Gemini Rendezvous Program

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One of the main purposes of the Gemini program is to establish techniques for the rendezvous and docking of space vehicles. Since it is expected that these techniques will be used in many other space programs, the Gemini rendezvous experiments have been planned to provide the basic information needed, and provision has been made for accomplishing rendezvous in several ways, each having special features to be explored in the program so that an optimized operational procedure can be established.

Introduction

GEMINI, the manned space vehicle system succeeding Mercury, has been planned to provide the experience required for future manned space programs. One of the main goals of the Gemini program is to establish techniques for rendezvous and docking with another space vehicle—of great interest because rendezvous is considered essential in many future manned space flight programs. For instance, rendezvous has been selected as a primary phase in the Apollo lunar landing mission. Types of programs requiring rendezvous are lunar orbit rendezvous (Apollo), earth orbital rendezvous (deep space), space stations, and satellite inspection and repair missions.

In order to develop general rendezvous techniques applicable to future programs, similarities among the possible rendezvous plans must be determined. Several characteristics are shared, assuming the basic premise that both operations and fuel should be minimized. For example: 1) the target vehicle would be established in orbit prior to spacecraft launch; 2) a launch window of sufficient length would provide the necessary confidence level for operational use; 3) launch techniques for the manned spacecraft would limit the out-of-plane error to within the terminal maneuvering capabilities of the chaser vehicle; 4) one vehicle would maintain a fixed orbit unless an emergency condition arose; 5) the initial orbit of the two vehicles would be of different periods for overcoming phase differences; and 6) terminal guidance techniques would be capable of overcoming dispersions generated through the launch and midcourse phases of the mission.

General Method

From the forementioned basic characteristics, a generally applicable operational rendezvous technique has been established. Shown in Fig. 1 is a flow diagram of the possible steps in an operational mission from the launch phase to the docking phase.

The target vehicle would be in orbit prior to the initiation of the manned spacecraft launch sequence. The spacecraft would be launched by using a variable-azimuth launch technique (to be presented in more detail in a subsequent section). This technique controls the out-of-plane displacement for an acceptable period of time within the maneuvering capabilities of the onboard propulsion system.

If launch of the spacecraft occurs at a time in the launch window when the phase relationship between target vehicle and launch site is optimum, then the mission sequence will go directly from the launch phase into the terminal phase, this transition being called an immediate rendezvous. Should the phase relationship be less than optimum, then the period difference created by the altitude difference of the target and spacecraft orbits would provide a fixed initial catch-up rate for correcting the phase error. If a different catch-up rate is required to reduce trailing displacement dispersions in the terminal phase of the mission, the chaser vehicle, which in the majority of cases will be the spacecraft, would adjust its orbital period to provide the desired catch-up rate. The ground computer complex can determine the adjustment and provide the chaser vehicle with the appropriate information through a command link or by voice communications. It is also possible to determine this adjustment at launch, and with suitable programs the adjustment can be implemented by the spacecraft's on-board computer.

There are two primary methods of conducting a terminal maneuver. In one method, an optimum intercept course of the chaser vehicle is established by the use of a radar with angular measuring capability and an on-board computer. This minimizes the fuel required and thus allows relatively large initiation ranges between the target and chaser vehicle.

A second method uses optical guidance. This requires considerably more fuel—hence the initiation ranges would be more restricted, but the hardware requirements would be minimized; and ranging information would have to be provided to the astronauts either by a radar with range and range-rate capability only or by cruder optical measuring devices[1] such as a reference grid or simple sextant device.

As the requirements of rendezvous programs vary, so will parameters, such as time, altitude, inclination, and velocity, which form the restraints and boundaries of the steps shown in Fig. 1. In order to determine these values for a given program, five major parameters must be evaluated and interrelated until they are compatible: 1) operational launch window;

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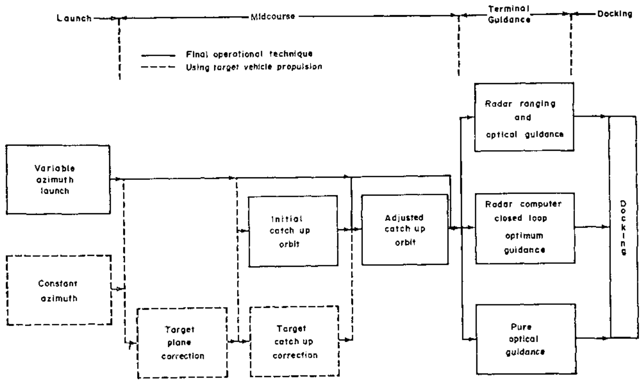

Fig. 1 Rendezvous technique flow diagram.

2) fuel allotted for rendezvous; 3) target inclination; 4) orbit altitudes; and 5) mission time.

The operational launch window is the actual time period required to provide an acceptable confidence level for operational use of the launch vehicle and spacecraft under consideration.

Control of the out-of-plane error requires the most rendezvous propulsion. Using controlled or variable-azimuth launch techniques in conjunction with the target inclination relative to the launch site will control the out-of-plane errors between certain limits for a period of time. For a given amount of fuel (or velocity) to be used for maximum out-of-plane conditions in the terminal phase, a plane launch window is established. This launch time period with respect to out-of-plane errors must be as large as the actual operational launch period required. If the plane window is smaller, either the target inclination must be lowered with respect to the launch site, the fuel allotment must be increased, or countdown and launch procedures must be improved so that a smaller operational launch window is required.

The difference in the orbit altitudes of the two vehicles will provide a fixed catch-up rate between the target and spacecraft for correcting phase differences. The mission time sets the amount of time in which the midcourse catch-up maneuver must be completed. Catch-up time and rate regulate the degrees of phase difference that can be allowed at launch. This phase difference limit provides a "phase launch window." Again, as with the out-of-plane consideration, this phase launch period must be as large as the operational launch window. If the phase window is smaller, either the catch-up rate must be increased, the mission time extended, or the required launch window reduced.

Depending on the relative sizes of the phase and plane windows, either one or both of which may be much larger than the operational window, the optimum time of launch might be with respect to the plane or to the phase. In the Gemini program, as will be shown later, the optimum launch time is with respect to phase.

If both the plane and phase windows are equal to, or very near, the size of the operational launch window, additional evaluation must be made, since the phase and plane windows do not necessarily begin near the same local time on the acceptable launch days. A study of the phasing times with respect to the plane position on the possible launch days would determine whether adjustments to the plane or phase launch windows are required.

The point to re-emphasize is that although the values for the five major parameters just discussed will vary according to the program, the operational technique that uses these parameters can be basically the same.

Since the objective of the Gemini rendezvous program is to develop and demonstrate techniques that will be practical for the majority of possible space programs which utilize rendezvous, a more comprehensive program is required than one in which a specific type of rendezvous is accomplished a few times. The program must include different techniques and a wide range of variables, so as to arrive at the optimum and most practical procedures. Since the performance and accuracy of systems used in rendezvous are initially based on estimates and even the feasibility of some features is not established, a very conservative assessment of the design performance of these systems was adopted, at least for the early stages of the program. For this reason, hardware has been designed so that the alternate methods of rendezvous could be provided both in the spacecraft and the target, so that at least some of the objectives of any given mission can be accomplished, if one method cannot be carried out. Through experience, techniques will be refined and developed to establish optimum practical operating methods. In order to accomplish a flight program involving these concepts, all of the rendezvous vehicles are planned to have the same configuration and hence the same capability. The experiments will be designed to exploit this capability progressively, but if, for instance, the launch window actually achieved and the insertion accuracy are substantially better than anticipated, this fact can be exploited, even in real time, in optimizing the subsequent maneuvers.

Actual Rendezvous Parameters

In describing the actual Gemini rendezvous missions, the important parameters for the operational mission are presented first. Then, use of the propulsive capability of the target vehicle to provide increased launch-window tolerance, if required in the preliminary experiments, is described.

The orbits selected for the target vehicle and spacecraft are presented in Fig. 2. The target vehicle will be placed in a 160-naut-mile circular orbit, and the spacecraft's orbit will be an elliptical orbit having a perigee of 87 naut miles and an apogee equal to the orbital altitude of the target. These orbits were selected primarily from launch vehicle performance and orbital lifetime limitations. The period difference in these two orbits provides the Gemini spacecraft, which is the chaser vehicle, with a 5.5° per revolution catch-up rate for correcting phase differences.

For another performance reason, the spacecraft systems will have operational capability of 2 days; therefore, rendezvous will be completed within 2 days after spacecraft launch. Present plans indicate that 29 spacecraft inertial orbits measured from the first apogee can be used for completing the catch-up maneuver. With a catch-up rate of 5.5°, an out-of-

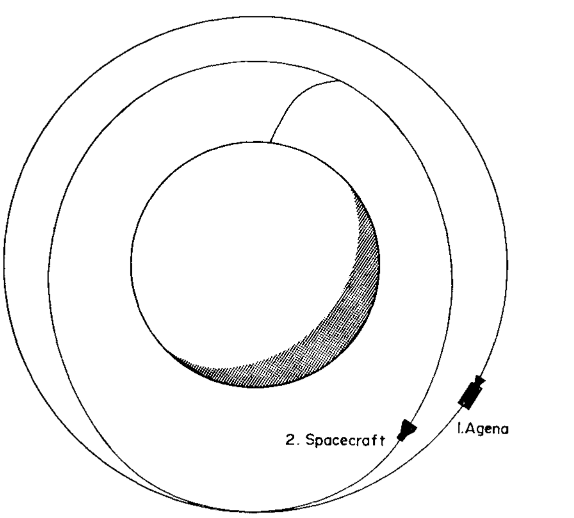

Fig. 2 Orbits selected for Gemini rendezvous program: 1) Agena targetvehicle placed in a 160-naut-mile circular orbit, and 2) Gemini spacecraft placed in an elliptical orbit perigee 87 naut miles—apogee 160 naut miles.

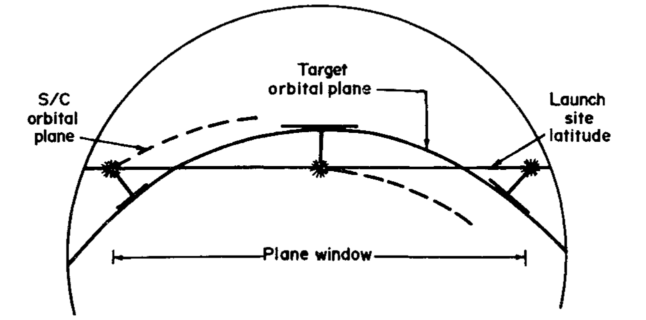

Fig. 3 Variable azimuth launch. Azimuth will vary with time of launch, launch is parallel with Agena plane, and intersection occurs 90° from launch point.

phase difference of approximately 160° will be a maximum. This phase difference corresponds to approximately 43 min of phase launch window. On several Gemini missions, post-rendezvous maneuvers with the spacecraft and target in the docked configuration are planned, so it will be desirable to limit the rendezvous time to 1 day in order to conduct these maneuvers during the second day. This time limit would reduce the launch-phase window to approximately 20 min.

The target vehicle will be placed into orbit prior to the time of spacecraft launch. The systems of the target will be capable of operating for 5 days; thus, the spacecraft would be provided 3, 4, or 5 possible launch days. A 1- or 2-day mission completion time, the start of the spacecraft launch window at the beginning of the target's second orbit or second day, or postrendezvous requirements of the target will cause this variation in some possible launch days.

For the Gemini missions, the particular target orbital inclination of 28.87° has been selected in order to optimize the plane launch window and rendezvous fuel requirements.

Using this inclination and variable-azimuth launch techniques, the out-of-plane errors can be restricted to relatively small values for an extended period. One variable-azimuth launch technique is shown in Fig. 3. The maximum out-of-plane error that must be accommodated by the Gemini spacecraft in the terminal phase will be the perpendicular distance between the launch site and the plane of the target vehicle. This accommodation to error is accomplished by launching the spacecraft on an azimuth parallel to the plane of the target vehicle at the point of launch. Intersection of the two planes will occur approximately 90° from the launch site. In order to provide a continuous plane window, as shown in Fig. 4, the target's maximum latitude point, which is established by the inclination in relation to the launch site latitude, must not exceed the spacecraft's terminal maneuvering capabilities. If the maximum latitude point of the target's plane relative to the latitude of the launch site exceeds an established spacecraft out-of-plane capability, the plane launch window will be broken into two sections. One section would be located around the ascending arc of the target plane and the second around the descending arc. An example demonstrating this point is given in Fig. 5. By using 0.53° for the upper limit of out-of-plane error, the plane launch window is reduced from one 135-min to two 33-min periods when the target inclination with respect to the launch site at Cape Canaveral is changed by only 1° from the optimum. A further increase of 1° reduces the launch window to two 26-min periods. This example shows the extreme sensitivity of launch window to the inclination of the target plane.

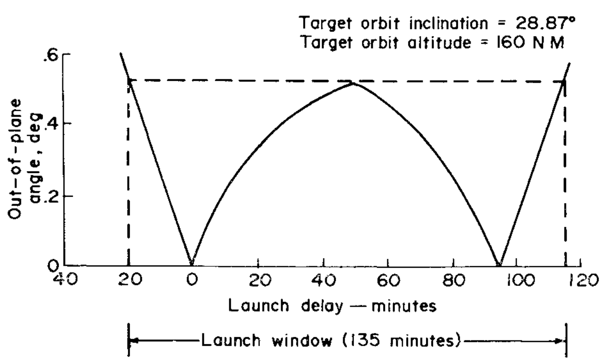

Fig. 4 Gemini plane window.

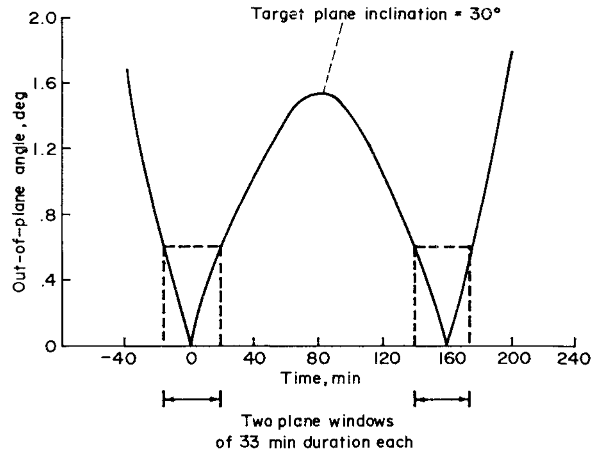

Fig. 5 Plane launch window with target orbital plane inclination of 30°.

This parallel-azimuth launch technique is relatively easy to implement during guidance. No additional performance is required from the launch vehicle, and its payload capability will have negligible variations due to the variation of the earth component as the azimuth changes with launch time.

A second variable-azimuth launch technique makes use of launch-vehicle guidance in yaw during the latter phases of powered flight to minimize or steer out plane errors. Guidance is accomplished by varying the launch azimuth of the spacecraft so that it is an optimum angle directed toward the target's plane, thus reducing the out-of-plane distance prior to initiation of yaw guidance. This technique requires some additional performance from the launch vehicle.

For the Gemini program, appreciable performance margins may not be available; therefore, the Gemini parameters are being selected using a parallel-launch variable-azimuth technique. However, provision is being made for possible use of the variable-azimuth yaw steering technique. The variable-azimuth launch techniques will provide biases to offset relative nodal regression effects of the two orbits, so that a minimum out-of-plane error is provided at the start of the terminal phase rather than at insertion of the spacecraft in its initial plane.

An out-of-plane correction capability of 0.53° during the terminal maneuver will be provided in the Gemini spacecraft, and the plane window for which the parallel launch technique will be used is shown on Fig. 4. This window, the total length of which is 135 min, is much larger than the Gemini phase window of 43 min, based on a 2-day mission time; therefore, the optimum launch time or time "zero" will be with respect to phase. Figure 6 shows the launch times for each possible launch day measured from the time of target lift-off. The beginning of each window represents the first permissible phase condition within the plane window. In most cases this phase position will be the optimum or zero condition. These numbers are based on a mission time of 2 days.

It is anticipated that these launch times will, at least ultimately, be adequate for actual launch requirements. Since, as stated previously, a 1-day rendezvous mission is desired, these launch windows would be reduced to panes of approximately 20 min.

Based on the expected maximum out-of-plane errors in the terminal phase of the mission and the energy level between

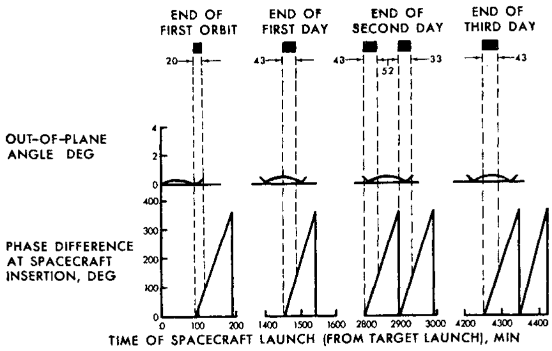

Fig. 6 Gemini launch window; inclination = 28.87°, altitude = 160 naut miles.

the two initial orbits, sufficient fuel for a velocity capability of 650 fps is provided in the spacecraft for conducting the rendezvous mission.

In the terminal phase of the mission, Gemini will develop maneuver techniques in which a radar-computer optimum guidance technique and an optical guidance method are used. An interferometer type of radar with both angular and range measuring capability, together with the on-board computer, will provide the hardware system necessary to compute an optimum solution. The on-board computer will be programmed with a set of equations that describe the motion of the spacecraft with respect to a rotating coordinate frame centered in the target vehicle. These equations will be "modified Clohessy-Wiltshire"[2] linearized equations of motion.

The spacecraft will acquire the Agena target vehicle with its radar at a range of approximately 250 naut miles. Upon acquisition, the radar will provide range, range-rate, and angular-displacement information. Range and range-rate information will be displayed to the astronauts while range and angular-displacement data are introduced into the computer. With these raw data, the computer will calculate the relative velocity components along the three axes. From these velocity components, the relative motion equations will be used to determine the velocity increment and direction required in real time to establish the spacecraft on the optimum intercept course. This information will be displayed to the astronauts, who will monitor the changing velocity and position requirements. After the velocity requirement tends to level off near a minimum value, the astronaut will select the time to orient the spacecraft to the proper attitude and apply the proper velocity impulse. When the range is reduced to approximately 2 miles, the astronaut will apply a braking impulse to reduce the closing rates for final docking operations. Between the initial and final impulses, small intermediate corrections will be computed and applied. The number of intermediate corrections will depend on the maneuver completion time after the initial impulse is applied. Depending on the accuracy of the midcourse catch-up maneuver, the time to complete the maneuver may correspond to as little as 30° of orbit travel but is normally as much as 270° of travel. Figure 7 shows an example of velocity requirements at different spacecraft positions from perigee using the radar-computer terminal maneuvering technique. These velocity values were derived using a specific set of initial conditions. The orbital travel to complete the maneuver after first impulse is 270° in this case. The plot of velocity represents the velocity that would be required to conduct the maneuver if it were initiated at any point over the final two orbits for the given set of starting conditions. The velocity requirement includes the initial and braking impulses. It might be noted that the catch-up rate over the final two orbits for the conditions shown in Fig. 7 was 1.5° per revolution. For faster rates (up to 5.5° per revolution) and/or larger out-of-plane errors (up to 0.53°), the required velocity would be greater, but the trend of the curve would be similar. Within the out-of-plane tolerances established by the launch technique, it should be possible to initiate the terminal phase, if required, at ranges as great as 250 naut miles.

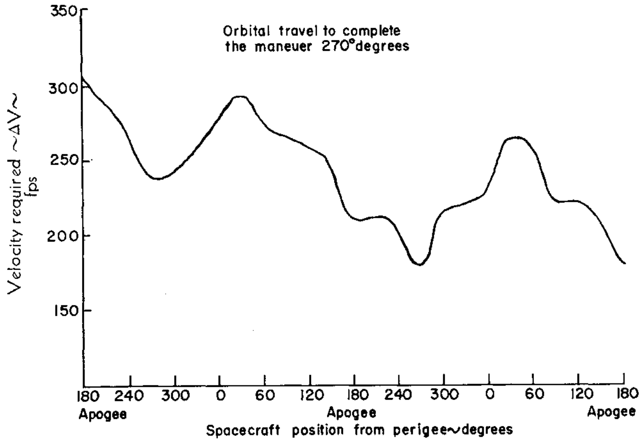

Fig. 7 Radar computer closed loop terminal maneuver velocity requirements.

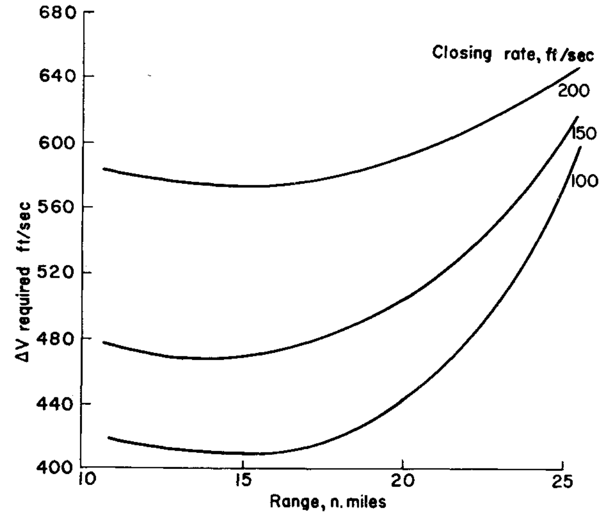

Fig. 8 Optical guidance terminal maneuver velocity requirements.

The optical guidance technique, which will be used for the Gemini missions, will be relatively simple from an operational aspect. The range and range-rate information from the Gemini radar will be displayed to the astronaut for establishing the initiation time and braking schedule. The astronauts will observe the relative motion between the spacecraft and the target vehicle with respect to a star background. The Agena vehicle will be equipped with a flashing light so that it can be detected against the star field.[3] When within proper range of the target vehicle, an astronaut, by using his on-board propulsion system, will thrust normal to the angular motion of the target vehicle. The astronaut will continue to thrust until he observes that the relative motion has been eliminated. At this instant, the spacecraft is on a constant line-of-sight approach to the target vehicle, this approach to be re-established periodically during the maneuver when the relative motion is again noticeable. By monitoring the range and range rate, the astronaut establishes the proper braking schedule similar to that required for the other technique. The maneuver time will be to the order of 20 min and the initiation range will be approximately 20 naut miles. Since the initiation range for this technique is much smaller than that required for the other technique, necessary midcourse adjustments will be oriented to provide minimum miss distances of less than 20 miles. This latter procedure makes it possible to use either technique; thus, redundancy is added to the terminal phase. Figure 8 shows the velocity requirements for conducting the optical guidance maneuver for a specific out-of-plane condition with different ranges and closing rates. These results were obtained from a simulation study and represent the most severe initial starting conditions with respect to the out-of-plane velocity component.

Investigation has shown that the velocity variation with initial range is primarily a result of the variation in the magnitude and direction of the velocity vector as the spacecraft approaches the target on the catchup trajectory. In general, a catch-up trajectory can be established such that as range decreases, the direction of the velocity vector approaches the direction of the line-of-sight between the target and the spacecraft, thus reducing the corrective velocity required to establish a flight path along the line-of-sight.

The optical system that uses an optical method for providing ranging information to the astronauts, to serve as backup in the event of radar failure, will be developed and exercised during the Gemini program.

Docking Phase

When the two vehicles are within a quarter of a mile of each other and the relative velocity has been reduced to approximately 8 to 10 fps, the docking maneuver will begin. Windows in the spacecraft provide adequate visibility so that the astronauts can perform the docking operation with manual control. A docking collar on the Agena vehicle which engages with the small end of the spacecraft is being designed to absorb shock loads up to $1\frac{1}{2}$ fps. The mechanical design of the shock absorbers and latches has been investigated[4] using a quarter scale model, and the results are satisfactory for the particular scheme. Also, some of the results obtained at the Langley Research Center using a fixed base simulator with a TV visual display are given in Figs. 9 and 10. These show that successful and consistent docking can be achieved within the design limits.

In order to prove the design concept and provide the astronauts with docking experience, two mechanical docking simulators will be used. One simulator has been built at the Langley Research Center and the other is being built at McDonnell Aircraft Corporation. The Langley simulator serves as a research simulator and will provide general information in rendezvous docking, and the McDonnell simulator[5] will utilize Gemini hardware and will be used for hardware qualification and training.

The "Active" Agena

As previously stated, Gemini will be the initial "workhorse" in developing general rendezvous techniques in each phase of the mission. In order to determine the actual accuracies and dispersions in developing rendezvous technology in all phases of a mission, it is necessary that the missions be completed. For the initial Gemini rendezvous missions, additional capability is considered necessary until adequate real-time knowledge of the accuracies and techniques is established.

In order to provide this additional capability, the Agena vehicle has been selected as the target vehicle for the Gemini program. The Agena is a space-stabilized vehicle which can be maneuvered in orbit. The Agena not only can aid the spacecraft in conducting the rendezvous mission, but it also provides the propulsion for performing postdocking maneuvers, which is another objective of Gemini. It provides a flexible rendezvous technique. Figure 1 shows this flexible technique. The dotted lines show the additions to the operational part of the diagram which was discussed previously.

The Agena vehicle, after insertion into its 160-naut-mile circular orbit, will have a velocity capability of approximately 5400 fps which can be used for midcourse corrections. This additional plane- and phase-correction capability provides additional launch-window capability.

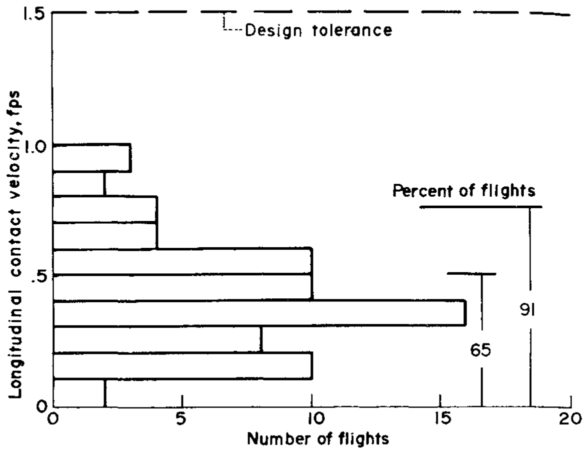

Fig. 9 Longitudinal velocity at contact.

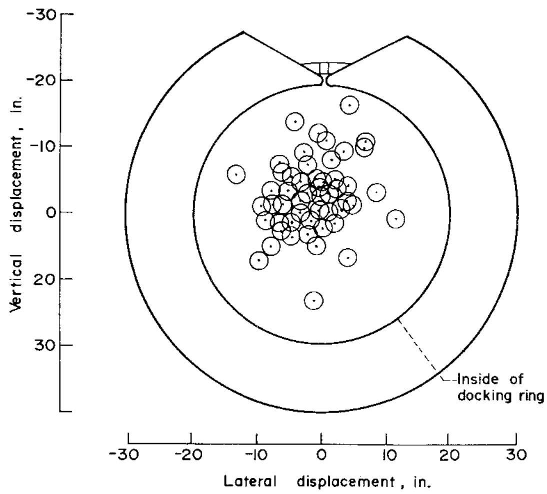

Fig. 10 Vertical and lateral displacements at contact.

For a 1-day mission completion time, the Agena can provide a catch-up rate that will allow a full 360° phase difference if required. This capability to correct all phase errors greater than 70° for a 1-day mission removes all restrictions with respect to a phase window. To perform this maneuver, the Agena would receive from a ground command site the required information for storage in its programmer. At the proper point in its orbit, the main Agena engine will be ignited and an elliptical orbit whose apogee might extend out as far as 550 naut miles will be achieved. The new Agena ellipse will increase the catch-up rate so that catch-up will occur on a selected orbit. The size of the ellipse will vary with the magnitude of the phase difference. The Agena will recircularize its orbit by applying a retro-impulse at its 160-naut-mile perigee when the phase difference has been reduced to between the values of 5° and 11°. This impulse provides two spacecraft catch-up orbits for completing the midcourse maneuver. Final midcourse adjustments and closing conditions will be identical to those required if the Agena had not maneuvered. The maximum Agena ellipse and the recircularization maneuver will require approximately 1300 fps from the Agena vehicle with two main engine ignitions.

Depending on the phase position relative to the plane position at spacecraft launch, the amount of Agena velocity available for correcting excessive plane errors will vary between approximately 5400 to 4100 fps. Since the relationship between phase and plane can be calculated, accurate plane windows have been determined. A velocity of 5400 fps corresponds to a 12° out-of-plane correction, and 4100 fps corresponds to a 9.4° correction.

When the spacecraft is launched and the angle formed between the Agena and spacecraft planes exceeds 0.53°, the

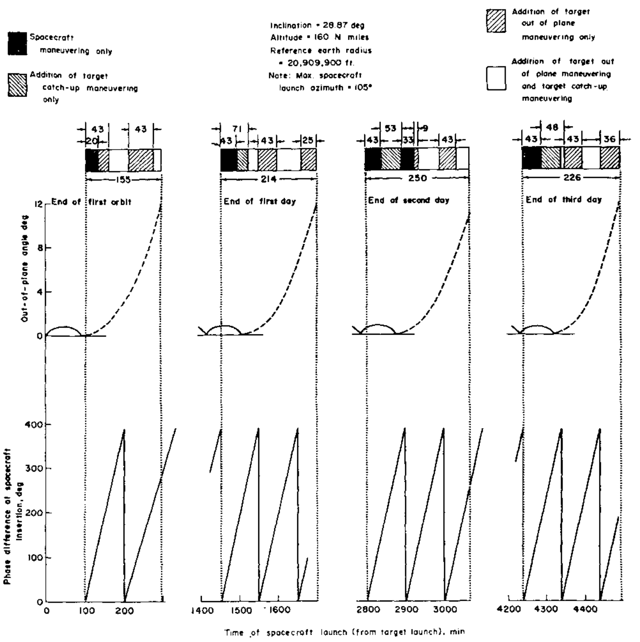

Fig. 11 Gemini launch window using "active" Agena.

Agena will make a plane maneuver. From tracking data, the ground computer complex will determine the two positions where the planes intersect, the required velocity and direction, and the time of engine ignition. The first convenient command site will transmit this information to the Agena where it will be stored in the Agena programmer until time to begin the maneuver. There will be portions of the launch window in which the Agena will be required to conduct both a plane and a phase maneuver. These maneuvers could be initiated simultaneously; however, from accuracy considerations, the present plan is to conduct them separately.

The Agena plane-correction capability will increase the plane window appreciably, but not as much as might be expected, for two reasons. The first is that the optimum or "zero" time of launch will still be at the first optimum phasing condition after the plane position is within 0.53° of the launch site. This allows the spacecraft to conduct the mission without Agena maneuvers. The window, therefore, will only be opened in one direction. The second reason is that the parallel-launch variable-azimuth technique cannot be employed for out-of-plane conditions as large as 9.4° because the spacecraft launch azimuth is becoming more southerly with time, and a range safety limit, as well as a landing-area restraint, will be imposed. A launch azimuth of 105° has been established as the southerly bound for the variable-azimuth launch technique. When the azimuth reaches 105°, the out-of-plane error using the parallel-launch procedure will be approximately 3.2°. A constant launch azimuth at 105° will be used until the plane angle, which will be formed between the two planes, reaches the limit of the Agena capability. This case is on the flow diagram shown in Fig. 1. The launch window, which includes the Agena capability, is shown in Fig. 11. The window is divided to show the time when spacecraft maneuvering only is required, when only Agena phase maneuvering is required, when only Agena plane maneuvering is required, and when both Agena plane and phase maneuvers are necessary.

As the program progresses and experience is gained, a better knowledge of the accuracies and dispersions associated with the techniques in the different phases of the mission will be acquired. It is then anticipated that the increased capability provided by the Agena will not be required and the rendezvous technique will become the operational technique shown in Fig. 1.

Rendezvous Capability

At the completion of an operational rendezvous exercise, the Agena, when docked with the Gemini spacecraft, will have a velocity capability of approximately 2340 fps for conducting postrendezvous maneuvers. After docking, the astronauts will be able to control the Agena's attitude, along with monitoring its systems, and igniting its engine. The on-board inertial navigation system[6] has the capability of performing the necessary guidance for these maneuvers and of making it possible to select new landing sites based on the updated orbital parameters. When this phase of the program is completed, rendezvous in space will have reached maturity.

Appendix

Definitions

Launch window is the time span over which a launch may be performed and still meet the timing conditions for rendezvous.

Operational launch window is the launch window as defined and limited by the practical hardware constraints of the launch system.

Phase (launch) window is the launch window as determined by the ability of the rendezvousing vehicles to catch up with each other when they are separated by a phase difference or geocentric angular displacement.

Phase (launch) window is the launch window as determined by the ability of the rendezvousing vehicles' maneuverability to compensate for angular differences between the orbital planes of the two vehicles.

Midcourse phase is the portion of a rendezvous flight from injection into orbit up to the point where contact is made between the two vehicles, so that command of the subsequent closing operations is on board the rendezvousing vehicles.

Terminal phase is that portion of a rendezvous flight during which the astronauts in the spacecraft determine and execute the proper maneuver to establish the precise intercept course between the target and spacecraft. This maneuver will bring the two vehicles close enough together in both position and velocity to permit final docking operations.

Docking is the final phase of a rendezvous flight in which the two vehicles are brought into bodily contact and latched solidly together.

Inclination of orbital plane is the angle between the equatorial plane and the plane of the satellite vehicle which passes through the center of the earth.

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