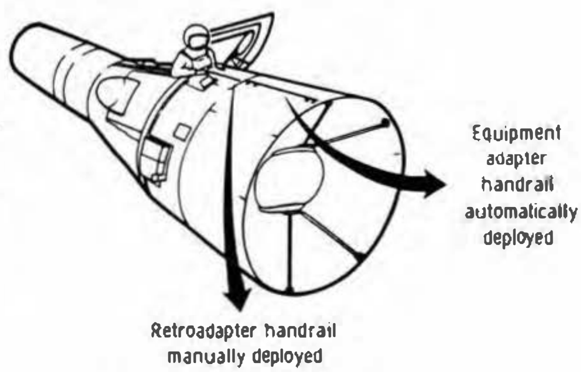

FIGURE 8–4.—Extendable handrails.

TABLE 8–I.—*Restraint Devices Used During Gemini Extravehicular Activities*

Restraint device configuration	Gemini mission			
	IX–A	X	XI	XII
Rectangular handrail	X	X	X	X
Large cylindrical handrail (1.38 in. dia)	X			X
Small cylindrical handrail (0.317 in. dia)				X
Telescoping handrail				X
Fixed handhold			X	X
Rigid Velcro-backed portable handhold				X
Flexible Velcro-backed portable handhold	X			
Waist tethers				X
Pip-pin handhold/tether-attach device				X
Pip-pin antirotation device				X
U-bolt handhold/tether-attach device				X
Foot restraints	X			X
Standup tether		X	X	X
Straps on space-suit leg			X	X

was preferred because the rectangular shape offered more resistance to rotation for a given hand force, and allowed better control of body attitude. In a pressurized Gemini suit, the width of the rectangular handrail (1.25 inches) was a good size for gripping.

Large Cylindrical Handrails

A pair of large cylindrical handrails (fig. 8-5) was furnished in the adapter equipment section to permit the pilot to move from the rectangular handrails to the work area, and to provide restraint while positioning his feet in foot restraints or while working. The two handrails were symmetrically located on each side of the work station. Although the pilots indicated a preference for rectangular cross section, they were able to use the cylindrical handrails to introduce the significant body torques required to position their feet in the foot restraints. The diameter (1.38 inches) of the cylindrical handrails was the most favorable size.

Small Cylindrical Handrails

There were two segments of small cylindrical handrails (figs. 8-6 and 8-7) rigidly mounted on the forward surface of the cylindrical portions of the Target Docking Adapter on the Gemini XII target vehicle. The handrails were small enough to be used as waist tether-attach points, as well as for handholds. Although the handrail was not evaluated extensively, the configuration was usable as a handhold, and the pilot considered the size a good feature since it permitted direct attachment of the waist tethers.

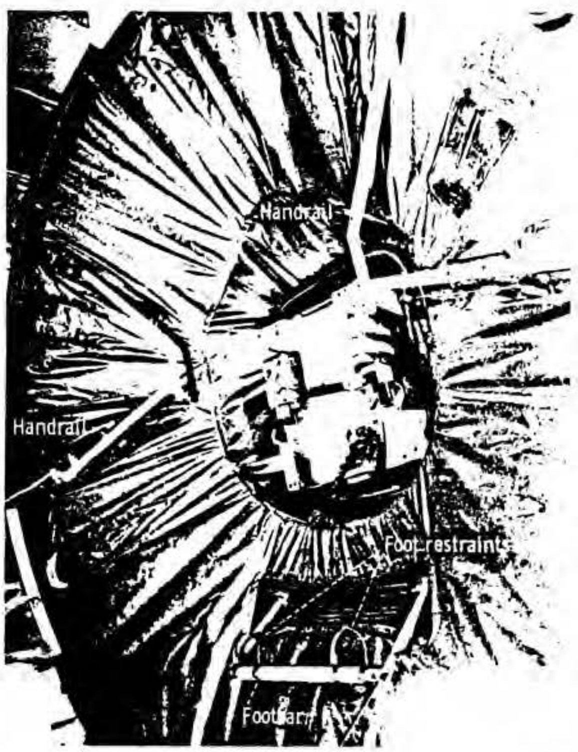

FIGURE 8-5.—Handrails and foot restraints in the Gemini IX-A spacecraft adapter equipment section.

FIGURE 8-6.—Handrail on left side of target vehicle.

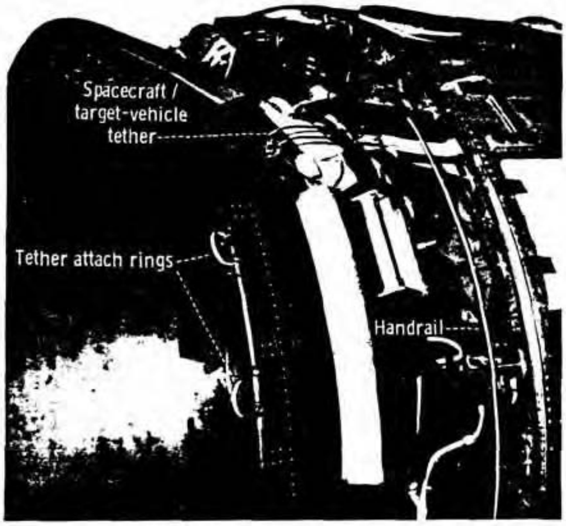

FIGURE 8-7.—Handrail on right side of target vehicle.

Telescoping Cylindrical Handrail

The Gemini IX-A and XI pilots used the spacecraft Reentry Control System thrusters as handholds for travel from the spacecraft hatch to the spacecraft nose; however, the thrusters were neither well located nor easy to use for that purpose. On each of these missions, the extravehicular pilot went over the top of the docking bar on his first attempt to propel himself from the thrusters to the spacecraft nose.

During Gemini XII, the telescoping handrail (figs. 8-8 and 8-9) solved the problem of travel from the spacecraft hatch to the spacecraft nose. The telescoping handrail was stowed in the compressed condition near the hinge of the right hatch, located above the pilot's right shoulder. After the cabin was depressurized and the hatch was opened for standup extravehicular activity, the pilot unstowed and manually extended the handrail. The pilot then installed the small end of the handrail in a special receptacle in the target-vehicle docking cone, and the large end on a mounting bolt in the spacecraft center beam, between the hatches. During the umbilical extravehicular activity, the pilot used this handrail for two round trips between the spacecraft hatch and the spacecraft nose, and as a handhold for several changes in body attitude. The nonrigidity of the handrail was considered undesirable by the pilot; when the handrail flexed, the pilot no longer had absolute control of body position and attitude. While attaching the spacecraft/target-vehicle tether, the pilot also used the ring on the telescoping handrail for a waist tether-attach point. At the conclusion of the umbilical extravehicular period, the pilot removed and jettisoned the handrail.

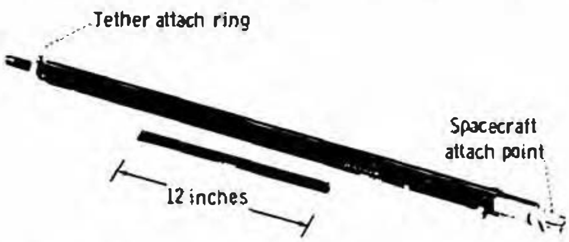

FIGURE 8-8.—Telescoping handrail compressed.

FIGURE 8-9.—Telescoping handrail attached to vehicles.

Fixed Handhold

Three fixed handholds (fig. 8–10) were provided on the back of the docking cone on the Gemini XI target vehicle to provide restraint for the spacecraft/target-vehicle tether attachment. Two identical handholds were provided on the back of the docking cone on the Gemini XII target vehicle. The handholds proved very useful in flight, and the friction coating was a good feature.

Flexible Velcro-Backed Portable Handhold

Flexible Velcro-backed portable handholds (fig. 8–11) were evaluated as restraints and as maneuvering aids during the Gemini IX–A mission. Two fabric-backed nylon Velcro pile pads were carried in the spacecraft. The pilot attached the pads to his gloves with an elastic strap wrapped around the palms of the hands. There were about 80 patches of nylon Velcro hook on the surface of the spacecraft to engage the pile handholds. Some of the significant results included the following: (1) the elastic attachment was not adequate, as one of the handholds was pulled off his glove; (2) the contact forces were not sufficient to accommodate controlled maneuvering or control of body attitude, but were sufficient for station keeping; (3) the unprotected Velcro hook on the spacecraft nose was degraded by launch heating.

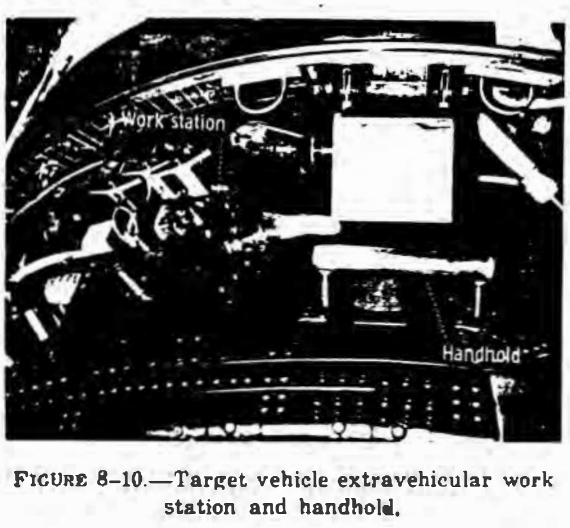

FIGURE 8–10.—Target vehicle extravehicular work station and handhold.

FIGURE 8–11.—Flexible Velcro-backed portable handhold.

Rigid Velcro-Backed Portable Handhold

For Gemini XII, four trowel-shaped, rigid, Velcro-backed, portable handholds (fig. 8–12) were installed in the extravehicular work areas. The handholds were coated with resilient material, with a tether-attach ring at one end. Two of the handholds had about 9 square inches of nylon-pile Velcro, and two had about 16 square inches of polyester-pile Velcro. The handholds were stowed for launch on a surface of hook Velcro and further restrained by a pip-pin device. Four areas of polyester hook Velcro on built-up flat surfaces were located on the target vehicle to engage the Velcro on the handholds. Polyester Velcro has greater adhesive force than nylon Velcro, and does not require protection from launch heating.

Detailed evaluations of the rigid Velcro-backed portable handholds were not included in the flight plan for Gemini XII extravehicular activity. Analyses and simulations indicated a number of limitations concerning

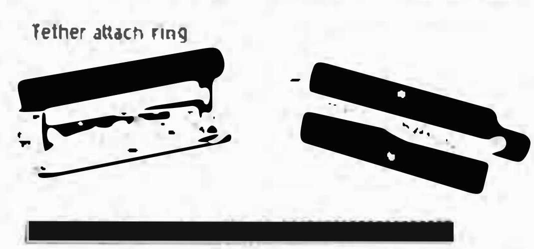

FIGURE 8–12.—Rigid Velcro-backed portable handhold.

the usefulness of the devices. For example, best utilization requires that the Velcro be placed in shear rather than tension, and this complicates the usage. Also, the restraint force should be significantly greater than the required applying force; this is not true of nylon Velcro. Polyester Velcro is better, but has not been evaluated as thoroughly as the nylon. The use of steel Velcro would make these devices feasible, but the potential hazard to the space suit is not tolerable at this time.

Waist Tethers

The Gemini XII waist tethers (fig. 8–13) were made of stiff nylon webbing with a length-adjustment buckle and a large hook for attachment to the various tether-attach rings. The waist tethers were looped around the pilot's parachute harness and were fastened with two large snaps. A large fabric tab was provided to facilitate opening the snaps of a pressurized suit. A D-shaped ring was provided for making length adjustments, and was used several times by the pilot. The adjustment buckle, a conventional single-loop buckle, allowed length adjustment between approximately 32 and 21 inches.

The tether attachment to the pilot, slightly below waist level, was considered well located by the pilot. A special device, consisting of a thin metal plate with a ring on each end for attaching the waist tether hooks, was provided to restrain the waist tethers while not in use. The device was slightly longer than the front width of the life-support system chest pack and was attached with Velcro. The pilot used a variety of devices for attaching the tethers in the spacecraft adapter section and on the target vehicle. The pilot used about six different pairs of tether-attach points which had been selected during training. At one time, because of the lack of good control of body attitude, the Gemini XII pilot experienced a slight difficulty in moving a tether to a new attach point. With one hand occupied in making a waist tether attachment, the pilot had to use the other hand to control body attitude. Therefore, a pair of handholds or other restraints near each pair of tether-attach points was desirable. Also, it was determined that the waist tether-attach points should be as far apart as possible, consistent with the pilot's reach in the pressurized suit. The attachments were easier to make when the attach points were located at the pilot's sides rather than directly in front of him; and torques were cancelled better with widespread tether-attach points. The pilot observed that few adjustments were required to the tether length; consequently, provisions for adjustments could be eliminated from future body tethers.

With only the waist tethers for restraint, the pilot was able to use a conventional torque wrench to install and tighten a bolt to about 200 inch-pounds on the spacecraft adapter section work station (fig. 8–14). Again, with only the waist tethers for restraint, the pilot was able to pull nylon Velcro pile strips 4

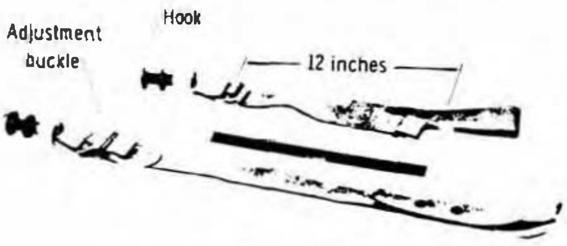

FIGURE 8–13.—Waist tethers.

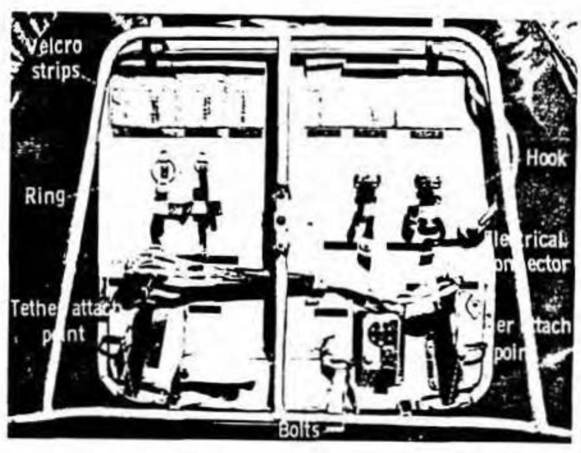

FIGURE 8–14.—Gemini XII extravehicular adapter work station.

inches long and 5 inches wide from both nylon and steel Velcro hook, and to disconnect and reconnect three electrical connectors. The pilot also made a variety of hook and ring connections, including hooks and rings of the same sizes which had proved impossible for the Gemini IX–A pilot to connect.

The waist tethers, when attached to the tether-attach points on the target vehicle (fig. 8–15), provided the required restraint for the Gemini XII pilot to attach the spacecraft/target-vehicle tether; activate the Experiment S010 Agena Micrometeorite Collection package; and disconnect and connect fluid connectors and an electrical connector. The pilot used the Apollo torque wrench to exert greater than 100 inch-pounds of torque; he concluded that man's capability is even greater, and could be determined in the underwater zero-g simulation. The pilot was able to perform these tasks with one waist tether attached and one hand on a handhold, and then to repeat the tasks without using waist tethers. He strongly recommended, however, that body tethers be included in the restraint systems for future tasks involving torque. It is probable that body tethers will provide a greater capability for applying torque; minimize the effort required in controlling body position; and, if a tool should slip, eliminate the possibility of it drifting away.

One of the best features of body tethers is the elimination of the constant anxiety of drifting into an unknown and uncontrolled body position, while performing work or while resting. The waist tethers permitted the Gemini XII pilot to relax completely during the designated rest periods and at any other desired time. During previous umbilical extravehicular activity, the pilots had been required to hang on with one or both hands and rest, as well as possible, in this condition. Of course, the work required to control body position eliminated the possibility of complete rest.

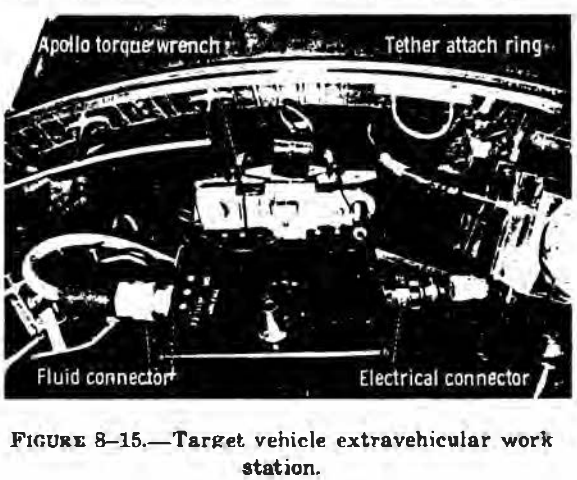

Figure 8–15.—Target vehicle extravehicular work station.

Pip-Pin Handhold/Tether-Attach Devices

Seven pip-pin handhold/tether devices (fig. 8–16) were used during Gemini XII. These devices used a conventional pip-pin mechanism with ball detents for attachment to the spacecraft. The T-shape of the pip-pins facilitated their use as handholds, and a loop was installed for tether attachment. The pilot used the devices as handholds during changes in body position and as waist tether-attach points during some of the work tasks on the target vehicle.

The T-shaped pip-pins were a convenient shape and size for hand gripping. When the rotational freedom of the devices was removed, the devices made excellent handholds, and allowed complete control of body attitude. The elimination of rotational freedom also made waist tether attachment much easier.

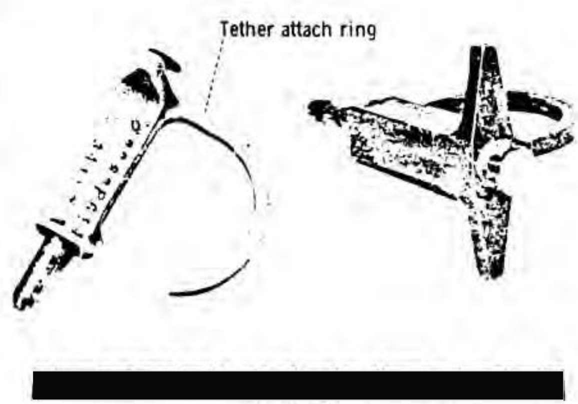

Figure 8–16.—Pip-pin device.

Pip-Pin Antirotation Devices

The pip-pin antirotation devices (fig. 8–17) were installed over 11 of the pip-pin attachment holes. Without the antirotation device, the pip-pins were free to rotate, and would do so when given any small torque. Experience during Gemini XII showed that the antirotation devices were valuable when the pilot applied torque to the pip-pins, such as performing most tasks while tethered. However, with the antirotation device in place, the pip-pins had to be installed in one of eight specific orientations, which complicated the installation. Therefore, if pip-pin devices of this type are to be used, antirotation devices are very desirable, but the requirement for such precise alinement is undesirable.

U-Bolt Handhold/Tether-Attach Devices

Nine U-bolt handhold/tether-attach devices (fig. 8–18) were installed in the extravehicular work areas on Gemini XII. The pilot used two of the U-bolts installed in the spacecraft adapter as waist tether points during the work without foot restraints, but the close proximity (about 4 inches) to the bolt platform caused some inconvenience during the bolt torquing. The pilot found the U-bolts on the target vehicle useful for waist tether attachment and as handholds during work tasks and position changes.

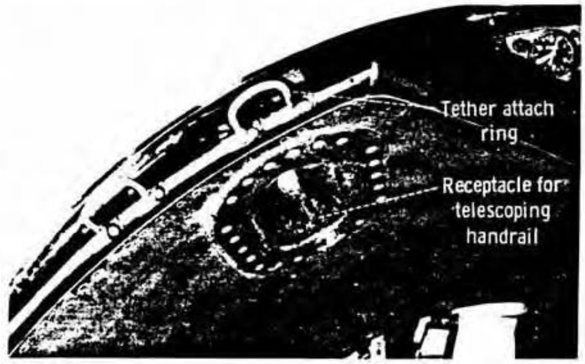

FIGURE 8–18.—Extravehicular restraint provisions on target vehicle docking cone.

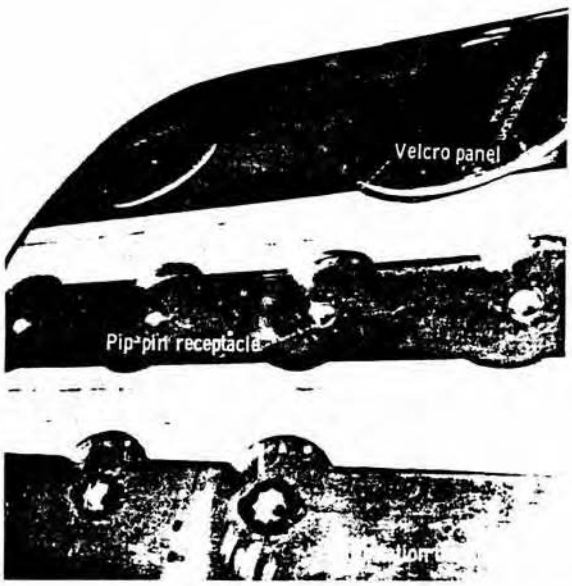

FIGURE 8–17.—Pip-pin and Velcro attachment points.

Foot Restraints

The Gemini IX–A foot restraints (fig. 8–5) were not adequate for body restraint even in the absence of external forces. The molded foot restraints on the Gemini XII spacecraft, however, were considered by the pilot to be far superior to all other restraint devices he evaluated. With his feet in these restraints (fig. 8–19), the pilot was able to nearly duplicate his 1g proficiency in performing tasks. He applied torques in excess of 200 inch-pounds, and performed alinement (fluid connector) and cutting operations. In addition to performing work tasks, the Gemini XII pilot evaluated the body-attitude constraints imposed by the foot restraints. The pilot was able to force himself backward (pitch up) about 90°; however, a significant effort was required to maintain that position. He was able to roll ±45°, and his yaw capability was almost ±90°.

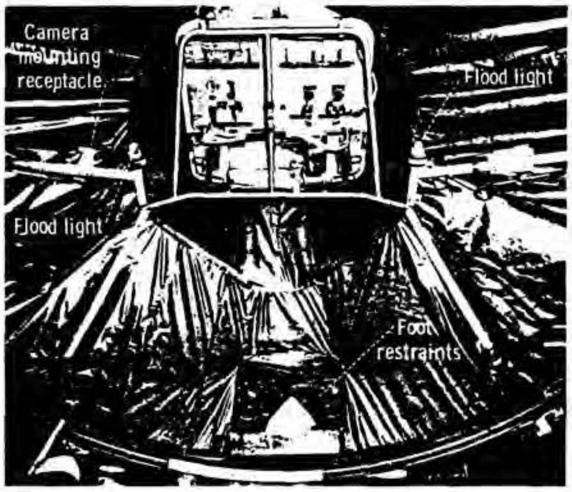

FIGURE 8-19.—Gemini XII adapter provisions for extravehicular activity.

Standup Tether

To prevent stressing the pilot's oxygen and electrical connections with the spacecraft, standup tethers (fig. 8-20) were used during the standup extravehicular activity on Gemini X, XI, and XII. The standup tethers were attached to the extravehicular pilot's parachute harness and to the left side of the pilot's seat. The tethers were constructed of thin nylon webbing and had a conventional single-loop adjustment buckle. The command pilot held the free end of the tether and usually performed the required adjustments, although on Gemini XII the extravehicular pilot was also able to make adjustments.

Space-Suit Leg Straps

For Gemini XI, a strap (fig. 8-21) about 9 inches in length was sewed on the left leg

FIGURE 8-20.—Standup tether.

FIGURE 8-21.—Space-suit leg strap.

(in the calf area) of the pilot's space suit. When not in use, the strap was folded inside a Velcro pocket on the space suit. During the umbilical extravehicular activity, with the pilot standing in the seat, the command pilot opened the Velcro pocket and pulled out the strap. The strap was intended to serve the same purpose during umbilical extravehicular activity that the standup tether served during the standup extravehicular activity.

On the Gemini XII mission, identical straps were sewed on both legs of the pilot's space suit. The straps were not used, however, because the command pilot found it easier to hold the pilot's foot to secure him.

Concluding Remarks

Provision of adequate body restraints is one of several factors which can assure the success of an extravehicular activity mission.

Based on the extravehicular experience accumulated in the Gemini Program, it was concluded that thorough analysis and detailed training for extravehicular activity must be continued, and that the body-restraint requirements indicated by the analysis and the training must be met. During the extravehicular activity, restraints must be provided for rest as well as for work tasks.

The restraints that were found to be most satisfactory during the Gemini Program included:

(1) Gemini XII foot restraints, for rest and localized work

(2) Gemini XII waist tethers, for rest and localized work

(3) Rectangular handrail, for translating across a spacecraft surface

(4) Pip-pin devices, for combination tether-attach points and handholds

9. EXTRAVEHICULAR MANEUVERING ABOUT SPACE VEHICLES

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Introduction

The purpose of this report is to summarize what has been learned from the Gemini Program concerning extravehicular maneuvering in the near vicinity of the spacecraft. Maneuvering with the Hand Held Maneuvering Unit was scheduled for the Gemini IV, VIII, X and XI missions, and with the Astronaut Maneuvering Unit for the Gemini IX–A and XII missions.

The evaluations of the maneuvering equipment planned for Gemini VIII, IX–A, X, and XI were not completed because of problems with spacecraft equipment before the evaluations were scheduled. Because of increased emphasis on the evaluation of body-restraint problems, the Astronaut Maneuvering Unit was not carried on Gemini XII.

Even though only limited extravehicular maneuvering was accomplished during the Gemini Program, a number of significant maneuvering systems were readied for flight and were actually carried into space. One purpose of the first portion of this report is to describe, in general, the maneuvering equipment used for extravehicular activity during the Gemini Program. The second portion describes the ground training equipment and the methods used in preparing the flight crews for extravehicular maneuvering. The third portion recounts the brief, but interesting, flight results obtained with the Hand Held Maneuvering Unit during Gemini IV and Gemini X, and draws a comparison between flight performance and ground training indications.

Gemini Extravehicular Maneuvering Units

Prior to the development of the Hand Held Maneuvering Unit utilized on the Gemini IV mission, several experimental hand-held gas-expulsion devices were evaluated at the Air Bearing Facility, Manned Spacecraft Center. While working with the early Hand Held Maneuvering Units, some preconceived ideas were abandoned and some new ideas were generated. The following were learned from the early concepts:

(1) For translating, the tractor mode was inherently stable and easiest to control.

(2) Tractor nozzles placed far apart and parallel provided much less gas-impingement loss than nozzles placed side by side and canted outward.

(3) Due to lack of finger dexterity in pressurized gloves, the trigger operating the pusher and tractor valves had to be operated by gross movements of the hand as opposed to finger or thumb manipulation.

(4) Because of the constraints placed on arm and hand movement by the pressurized suit, together with the need to easily aline the thrust with the operator's center of gravity, the handle of the space gun had to be on top, and certain angles had to be built into the Hand Held Maneuvering Unit to insure easy aiming of thrusters when the pilot's arm and the hand were in a natural hard-suit position.

(5) Precise attitude control was enhanced by utilizing a proportional thrust system, rather than an off-on system, for controlling thrust level.

Gemini IV Hand Held Maneuvering Unit

The configuration for the Gemini IV Hand Held Maneuvering Unit (fig. 9–1) was evolved from early concepts, mission requirements, and available qualified components. The 4000-psi storage tanks were the same as the emergency bailout bottles used in the Gemini ejection seat. The pressure regulator had been used in the Mercury Environmental Control System. A summary of the operating characteristics of the Gemini IV maneuvering unit is provided in table 9–I, and a cutaway drawing is shown in figure 9–2.

Mission requirements dictated that the Hand Held Maneuvering Unit be stowed inside the spacecraft cabin. This required the selection of a propellant gas which would not contaminate the spacecraft atmosphere if leakage occurred: oxygen in the gaseous form was chosen as the propellant. Since very limited storage space was available, the Hand Held Maneuvering Unit was stowed in two sections: the handle assembly and the high-pressure section. The two sections were joined by connecting a coupling at the regulator and inserting a pin adjacent to the pusher nozzle (fig. 9–2).

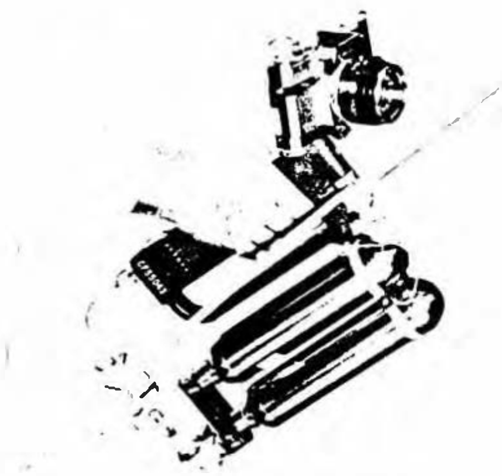

FIGURE 9–1.—Gemini IV Hand Held Maneuvering Unit showing hand position for tractor thruster application.

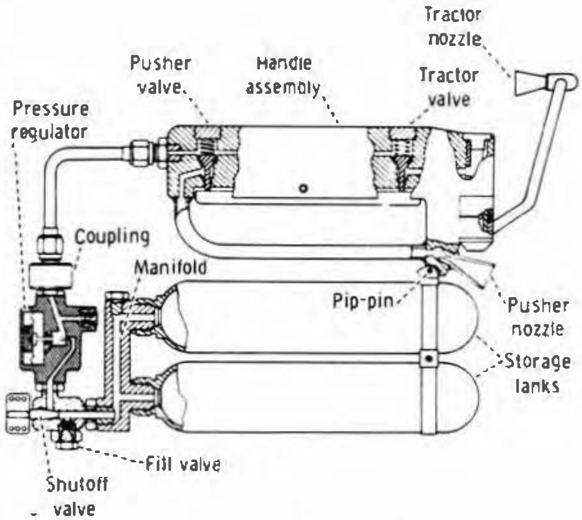

FIGURE 9–2.—Cutaway drawing of Gemini IV Hand Held Maneuvering Unit.

TABLE 9–I.—*Gemini IV Hand Held Maneuvering Unit Characteristics*

Thrust, lb	0 to 2
Total impulse, lb×sec	40
Total available ΔV, ft/sec	6
Trigger preload, lb	15
Trigger force at maximum thrust, lb	20
Storage-tank pressure, psi	4000
Regulated pressure, psi	120
Nozzle-area ratio	50:1
Empty weight, lb	6.8
Oxygen weight, lb	0.7
Gross weight, lb	7.5

After gaseous oxygen left the 4000-psi storage tanks (fig. 9–2), it passed through a manifold to a shutoff and fill valve. When this valve was opened, the oxygen entered a pressure regulator which reduced the pressure to 120 psi. The low-pressure oxygen entered the handle of the Hand Held Maneuvering Unit and passed through a filter to two valves. The valve located at the rear of the handle permitted the gas to flow through the trigger guard to the pusher nozzle. The valve located at the forward end of the unit ported gas through a swivel joint, then through two arms to the tractor nozzles. The arms of the tractor nozzles folded back for compact storage. The pusher and tractor valves were actuated by depressing the trigger. The amount of force applied to the pusher or tractor valve determined the thrust level. A force of 15 pounds applied to the valve poppet initiated gas flow to the nozzle; as the force was increased to 20 pounds, the thrust level increased proportionately from 0 to 2 pounds.

The gas storage tanks held only 0.7 pound of oxygen. This provided a total impulse of 40 lb $\times$ sec, or 2 pounds of thrust for 20 seconds. If used continuously, this total impulse would accelerate the extravehicular pilot and the life-support system (215 pounds) to a velocity of 6 ft/sec.

Gemini VIII Hand Held Maneuvering Unit

In the Gemini VIII mission, the total impulse was increased to 600 lb $\times$ sec (15 times more than the Gemini IV unit). A summary of the Gemini VIII maneuvering system characteristics is given in table 9–II. Eighteen pounds of Freon 14 gas were stored at 5000 psi in a 439-cubic-inch tank. The tank was mounted in a backpack (fig. 9–3) which also housed an identical tank filled with 7 pounds of life-support oxygen. Freon 14 was chosen as a propellant because, even though its specific impulse (33.4 seconds) was lower than oxygen (59 seconds) or nitrogen (63 seconds), its density was almost three times as great, therefore providing more total impulse for a slight increase in total mass. This can be illustrated by the following calculations:

$$7 \text{ lb } O_2 \times 59 \text{ lb} \times \text{sec/lb} = 413 \text{ lb} \times \text{sec total impulse}$$

$$18 \text{ lb Freon } 14 \times 33.4 \text{ lb} \times \text{sec/lb} = 600 \text{ lb} \times \text{sec total impulse}$$

The calculations indicate a 45-percent increase in total impulse for Freon 14 over oxygen at the same maximum tank pressure (5000 psi). Inasmuch as the weight of the extravehicular pilot with all gear except propulsion gas was about 250 pounds, the use of Freon 14, rather than oxygen or nitrogen, was an excellent tradeoff as far as the change-in-velocity capability was concerned.

TABLE 9–II.—*Gemini VIII Hand Held Maneuvering Unit Characteristics*

Propellant, gas	Freon 14
Thrust, lb	0 to 2
Specific impulse (calculated), sec	33.4
Total impulse, lb $\times$ sec	600
Total available ΔV, ft/sec	54
Trigger preload, lb	15
Trigger force at maximum thrust, lb	20
Storage-tank pressure, psi	5000
Regulated pressure, psi	110$\pm$15
Nozzle-area ratio	50:1
Weight of propellant, lb	18
Weight of Hand Held Maneuvering Unit, lb	3

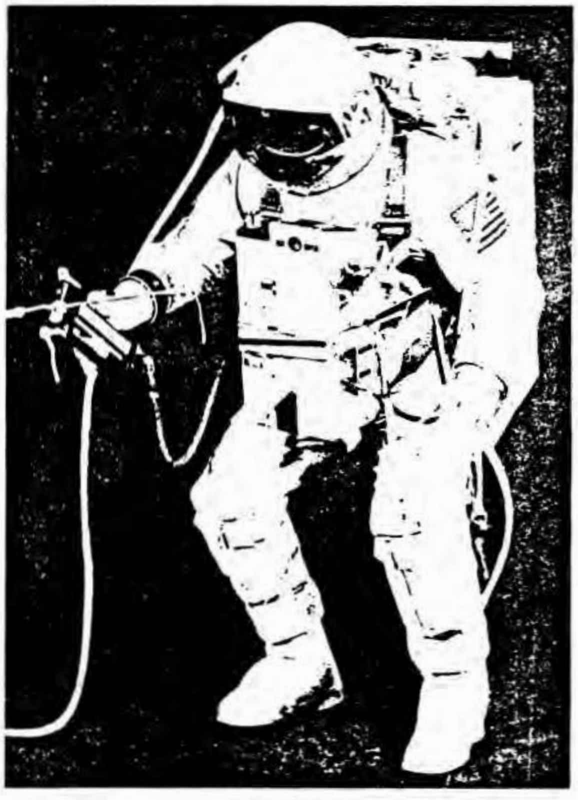

FIGURE 9–3.—Gemini VIII Hand Held Maneuvering Unit, backpack, and chest pack.

The expansion of the Freon 14 from 5000 psi to 110 psi resulted in temperatures of approximately —150° F in the Hand Held Maneuvering Unit handle assembly. The low temperatures caused the poppet valves to stick open when actuated. To make the valves operable at —150° F, Teflon cryogenic seals were used in place of the elastomer seals which had been satisfactory for the Gemini IV Hand Held Maneuvering Unit. Even though qualification testing demonstrated that the redesigned poppet valves would operate at low temperatures, two shutoff valves were incorporated in the system. One of the valves (fig. 9-4) was located immediately upstream of the coupling, and was designed to prevent the gas from escaping in case the poppet valves failed to close. The other shutoff valve was located in the backpack, upstream of the flexible feedline and was designed to shut off the gas flow in the event of an accidentally severed hose. The extra precautions were taken to reduce the possibility of uncontrolled gas escaping from the system and causing the extravehicular pilot to tumble. The handle of the Hand Held Maneuvering Unit was also modified to provide the pilot with a better grip (fig. 9-4).

Gemini X Hand Held Maneuvering Unit

For Gemini X, the handle of the Hand Held Maneuvering Unit (fig. 9-5) was further modified by sloping the handle to provide easier movement of the pilot's hand from pusher to tractor actuation. Grooves were cut in the handle to accommodate the restraint wires in the palm of the suit glove. The single rocking trigger was replaced with two shorter triggers pivoted at the end. This modification reduced the actuation forces from between 15 and 20 pounds to between 5 and 8 pounds, and also reduced the distance the hand had to be shifted to go from pusher to tractor mode or vice versa.

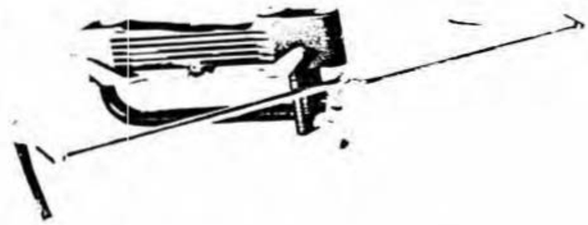

FIGURE 9-5.—Gemini X Hand Held Maneuvering Unit configuration.

On the Gemini X flight, the propellant was stored in two 439-cubic-inch tanks in the spacecraft adapter section and was fed to the Hand Held Maneuvering Unit through a 50-foot dual umbilical (fig. 9-6). One hose in the umbilical provided life-support oxygen and the other hose provided nitrogen gas to the Hand Held Maneuvering Unit. Nitrogen was selected as a propellant to reduce slightly some of the low-temperature problems encountered with Freon 14. The two nitrogen tanks provided a total impulse of 677 lb ×

FIGURE 9-4.—Shutoff valve upstream of coupling of Gemini VIII Hand Held Maneuvering Unit. Arms in near folded position.

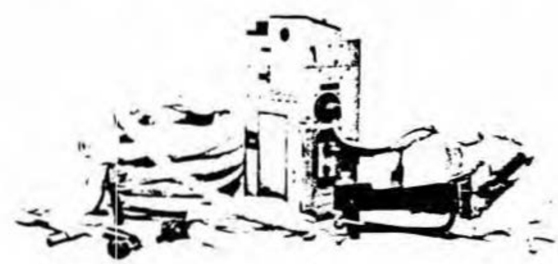

FIGURE 9-6.—Fifty-foot dual umbilical used in Gemini X shown connected to Extravehicular Life-Support System and Hand Held Maneuvering Unit.

sec, amounting to 84 ft/sec change in velocity of the extravehicular pilot. A list of other pertinent characteristics is provided in table 9–III.

TABLE 9–III.—*Gemini X and XI Hand Held Maneuvering Unit Characteristics*

Propellant	Nitrogen gas
Thrust, lb	0 to 2
Specific impulse, sec	63
Total impulse, lb×sec	677
Total available ΔV, ft/sec	84
Trigger preload, lb	5
Trigger force at maximum thrust, lb	8
Storage-tank pressure, psi	5000
Regulated pressure, psi	125±15
Nozzle-area ratio	50:1
Weight of usable propellant, lb	10.75
Weight of Hand Held Maneuvering Unit, lb	3
Weight of extravehicular pilot, lb	260

A hardline was routed from the tank installation in the spacecraft adapter section to a recessed panel behind the hatch. The hardline was clamped to the adapter-section structure at numerous points to provide heat shorts for warming the cooled gas (due to adiabatic expansion during use).

After connecting the life-support side of the dual umbilical to the oxygen system in the pressurized spacecraft and making the proper connections to the Extravehicular Life-Support System chest pack, the pilot egressed the cabin and moved to a recessed panel behind the hatch. The pilot connected the Hand Held Maneuvering Unit propellant side of the dual umbilical to the nitrogen supply by means of a push-on connector and a shutoff valve provided on the recessed panel.

Gemini XI Hand Held Maneuvering Unit

In the Gemini XI mission, the Hand Held Maneuvering Unit was stowed in the spacecraft adapter section rather than in the cabin. The screw-on coupling was changed to a quick-disconnect coupling (fig. 9–7) to simplify connecting the Hand Held Maneuvering Unit to the umbilical. The extravehicular pilot had to perform this operation with one hand in a limited access area and in a pressurized suit. Several features were incorporated in the push-on coupling to provide immediate interchanging of the Hand Held Maneuvering Unit with a gas-powered tool for possible future maintenance and assembly operations in space.

FIGURE 9–7.—Gemini XI Hand Held Maneuvering Unit in inverted position showing quick-disconnect coupling.

The propellant gas storage-tank installation for Gemini XI was identical to the Gemini X configuration and provided the same operational characteristics (table 9–III). A 30-foot dual umbilical was employed rather than the 50-foot dual umbilical used on Gemini X.

Astronaut Maneuvering Unit

The Air Force Astronaut Maneuvering Unit (fig. 9-8) was scheduled for evaluation on the Gemini IX–A and the Gemini XII missions. Pertinent characteristics of the Astronaut Maneuvering Unit are listed in table 9–IV.

The Astronaut Maneuvering Unit backpack contained hydrogen peroxide, nitrogen, and oxygen tanks; two sets of rate gyros; twelve 2.3-pound thrust chambers with associated solenoid-operated valves; self-contained radio and telemetry equipment; and

FIGURE 9-8.—The Air Force Astronaut Maneuvering Unit as configured for Gemini IX-A. Extravehicular Life-Support System (chest pack) also shown.

other miscellaneous equipment. The backpack was designed to provide attitude control and stabilization about the yaw, pitch, and roll axes, as well as translation in the fore-and-aft and up-and-down directions. Attitude control could be achieved either by using the thrusters in a direct manual on-off mode or in a rate-command mode.

The Astronaut Maneuvering Unit was capable of providing a change in velocity of about 250 feet per second for an all-inclusive extravehicular pilot weight of 407 pounds. The gross weight of the Astronaut Maneuvering Unit, 168 pounds, included a 19-pound oxygen bottle which held 7 pounds of gaseous oxygen for the Extravehicular Life-Support System. The nitrogen in the Astronaut Maneuvering Unit was used to expel the hydrogen peroxide through the catalyst beds and then through the reaction nozzles.

TABLE 9-IV.—*Gemini IX-A Astronaut Maneuvering Unit Characteristics*

Propellant	90 percent hydrogen peroxide
Total thrust (fore-and-aft or up-and-down), lb	4.6
Pitch moment, in.-lb	63.5
Roll moment, in.-lb	44.2
Yaw moment, in.-lb	47.7
Specific impulse, sec	169
Total impulse, lb×sec	3100
Total available ΔV, ft/sec	250
Controller characteristics:	
Breakout:	
Fore-and-aft, lb	4.5
Up-and-down, lb	4.5
Pitch, lb	4.0
Roll, lb	4.0
Yaw	Small
Maximum force:	
Fore-and-aft, lb	9.75
Up-and-down, lb	9.75
Pitch, lb	10.5
Roll, lb	10.5
Yaw, in.-lb	13.0
Maximum deflection, deg:	
Fore-and-aft	6
Up-and-down	6
Pitch	6
Roll	6
Yaw	4.5
Attitude-limit cycle periods, sec:	
Pitch	59
Roll	50
Yaw	3.2
Attitude deadband, deg	(3 axes) ±2.4
Maximum control rates, deg/sec:	
Pitch	18
Roll	27
Yaw	18
Maximum nitrogen tank pressure, psi	3500
Regulated hydrogen peroxide pressure, psi	455
Nozzle-area ratio	40:1
Weight of propellant, lb	24
Weight of Astronaut Maneuvering Unit, lb	168
Weight of extravehicular pilot, lb	407

Ground Training for Extravehicular Maneuvering

Hand Held Maneuvering Unit Control Logic

A number of different procedures could be used successfully to move from one point to another in space with a Hand Held Maneuvering Unit. Figure 9–9 illustrates the particular procedures selected for use with the Gemini systems. The figure illustrates tractor thrusting for either forward or backward translation, as well as pusher thrusting, and applies to any of the three possible rotational control axes: yaw, pitch, or roll. For example, in figure 9–9(a) assume that the illustration refers to the yaw axis so that our view of the man is from directly above; that is, the label "MAN" refers to the end of a line running from the operator's head to foot. The Hand Held Maneuvering Unit is held in front of the man's center of gravity at the position of the label "FORCE." The force in this case is pointed forward as it must be when considering the tractor mode. Assume that a disturbance occurs and causes a rotation to the right, indicated by the curved velocity arrow labeled "+ω." To eliminate this disturbance, the Hand Held Maneuvering Unit must be moved laterally toward the right side; however, the thrust line of the Hand Held Maneuvering Unit must be pointed directly at the target. By pointing directly at the target at all times, the operator (1) insures that he will eventually arrive exactly at the target, (2) maximizes the desired control moment, and (3) minimizes the amount of fuel required for attitude control. The third rule on the illustration refers to phase lead and states that the control motions should lead the disturbances if the rotational motions are to be completely damped. If, instead of leading the rotational motions, the control motions remain exactly in phase with the rotational motions, the result is a con-

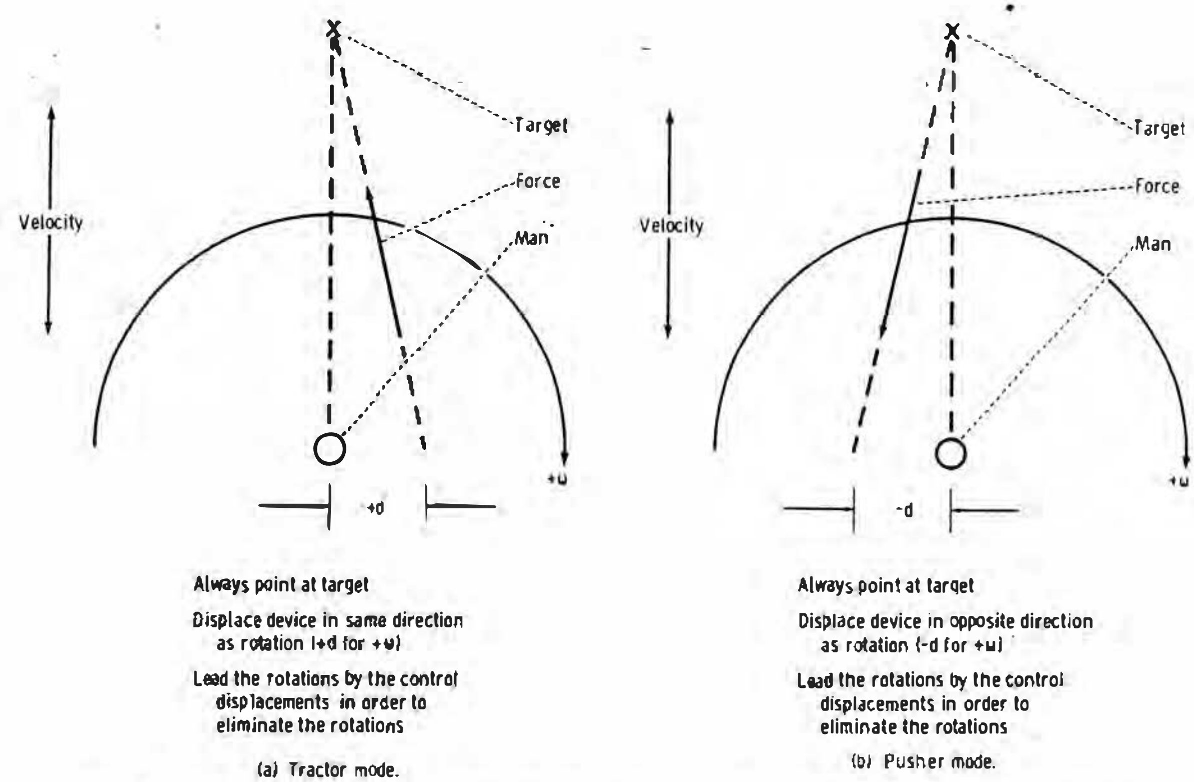

FIGURE 9–9.—Rules for attitude control during translation with Hand Held Maneuvering Unit.