JOURNAL OF SPACECRAFT AND ROCKETS
Vol. 36, No. 2, March–April 1999

Technology of Automated Rendezvous and Capture in Space

Michael E. Polites*
NASA Marshall Space Flight Center, Huntsville, Alabama 35812

Results are presented of a study into the technology of automated rendezvous and capture (ARC) in space. The history of manual and automated rendezvous and capture and rendezvous and dock is presented. The need for ARC in space is defined. Today's technology and ongoing technology efforts related to ARC in space are reviewed. In light of these, ARC systems are proposed that meet the future needs for ARC, but that can be developed in a reasonable amount of time with a reasonable amount of money. Technology plans for developing these systems are presented; cost and schedule are included.

Introduction

THE United States has several space missions on the horizon that will require a capability in automated rendezvous and capture (ARC). However, it has not yet developed an ARC capability that will allow these missions to be accomplished, nor does it have a serious technology program for developing such an ARC capability. This is in stark contrast to other countries involved in space. The Russians were the first to develop an automated rendezvous and dock (ARD) capability. They used it extensively in resupplying their Mir space station and plan to use it for autonomously resupplying their part of the International Space Station (ISS). The Europeans and the Japanese do not have this capability as yet, but both have independent, ongoing technology programs for developing it. They, too, intend to use it for autonomously resupplying their part of the ISS, but they also have other broad, far-reaching uses for it.

Because of the obvious disparity between the ARC capability required by the United States for some future space missions and the limited ARC technology that it presently has available for accomplishing these missions, an assessment of ARC technology was made. This paper presents the results of that assessment. The objectives were to: research the history of both manual and automated rendezvous and capture and rendezvous and dock and the systems that have flown in space; identify the future United States needs for ARC in space; review today's technology and ongoing technology efforts related to ARC; in light of these, propose ARC systems that can be matured in a reasonable amount of time with a reasonable amount of money and still meet the needs and requirements of future U.S. space missions; develop a technology plan for maturing these systems; and offer any final comments and conclusions.

To effectively present the results of the ARC technology assessment, it is necessary to define some terms that will be used throughout this paper. They are as follows. The chase vehicle is a spacecraft that has both attitude and translational control capability. It actively navigates to the target vehicle in the rendezvous process. The target vehicle is a passive spacecraft in the rendezvous process. It normally has only attitude stabilization capability. Phasing is the initial segment of the rendezvous process that gets the chase vehicle to within about 40 km of the target vehicle. The next segment is proximity operations, when the chase vehicle navigates from about 40 km to within about 100 m of the target. The terminal phase is the final segment, when the chase vehicle closes from about 100 m to the point of dock, capture, or berth. Docking means mechanically connecting the chase vehicle to the target vehicle by propelling the chase vehicle into the target vehicle at a nonzero linear velocity. Capture means mechanically connecting the chase vehicle to the target vehicle using mechanical devices on the chase vehicle that grasp structure on the target vehicle. Capture is like a zero velocity dock. Berthing is mechanically connecting the chase and target vehicles together using a manipulator arm on one of the vehicles. The manipulator arm grasps the other vehicle and positions it into restraints on the vehicle with the manipulator arm.

History of Manual and Automated Rendezvous and Capture and Rendezvous and Dock

Virtually all of the world's space flight experience in manual and automated rendezvous and capture and rendezvous and dock comes from the U.S. and the Russian space programs, and all of this is of the rendezvous-and-dock type. Interestingly enough, the United States took a manual approach to rendezvous and dock and one that was also mission unique; i.e., the rendezvous-and-dock scenarios and timelines were all tailored to a specific mission. No attempt was made to standardize these. Consequently, the U.S. approach has been very labor intensive and expensive, requiring extensive crew training and system redundancy to ensure mission success. In contrast to this, the Russians pursued a course in rendezvous and dock that was primarily automated, with standardized operations. The flight crew was relegated to override and monitoring functions (Ref. 1, pp. 3 and 4).

The U.S. experience in rendezvous and dock dates back to the Gemini program of the 1960s, in which the in-flight rendezvous and dock tests and demonstrations served as a testbed for the Apollo lunar landing missions (Ref. 2, p. 3). Here the Gemini spacecraft was the chase vehicle and a modified Agena booster second stage was the target vehicle.[3] The modified Agena was inserted into a near circular orbit and stabilized along the local vertical. It was equipped with a special docking adapter that had a radar transponder to provide a strong return for radar signals transmitted from the Gemini's rendezvous radar. The docking adapter also had two high-intensity flashing lights that provided good optical targets for the Gemini crew. The docking adapter had a spring and shock-mounted cone that mated with the Gemini nose and absorbed the docking forces.[4] The Gemini spacecraft carried two crewmen, who interacted with the onboard guidance and control system, to accomplish rendezvous and dock. A block diagram of the Gemini guidance and control system is shown in Fig. 1 (Ref. 2, pp. 46 and 47). It is divided into pilot displays, the sensing and computing system, and the control system. In the sensing and computing system, the rendezvous radar, which is an interferometric-type system, estimates range and bearing to the target vehicle. This information is supplied to the computer at a range varying from 450 km to 150 m; it is displayed to the crew, along with range rate, from 90 km to 6 m (Ref. 5). Also displayed to the crew are the Gemini attitude, attitude rate, and linear velocity change required for a midcourse rendezvous correction. This displayed information allows the crew to rotate the spacecraft to the correct attitude and fire the maneuver thrusters to produce the required velocity change, using the attitude and maneuver control handles that are a part of the control system.

Received June 29, 1998; revision received Sept. 23, 1998; accepted for publication Sept. 23, 1998.

*Deputy Director, Astrionics Laboratory. Associate Fellow AIAA.

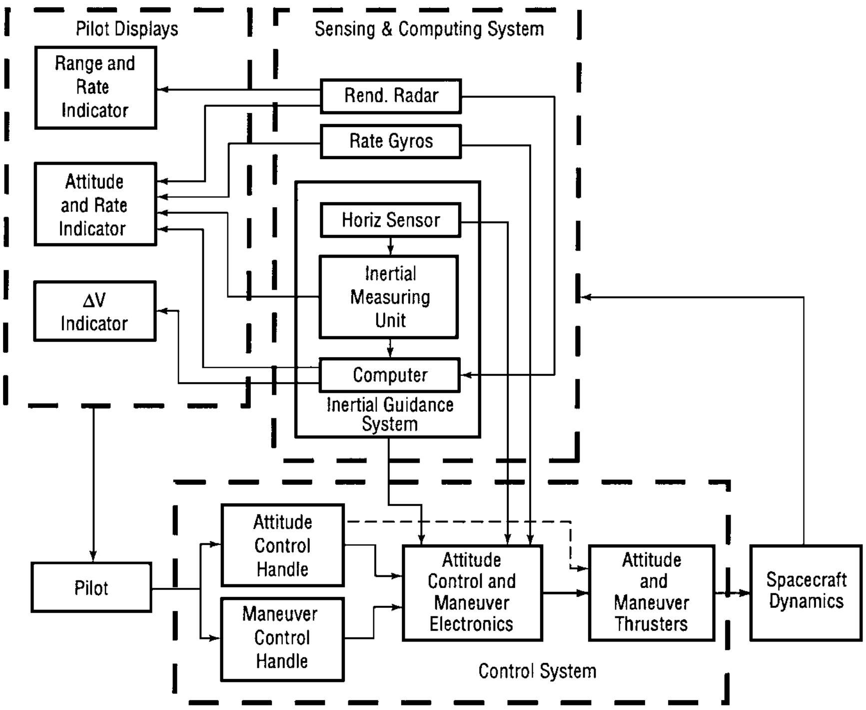


Fig. 1 Gemini guidance and control system where ΔV is change in velocity in meters per second.

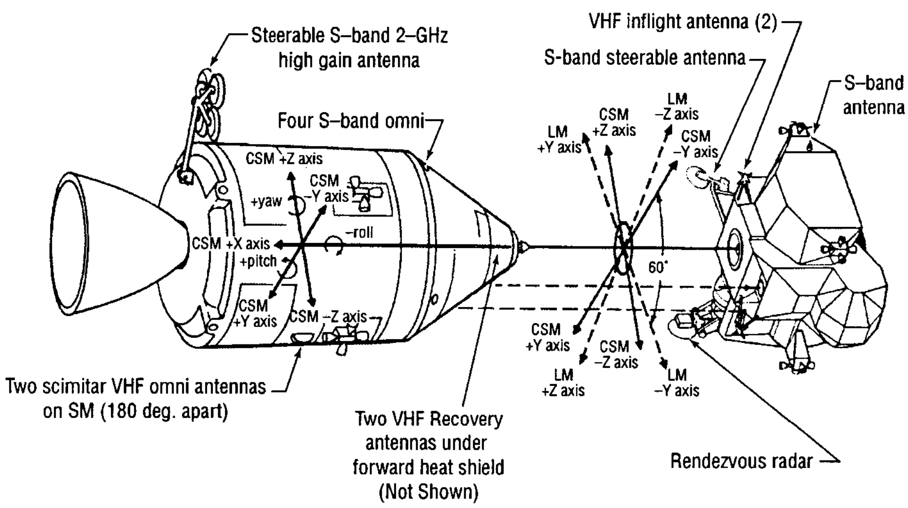


Fig. 2 Apollo command/service module and LEM ascent stage.

When the Gemini spacecraft is close enough to the target vehicle, the crew can complete the rendezvous and docking process using the control handles, observing the pilot displays, and observing the optical targets through windows in the spacecraft. At some point in the approach, typically, 60–15-m separation, the rendezvous radar can no longer give an accurate estimate of range because of the closeness of the target. Then, visual observations of the docking targets by the crew are heavily relied upon (Ref. 2, p. 31). Successful rendezvous and docks were accomplished by the flight crews on Gemini VIII in March 1996, Gemini X in July 1966, Gemini XI in September 1966, and Gemini XII in November 1996.

The Apollo program had a complete rendezvous and docking operation that was performed in lunar orbit. The approach was similar to that demonstrated in the Gemini program, which is no surprise. Here, the ascent stage of the lunar excursion module (LEM) was the chase vehicle. The command/service module (CSM) functioned as the target vehicle (see Figs. 2 and 3). The LEM ascent stage was launched from the lunar surface and then rendezvoused and docked with the CSM, which was parked in a circular lunar orbit. The LEM had two crewmen who participated in all phases of the process from monitoring the launch from the lunar surface to flying the LEM during docking.[6] They interacted with the LEM guidance and control system in the rendezvous and docking procedures much as in the Gemini program. The LEM guidance and control system was similar to Gemini's. It had a guidance digital computer, an inertial measurement unit (IMU), optical equipment for IMU alignment, and rendezvous radar. The rendezvous radar provided CSM range, range rate, and bearing to crew displays and to the guidance computer for maneuver computations. The operating range of the rendezvous radar was from 740 km to 24 m. The guidance and control system equipment, along with crew displays and controls, were all utilized in rendezvous and docking. A diagram of the docking mechanism is shown in Fig. 4.

The Space Shuttle Orbiter's approach to rendezvous and docking is much like its predecessors', which again is no great surprise. Here, the ground team computes the rendezvous burns to get the Orbiter within 74 km of the target. From this point on, most of the maneuvers are calculated and executed onboard, either automatically by the Orbiter's guidance, navigation, and control (GNC) system or manually by the flight crew interacting with the GNC system using hand controls and displays. The Orbiter's GNC system is similar to the LEM's so far as rendezvous and docking are

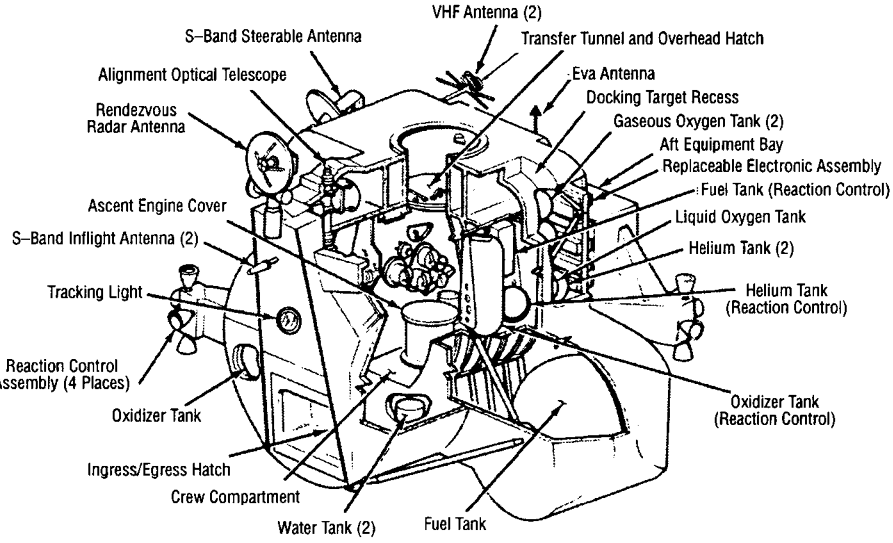


Fig. 3 Apollo lunar module ascent stage.

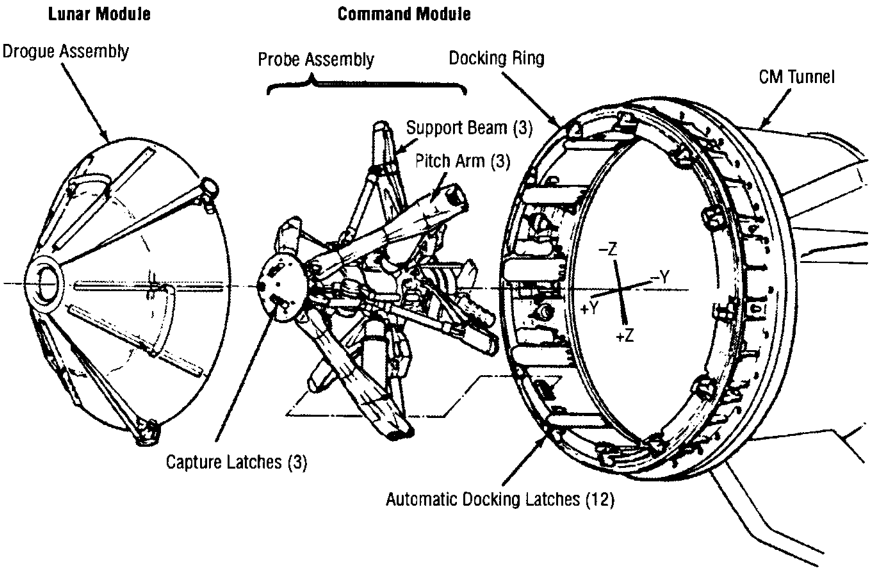


Fig. 4 Apollo docking mechanisms.

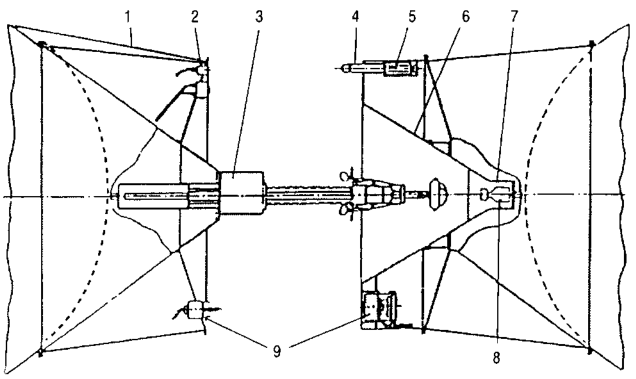
concerned. It has guidance digital computers, IMUs, optical equipment for IMU alignment, and rendezvous radar that are used in rendezvous and docking.[7] The rendezvous radar can operate in both active and passive modes. In the active mode, the target vehicle must have a transponder that generates a return signal for the radar signal transmitted by the Orbiter rendezvous radar. In this mode, the rendezvous radar has a range varying from 555 km to 30 m. In the passive mode, the return signal simply is the transmitted signal reflected off of the target vehicle. This is what is also known as skin tracking. In this case, the rendezvous radar has a range from 22 km to 30 m. In addition to this GNC equipment for rendezvous and docking, the Orbiter also has three additional items that are used in the rendezvous and docking process.[8] There is the trajectory control sensor (TCS), which is a laser ranging device that is mounted in the Orbiter's payload bay. It provides range, range rate, and bearing to the target for display to the crew at ranges varying from 1.5 km to 1.5 m. There is the centerline camera that is fixed to the center of the Orbiter's docking mechanism. The image from it is displayed to the crew as a visual aid for docking within about 90 m of the target. The crew also has a hand-held laser ranging device, which can be used during the approach to supplement range and range rate measurements made by the other navigation equipment.

A frequent target vehicle for the Orbiter is the Russian space station Mir. A typical scenario for the Orbiter to rendezvous and dock with the Mir is as follows. As the Orbiter approaches, its rendezvous radar will begin tracking the Mir and measuring range, range rate, and bearing. The Orbiter crew will also begin air-to-air communications with the Mir crew using a vhf radio. As the Orbiter reaches close proximity to the Mir, the TCS supplements the Orbiter's navigation information by supplying additional data on range, range rate, and bearing. In addition, the crew begins using the hand-held laser ranging device to supplement the other measurements of range and range rate. The Orbiter crew will fly the Orbiter toward the Mir using aft flight deck controls. Viewing displayed images from the centerline camera, the Orbiter crew will center the Orbiter docking mechanism with the Mir docking module mechanism, continuously refining this alignment as the Orbiter approaches within 90 m of the Mir. At a distance of about 9 m from the Mir, the Orbiter crew will stop the Orbiter, station keep momentarily, and adjust the docking mechanism alignment, if necessary. Then, a go or no-go decision to proceed with the docking will be made by ground flight-control teams in both the United States and Russia. When the Orbiter proceeds with docking, the Orbiter crew will use ship-to-ship communications with the Mir to inform the Mir crew of the Orbiter's status and keep them informed of major events, such as the confirmation of contact, latch up, and the completion of damping. Damping is the decaying relative motion between the Orbiter and the Mir that occurs after docking and is positively affected by the shock-absorber-type springs within the docking device. These springs also help to gently push the Orbiter away from the Mir during undocking.

The Russians took a different approach to rendezvous and docking, one that was primarily automated with the crew being used for monitoring and manual backup functions. Their effort in ARD dates back to October 1967 when they joined two uncrewed Cosmos spacecraft in orbit.[9,10] The ARD system that they have developed and refined over the years has been used repeatedly for docking the uncrewed Progress and the crewed Soyuz vehicles to the Mir. They also plan to use this system for docking their vehicles to the ISS. Interestingly, the hardware in this system is similar to that employed in the manual rendezvous and docking systems flown by the United States. It includes guidance digital computers; IMU-type inertial sensors, i.e., rate gyros and accelerometers; optical devices for inertial sensor alignment, rendezvous radar; and television cameras. The rendezvous radar system is called Kurs. Appropriate displays and controls are available to the flight crews and ground controllers to fly the system manually, if desired or required. The displays include both data and television camera images.[11–13] Docking devices employed include both the rod-and-cone-type system and the androgynous peripheral assembling system[9]; see Figs. 5 and 6, respectively. The latter was originally developed for the Apollo–Soyuz rendezvous and dock in 1975 (Ref. 14).

The ARD scenario for rendezvous and docking the Soyuz or the Progress vehicles to the space station Mir is as follows[15]: The process begins with the Mir transmitting a beacon radio frequency signal from hemispherical-coverage antennas on the ends of its solar panels. The chase vehicle, which could be either the Soyuz or the Progress, has a gimbaled, 0.5-m dish antenna that searches for this signal. The 0.5-m dish antenna system can detect and acquire it up to 200 km away. Once this is accomplished, the gimbaled antenna then begins to angle track the signal from the Mir. At this point, the rf beacon signal is turned off, and a transponder on the Mir is connected to the antennas on its solar arrays. The chase vehicle now uses the return signal from the transponder to determine range using the time delay and range rate using the Doppler shift of the returned signal. Using this information, the chase vehicle closes in on the Mir until a range of about 200 m. The chase vehicle then executes a fly-around maneuver at a constant range of 200 m, until signals from three docking antennas mounted around the selected docking port on Mir are received. Each docking port on Mir has three docking antennas like these. The transponder on Mir now begins to transmit through one of these antennas to provide range and range-rate information to the chase vehicle. Relative attitude is also derived from the signals received from the three docking antennas (Ref. 1, pp. 34 and 35). The chase vehicle now proceeds with the approach. At 20-m separation, relative attitude can no longer be derived from the docking antenna signals. Now, the integrals of the rate gyro outputs are relied upon for attitude information. The automatic docking process can be aborted by turning off passive equipment on the Mir. Then, the chase vehicle performs an automatic back-away maneuver. When the Soyuz is the chase vehicle, it can be manually docked by the Soyuz flight crew using hand controls and displays.[11] When the Progress is the chase vehicle, it too can be manually docked. In this case, docking is accomplished by the Mir crew or ground controllers, using similar hand controls and displays.[12,13,16] Commands from the hand controls and data for the displays are telemetered between the Mir and the Progress and the Mir and the ground stations.

Some comments about the Russian ARD system are in order. Although this system does the job for which it was designed, it has some significant drawbacks. The Kurs radar system can only be procurred from one source, namely, Energia Scientific Production Association (NPO Energia) in Russia. Its electronics consume a lot of power, must be cooled by forced air, and use vacuum tube technology with questionable lifetime. Little is known about the construction of the electronic parts. Their docking devices require high-impact loads to latch up.

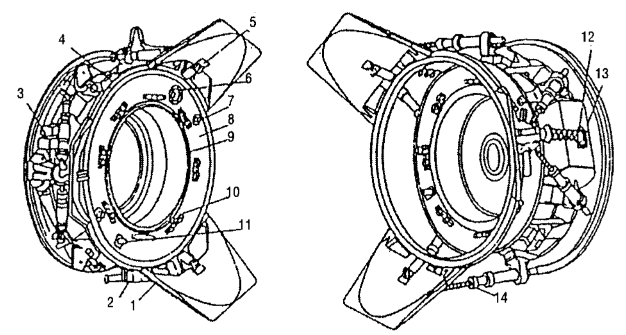


Legend: (1) Active Docking Assembly; (2) Socket; (3) Docking Mechanisms; (4) Guide Rods;(5) Passive Docking Assembly; (6) Receiving Cone; (7) Socket; (8) Grooves for Latches; (9) Electrical Connectors

Fig. 5 Soyuz docking assemblies.

Apollo side shown in passive configuration

Soyuz side shown in active configuration



Legend: (1) Ring with guides; (2) Hudraulic shock absorbers; (3) Docking mechanism drive; (4) Latch catch; (5) Latch; (6) Socket; (7) Push rod; (8) Docking frame; (9) Seal; (10) Lock; (11) Guide rod; (12) Spring cable; (13) Differential unit; (14) Screw with converter

Fig. 6 Apollo/Soyuz docking mechanisms.

Need for ARC in Space

The need for ARC in the U.S. space program surfaces periodically in two distinct places. One is in the autonomous delivery of uncrewed vehicles to the ISS for reboost and/or resupply. The other is in the execution of uncrewed and crewed missions to and from Mars.

In the early 1990s, the cargo transfer vehicle (CTV) was conceptualized as an uncrewed, orbital stage for the U.S. National Launch System (NLS). One of the functions of the CTV was to resupply the ISS by transferring payloads from the NLS to the ISS (Ref. 1, pp. 2–4). Hence, an automated, active uncrewed space vehicle was to operate in the vicinity of and dock with an essentially passive, crewed space vehicle. This requirement for ARC led a comprehensive evaluation and review of the U.S. capabilities in ARC in 1991 (Ref. 17). Another independent assessment of it was made in 1993 (Ref. 18). In both cases, the conclusion was that the ARC capability required by the CTV and other vehicles on the horizon did not exist and needed to be developed. Somewhere in this timeframe, it was decided to resupply the U.S. part of the ISS with the Space Shuttle and the CTV and the NLS projects were canceled.

In the fall of 1993, the Russians became an active participant in the ISS program. Their involvement included building the functional energy block and the service modules for periodically reboosting the ISS. They planned to use the ARD system that they developed for the Soyuz and Progress vehicles to deliver the uncrewed service modules to the ISS. In 1995, there was concern that the Russians did not have the wherewithal to build the service modules as they had committed to do. Then NASA began defining the requirements for a vehicle to replace the service modules, in case the Russians could not deliver them. This vehicle was called the U.S. Control Module (USCM) and had a requirement for ARC.[19] Derived capture requirements for the USCM to be able to capture with the ISS were generated and are as follows. The position of the USCM relative to the ISS must be controlled to ± 1.5 cm in each axis. The attitude of the USCM relative to the ISS must be controlled to ± 0.5 deg in each axis. The magnitude of the linear velocity of the USCM center-of-mass relative to the ISS must be controlled to ± 1.5 cm/s. Subsequently, it was determined that the Russians would indeed build the service modules, and the USCM project was canceled.

Now on the horizon is the VentureStar reusable launch vehicle (RLV), being developed in the United States. It will be primarily an uncrewed vehicle, which necessitates a fully autonomous capability. An important use of the VentureStar RLV will be to ferry cargo to and from the ISS, which again leads to the requirement for ARC. Again the need for ARC to autonomously deliver uncrewed vehicles to the ISS for reboost and/or resupply has surfaced. This is a requirement that just will not go away.

ARC has also been identified as a needed technology for executing uncrewed and crewed missions to and from Mars. In the mid-1970s, studies of an uncrewed Mars sample return mission showed the need for ARC in Mars orbit to reduce the payload required so that spacecraft of the day could be used to execute the mission.[20,21] In this concept, a Mars orbiter and a Mars lander make the journey to Mars. The orbiter is inserted into Martian orbit while the lander descends to the surface. A 1-kg soil sample is collected and stowed in a sample canister on the ascent stage of the lander. The ascent stage then lifts off from the surface into orbit around Mars. Subsequently, the orbiter rendezvous and captures with the ascent stage. The soil sample canister is then transferred to the orbiter. Now, the ascent stage separates and the Earth return portion of the orbiter makes its way back to Earth. Whereas the Mars sample return mission studied in the 1970s never proceeded to fruition, new studies of Mars sample return missions have recently begun, with assumed launch dates in 2005 and 2007. These studies are producing a mission concept that is similar to the one generated in the 1970s, with a requirement for ARC. The following is a leading scenario for the ARC concept associated with these missions.

Star trackers and IMU rate sensors on the orbiter enable the attitude and attitude rate of this spacecraft to be accurately determined and controlled onboard. The orbit ephemeris of the orbiter is accurately determined on Earth, based on ground tracking of the spacecraft. This is accomplished by transmitting from Deep Space Network ground stations on Earth to the spacecraft an extremely accurate and stable carrier frequency modulated with a pseudorandom signal. A transponder on the spacecraft retransmits this signal back to Earth at a different carrier frequency. Based on the time delay between the transmitted and the returned signal, range is determined. Based on the Doppler shift of the returned signal, range rate is determined. Observing these parameters over a period of time allows the orbit ephemeris of the orbiter to be accurately determined on Earth.

Star trackers and IMU rate sensors on the ascent stage enable the attitude and attitude rate of this spacecraft to be accurately determined and controlled onboard. The orbiter is equipped with a transceiver. It also has a direction finder with antennas and associated electronics. The ascent stage has a transponder. The orbiter transmits a pseudorandom encoded rf signal to the ascent stage transponder, which retransmits it back to the orbiter but at a different carrier frequency. Based on the time delay between the transmitted and the returned signal, relative range between the two spacecraft is determined onboard the orbiter. The relative-range rate between the two spacecraft is determined onboard the orbiter by differencing range measurements and dividing by the time interval between these measurements. Information derived from the returned signal from the ascent stage that is received by the direction-finder antennas enables the direction-finder electronics to compute the direction of the relative-range vector in orbiter body-fixed axes. The relative-range magnitude, direction, and rate-of-change vs time are then transmitted back to Earth. This information and the estimated orbit ephemeris of the orbiter enable the ground team to accurately compute the orbit ephemeris for the ascent stage.

Knowing the orbit ephemeris of both spacecraft on Earth enables the ground team to compute the orbiter delta-velocity commands required for the orbiter to rendezvous with the ascent stage. These delta-velocity commands and the associated orbiter attitude commands are transmitted to the orbiter from the ground and executed by the orbiter. Using this process and iterating, the orbiter can rendezvous to within 1 km of the ascent stage. At this point, the transceiver and direction finder on the orbiter, the transponder on the ascent stage, and the orbiter's GNC system are used to autonomously rendezvous the orbiter to within 100 m of the ascent stage. At this point, an optical guidance system on the orbiter determines the relative position and orientation of the orbiter with respect to the ascent stage. This information and measurements from orbiter IMU accelerometers and rate sensors are input into the orbiter's onboard GNC system and used to autonomously execute the terminal phase of rendezvous and capture with the ascent stage or the soil sample canister on it. Required accuracies of the orbiter optical guidance system, the IMU, and the closed-loop GNC system for rendezvous and capture are on the order of 1 cm, 1 cm/s, and 1 deg. The required accuracy of the orbiter and ascent stage star trackers for attitude determination and control is in the neighborhood of 7 arc-min.

Recently initiated studies of a human Mars mission in 2011 are also generating a requirement for ARC, both in low Earth orbit (LEO) and in Mars orbit. The following is a leading scenario for the ARC concept in LEO, assuming a chemical trans-Mars injection (TMI) stage. The liquid oxygen (LOX)-only element of the TMI stage is launched into LEO first. It plays both active and passive roles in the ARC process. It has global positioning system (GPS)/inertial navigation system (INS) for attitude, attitude rate, position, and velocity determination. On one end is a uhf transmitter and a video guidance sensor (VGS) target.[22] These are utilized when the LOX-only element plays the role of a passive vehicle in the ARC process. On the other end is a VGS, which is used in the terminal phase of ARC when this LOX-only element is part of the active vehicle in the ARC process. The payload vehicle will be launched into LEO second. It plays an active/chase-vehicle role in the entire ARC process. It has a propulsion system for orbit adjustment. It has a reaction control system (RCS) system configured for attitude and translational control. It has GPS/INS for attitude, attitude rate, position, and velocity determination. It has a uhf receiver to receive GPS information transmitted by the passive/target vehicle for relative GPS. It has a VGS for the terminal phase of ARC. The LOX/liquid hydrogen (LH_2) element of the TMI stage is launched into LEO last. It is a passive/target vehicle in the ARC process. It has an RCS for three-axes attitude stabilization. It has GPS/INS for attitude, attitude rate, position, and velocity determination. It has a uhf transmitter to transmit GPS information to the active/chase vehicle for relative GPS navigation on the active/chase vehicle. It has a VGS target for the terminal phase of ARC.

After the LOX-only element has been launched and inserted into LEO and prior to launching the payload vehicle, the orbit ephemeris for the LOX-only element is accurately determined by the ground team. This information is loaded into the payload vehicle's onboard computer (OBC) while it is still on the launch pad. After the payload vehicle is launched and separated from its launch vehicle, the payload vehicle's GPS information begins to update the state vector and attitude propagated by its INS. The payload vehicle's OBC software now begins to determine the orbit ephemeris for the payload vehicle using the GPS/INS state vector estimates. Every 12 h, the ground team uplinks new parameters for the LOX-only element orbit ephemeris model that is stored in the payload vehicle OBC. Based on the onboard orbit ephemeris models of both vehicles in the payload vehicle OBC, the OBC software computes and issues commands to execute a series of phasing maneuvers for the payload vehicle to approximately align the orbit plane of the payload vehicle with that of the LOX-only element. The propulsion system provides the thrust for these maneuvers. The RCS system generates the torques required for attitude control. Now, translational maneuvers are computed and executed by the payload vehicle to maneuver itself close enough to the LOX-only element so that it can receive GPS information transmitted by the LOX-only element's uhf transmitter. Typically, the payload vehicle must be within 7 km of the LOX-only element for this to happen.

At this point, relative GPS is used for navigation, and the RCS system is used for both translation and attitude control, to rendezvous the payload vehicle to within about 100 m of the LOX-only element. Now, sensing for rendezvous and capture is transferred to the VGS on the payload vehicle. A target for the VGS is mounted to the LOX-only element. The VGS determines the relative position and orientation of the payload vehicle with respect to the LOX-only

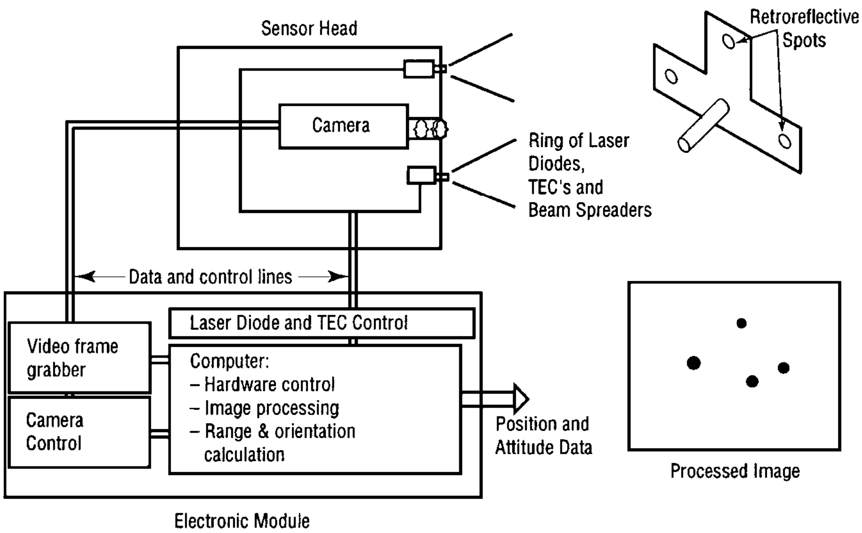


Fig. 7 VGS diagram.

element. This information and measurements from the INS are used to autonomously execute the terminal phase of rendezvous and capture with the LOX-only element. A zero-velocity capture is required. The required accuracies of the payload vehicle VGS, INS, and closed-loop GNC system for rendezvous and capture are on the order of 1 cm, 1 cm/s, and 1 deg. At this point, the ARC process is repeated with the LOX/LH_2 element as the target/passive vehicle and the LOX-only-element/payload vehicle as the active/chase vehicle.

In the human Mars mission, ARC is also needed in Mars orbit to return the crew to Earth. In this case, the crew will be relegated to manually backing up the automated procedures for the following reasons. It will take them 180 days to reach the planet. Following this, there will be a 500-day stay on the surface. Then, the rendezvous and capture procedures will be executed in Mars orbit, nearly two years after the crew first left Earth. Because they will not be able to practice rendezvous and capture techniques either en route to Mars or on the planet, it is risky or even dangerous to rely on them as the primary means for accomplishing this critical phase of the mission. Hence, ARC is the primary approach. A promising scenario for accomplishing it is similar to the one described earlier for the Mars sample return mission.

Whereas ARC will be needed for autonomous resupply of the ISS and the execution of uncrewed and crewed missions to and from Mars, other requirements for ARC are also on the horizon. In the late 1980s and the early 1990s, the orbital maneuvering vehicle (OMV) was conceived as an ISS-based vehicle for autonomous satellite retrieval and servicing.[23] It had a requirement for ARC. This vehicle was never built because of funding problems in the post-Challenger era, but spaced-based systems that provide the capability for autonomous satellite retrieval and servicing are still being studied.[24,25] The U.S. military has a future need for ARC with space vehicles that it is developing, such as the military spaceplane and the space maneuver vehicle.[26,27] It plans to do satellite retrieval with both cooperative, i.e., stabilized, noncooperative, i.e., nonstabilized, and uncooperative, i.e., evasive, targets. The crew return vehicle (CRV) that NASA is building as a lifeboat for the ISS crew will need ARC if it is to be delivered to the ISS autonomously. Advanced concepts for space solar power generation rely heavily on ARC for assembling large power-generating systems in Earth orbit, one piece at a time, cheaply and autonomously.[28]

Today's Technology and Ongoing Technology Efforts Related to ARC

Having defined the U.S. needs and requirements for ARC in space, the next step is to review today's technology and ongoing technology efforts related to ARC. Then, it can be determined whether and how these can contribute toward meeting the future U.S. needs and requirements for ARC. Hence, the remainder of this section will be devoted to reviewing the past and present ARC-related technology efforts around the world.

An ARC technology program for the OMV began in 1987 (Ref. 22). Although the OMV was eventually canceled, the ARC technology program continued because of the need for this technology on other programs such as the CTV, the USCM, the RLV, the CRV, and the Mars missions. Principal products of it have been the VGS, precise navigation algorithms for relative GPS, and the three-point docking mechanism (TPDM).

The VGS is an optical sensor that measures the range, bearing, and attitude of the chase vehicle relative to the target vehicle in the terminal phase of ARC, out to about 100 m (Ref. 22). It consists of a sensor head assembly on the chase vehicle and a target on the target vehicle. The sensor head assembly has 10 laser diodes, a solid-state video camera, a video frame grabber and digitizer, and a microprocessor (Fig. 7). Five of the laser diodes operate at 780-nm wavelength, while the other five operate at 830 nm. Each diode emits 30 mW of laser light in approximately a 10×30 deg beam. The diodes are arranged to produce approximately even light intensity over a 30×30 deg field of view at each wavelength. Hence, the target is illuminated with equal light intensity at both wavelengths. In the concept of Fig. 7, the target has four corner-cube retroreflectors. The middle retroreflector is mounted on a pole. In front of each retroreflector is an optical bandpass filter, with a center frequency corresponding to 830-nm wavelength. The filtered retroreflectors will reflect light at 830-nm wavelength but will filter it at 780 nm. This allows the target to be more easily discriminated from background clutter. The 830-nm laser diodes and the 780-nm laser diodes are alternately turned on and off. Digitized pictures of the alternating scenes are acquired using the video camera, the frame grabber, and the microprocessor. These are subtracted in the microprocessor to produce images of the target retroreflectors. From these images, relative range, bearing, and relative attitude are computed twice per second. The accuracy of the computed relative range is ± 0.3 cm in each axis. The accuracy of the computed relative attitude is ± 0.25 deg in each axis. The relative range rate can be derived from the relative range computations to an accuracy of ± 0.3 cm/s in magnitude. All of these are well within the system requirements previously specified for the USCM to capture with the ISS or for ARC to be successfully executed in the Mars missions.

The VGS concept just described has a passive target. An alternative concept is to make the target active by using laser diodes on it, instead of on the sensor assembly, and eliminating the corner-cube retroreflectors on the target. This approach offers the potential for significant savings in power and mass and is preferred in Mars-mission applications, where these parameters must be minimized. The passive target approach is more attractive in ISS applications, where minimizing power and mass is not as important as avoiding the need to deliver power to the target.

The VGS was flight tested on the STS-87 Shuttle mission in November 1997. A passive target was mounted on a free-flying Spartan spacecraft. The sensor head assembly was mounted in a get-away special canister in the Shuttle payload bay. The plan was to deploy the Spartan with the Shuttle remote manipulator system (RMS). The Shuttle was to then back away from the Spartan and then reapproach it while the VGS generated open-loop measurements of range, bearing, and attitude relative to the Spartan. However, the Spartan experienced power-up problems upon deployment, and this part of the mission never went as planned. No VGS data were collected with the Spartan deployed. However, 10 min of VGS data was collected with the Spartan attached to the end of the RMS. These data verified the predicted performance of the VGS at close ranges, on the order of 10 m. The VGS and the Spartan will be reflown on STS-95 in October 1998 with the goal of collecting data from 10 to 100 m.

Another technology that has been developed for ARC has been precise navigation algorithms for relative-GPS operation.[29] Specifically, a 19-state Kalman filter was developed that processes relative-GPS measurements to estimate relative range to an accuracy that is on the order of 1 m. This technology development continues with the goal of augmenting the 19-state Kalman filter with 13 additional states so as to estimate relative range to an accuracy that is on the order of 1 cm.

The TPDM is a device that was developed for a zero-velocity capture. It consists of three claws on the chase vehicle and three trunnion bars on the target vehicle. The claws have multiple light-beam sensors that detect when a claw has passed around a trunnion bar. When two claws have captured their trunnion bars, they begin to close, which aids the third claw in capturing its trunnion bar.

The TCS is a device that has potential applications in ARC. It was originally developed as a crew aid for docking the Space Shuttle Orbiter to the space station Mir. The plan is to use it in a similar manner for docking the Space Shuttle Orbiter to the ISS. It mounts in the Orbiter's payload bay; target optical retroreflectors are mounted on the Mir. The TCS generates range, range rate, and bearing (azimuth and elevation) information relative to the Mir and displays this to the Orbiter crew at ranges varying from 1.5 km to 1.5 m. Range and range rate are determined to accuracies of ± 3 cm and ± 3 cm/s in each axis, respectively. This capability is adequate for the proximity operations phase of ARC (400 km–100 m) but is marginal for the terminal phase (less than 100 m) in light of the ARC performance requirements specified earlier. This device uses a pulsed laser to measure ranges from 1.5 km to 400 m. At ranges closer than 400 m, a more-accurate continuous-wave (CW) laser diode modulated with three tones is used. The laser beams are scanned using a three-axes galvanometric beam-scanning system, which provides for a 20-deg-radius cone field of regard. A description from Ref. 30 follows:

> In the acquisition phase, the laser is scanned within the field of regard until a retroreflector is illuminated on the target vehicle. A quality retroreflector has the unique property that the incident and reflected laser beams are coaxial over about a 30 optical degree cone of angle. The laser energy returned by the retroreflector is detected and the range information for the near field CW operation is derived from the measured phase shift between the transmitted and received tones. This phase shift is due to the round trip travel time caused by the finite speed of light. Range rate is determined by back-differencing the CW range data. The range data from the pulsed operation is derived by time of flight. A high-speed counter is used to determine the elapsed time from when the leading edge of a pulse is transmitted and when it is received after reflection from a target vehicle. The time delay is due to the round trip travel time caused by the finite speed of light. Range rate in this mode is determined by back-differencing pulsed range data. The azimuth and elevation angles are determined by the position of the optical scanners at the time a retroreflector was encountered. The azimuth and elevation rates are determined by back-differencing of azimuth and elevation data.

A scannerless range imager (SRI) sensor is also being developed that may have application as a ranging device in ARC. The plan is to use it on the autonomous extravehicular robotic camera (AERCam) for ISS structural vibration identification.[31] In this scenario, the AERCam flies around and points at the ISS. The SRI sensor output provides accurate range and bearing information on the ISS structure. This information collected over time gives the structural dynamic characteristics of the ISS. The SRI sensor consists of an amplitude-modulated floodlight scene illuminator (laser or light-emitting diode transmitter), a gain modulated image intensified charge-coupled device (CCD) video camera, and a digital signal processor that transforms intensified video imagery into range imagery. The SRI concept is based on CW optical radar technology. It uses a low-cost focal plane array integrating-type detector and works by measuring the phase difference between a transmitted intensity-modulated optical signal and the corresponding reflected return signal from a target scene. A test flight of the AERCam with the SRI sensor is scheduled for December 1998 on the STS-96 Space Shuttle mission.

The space vision system (SVS) is being developed for use as an aid in ISS assembly with the Space Shuttle RMS and the ISS RMS. It is conceivably applicable to ARC but probably has more potential use for Space Shuttle man-in-the-loop rendezvous and docking operations. The SVS uses existing Space Shuttle payload bay camera views of targets on payloads and payload bay hardware to provide precise relative position, attitude, and rate cues in a concise graphical and digital format.[32] It was tested on the Space Shuttle STS-52 mission and provided RMS operators with precision position and attitude cues to support unberthing, maneuvering, and berthing operations. An upgraded version of the SVS was then flown as a detailed test objective (DTO) on the Space Shuttle STS-74, STS-80, and STS-85 missions to further evaluate its on-orbit performance. It will fly again as a DTO on STS-91 in May 1998. For this mission, the plan was to test the feasibility of using it to provide range, range rate, and bearing for Space Shuttle proximity operations associated with man-in-the-loop rendezvous and docking. However, the flight software to do this was never developed in time.

The space integrated GPS/INS (SIGI) has application for rendezvous in LEO.[33,34] This unit is being developed as a standardized, highly integrated, autonomous navigation system with a GPS position, velocity, and attitude capability integrated with an advanced inertial system. It has a Trimble Force-19 GPS receiver that reflects an integration of Trimble tactical airborne navigation system TANS Vector and Force-19 GPS receivers. This is coupled with a Honeywell H-764G IMU. SIGI will offer three independent navigation solutions: INS, GPS, and blended GPS/INS. It has been undergoing flight testing and demonstration on the Space Shuttle and flew most recently on the Space Shuttle STS-89 mission in January 1998. Upon the completion of flight qualification in 1998, it will ultimately be applied to a variety of space vehicles, including the Space Shuttle, the ISS, and the CRV.

There are ARC-related technology developments that will be used on the Deep Space Mission 3 (DS-3) that is scheduled to begin in the year 2002. Here, three spacecraft must be maintained hundreds of meters apart to an accuracy on the order of centimeters. The sensor for accomplishing this is the autonomous formation flying (AFF) sensor.[35,36] The AFF sensor estimates the relative range and relative attitude between two spacecraft using GPS-type technology, although observations of GPS satellites are not required. Hence, it can be used in deep space or in LEO, with or without GPS satellite data. With this device, one spacecraft emits a pseudorandom encoded rf signal. GPS antennas and a receiver on the other spacecraft receive this signal and from it determine relative range and relative orientation. It can be used for rendezvous in a relative range varying from 1300 km to 10 m. It has the potential for use down to 1 m, but this is pushing the envelope. At 1-m separation, electronic gain changes are required to reduce the rf energy emitted and avoid burning up hardware. Studies indicate that between 1300 km and 10 m this sensor can measure relative range to an accuracy of 1 cm and relative attitude to an accuracy of 1 arc-min, assuming a 1-m separation of antennas. Multipath is a concern when the two spacecraft are close together, but engineers have developed an algorithm to compensate for it. Presently, the sensor can be regarded as having a technology readiness level (TRL) of 3 (TRL-3) (analytical and experimental critical function and/or characteristic proof of concept) to TRL-4 (component and/or breadboard validation in laboratory environment). To date, ground demonstrations of the AFF sensor have been made using L-band rf signals, which have a 20-cm wavelength. Ground demonstrations need to be done with Ka-band rf signals, which have a 1-cm wavelength, to achieve better accuracies. Besides using the AFF sensor for DS-3, the plan is to use it for Deep Space Mission 4 and the Mars sample return missions.

In the early 1980s, the Europeans first began a program to develop a capability in ARD, ARC and automated rendezvous and berthing, with several future applications in mind. They saw future needs for these technologies to autonomously deliver: their future crewed space shuttle Hermes to their future crewed space station Columbus, Columbus to the ISS, Hermes to the ISS, and their uncrewed automated transfer vehicle (ATV) to the ISS.[37,38] Manual override of the automated procedures by the flight crews on Hermes, Columbus, and the ISS, plus flight controllers on the ground, was envisioned.[39] Whereas Hermes and Columbus should be viewed as long-range programs, the Europeans did commit in 1995 to build the ATV for refueling the ISS, reboosting the ISS orbit, delivering cargo to the ISS, and removing and destroying waste from the ISS (Ref. 16, pp. 121–124). Its first flight is scheduled for 2003, with a probable flight every 15 months until 2013 (Ref. 40). The ATV will be launched by an Ariane 5. Upon separation, it will attain a circular orbit. After 46 h of phasing, it will rendezvous with the ISS and dock to the Russian service module attached to the ISS using the same docking port as the Russian Progress vehicle.[41] During the final approach that leads up to docking, it will execute a collision avoidance trajectory; if it suffers a major failure, it will automatically back off. Once docked to the service module, it will remain there for up to six months and be used several times to reboost the ISS. ISS waste will be transferred to it, and it will separate from the ISS, deorbit, and disintegrate as it re-enters the Earth's atmosphere. Another ATV will eventually take its place.

The baseline GNC concept for executing ATV ARD has three IMUs for basic navigation information, two GPS receivers for absolute and relative position during the phasing and proximity operations segments of rendezvous, and a rendezvous sensor for relative range and relative attitude during the terminal phase. The rendezvous sensor is needed because shadowing and multipath effects do not permit the use of GPS during the last few meters of the approach.[42] It measures both relative range and relative attitude because strong coupling exists between these states during the last few meters, and these states must be controlled.[39] Two star trackers are provided for precise updates to the attitude derived from the IMUs. Two Earth sensors and two coarse sun sensors are provided for coarse attitude updates to this same information.[41] The rendezvous sensor is mounted on the forward section of the ATV. It emits a laser beam with a 905-nm wavelength. This beam is reflected by six retroreflectors mounted near the docking port of the Russian service module that is attached to the ISS. The reflected beam is detected by the rendezvous sensor on the ATV and processed to provide relative range, relative-range rate, and bearing for the last 200–100 m and relative attitude and relative-attitude rate for approximately the last 40 m (Ref. 43).

Because relative GPS and the rendezvous sensor are critical to successful ATV ARD and both are new technologies with significant unknowns, three flight experiments were flown in space to test them.[43] A relative-GPS experiment was flown on the Space Shuttle STS-80 mission in November–December 1996. Here, the Orfeus-Spas satellite was deployed from the Orbiter payload bay with the RMS. It had a GPS receiver and target retroreflectors mounted on it. A second GPS receiver and a TCS were mounted on the Orbiter. GPS data for each spacecraft were recorded separately and correlated postflight to generate relative-GPS measurements. These data were then compared with recorded data from the TCS, which acted as a truth sensor for the relative-GPS data.

Next, a relative-GPS and rendezvous sensor experiment was flown on the Space Shuttle STS-84 mission in May 1997. Here the Orbiter rendezvous and docks with the Russian space station Mir. Mounted on the Mir were a GPS receiver and target retroreflectors for both the TCS and the European rendezvous sensor. Mounted on the Orbiter were a GPS receiver, the TCS, and the European rendezvous sensor. Relative-GPS data were generated as on STS-80. The data and recorded measurements from the rendezvous sensor were compared with recorded measurements from the TCS, which again acted as a truth sensor. A third flight experiment was flown on the Space Shuttle STS-86 mission in September 1997. It was essentially a reflight of the one on STS-84. All three flight experiments did experience some problems. The results from all three are still being analyzed. The total cost of these three flight experiments, several preflight ground simulations, and postflight analysis of the flight data was $\$4.3 \times 10^7$ not counting flight experiment launch costs.

The Japanese have made a big commitment to the development of automated and remotely controlled systems for rendezvous and dock, capture, and berth. As stated by Yamagata, a project manager in this technology area: "unmanned rendezvous systems are very important for the 21st century for [National Space Development Agency] NASDA and the world because of the cost savings it will give us."[44] The Japanese consider capabilities in uncrewed docking, capture and berthing, and robotics as essential for their plan to conduct uncrewed servicing of future spacecraft, especially the Japanese experiment module (JEM) of the ISS. They plan to deliver supplies to the JEM with their H-2 orbiting plane (HOPE), which is an uncrewed miniaturized version of the U.S. Space Shuttle Orbiter.[45] They also plan to resupply the JEM with their uncrewed H-2 transfer vehicle (HTV), which is a 13-m-ton resupply vehicle that is similar to the Russian Progress vehicle.[46] Both are scheduled to make test flights in the year 2001 (Refs. 44 and 47).

Japan also intends to use the HOPE with an RMS "to refuel space platforms, change out experiment modules, and perform repairs."[44] A proposed Japanese orbital servicing vehicle (OSV) has similar, but even more ambitious, goals, as stated in Ref. 48:

> The initial OSV will be operated based on the Space Station. The major missions of the OSV are assumed to be: (1) Deployment and retrieval of unmanned co-orbiting platform; (2) Changeout payloads on platform; (3) Exchange of failed equipment on platform; (4) Resupply of consumable to platform; and (5) Supply of materials to and retrieval of products from mission payloads. To perform above missions accurately, the initial OSV has a capability of automatic maneuver including automatic rendezvous and docking. It has also a remote manipulator system. These are considered as key technologies of the OSV. The future upgraded OSV should have more autonomous ability, which will be helpful for more complicated missions such as: (1) Retrieval of non-cooperative objects, and (2) On-orbit construction or refurbishment of spacecraft. New advanced technologies such as robotics and artificial intelligence will be incorporated in the upgraded OSV.

A mission profile for the HTV to deliver payloads to the ISS is described by Yamanaka[46] as follows: The HTV is being designed to deliver 6-ton payloads to the ISS. It will be launched by the H-IIA rocket. After separation, the HTV autonomously executes a rendezvous sequence that consists of phase, height, and plane adjustment maneuvers. Eventually, it reaches the ISS and enters a berthing box. Then, all HTV thrusters are inhibited. Next the ISS RMS grapples the HTV and berths it to the ISS. Reference 44 describes a slightly different scenario for the terminal phase of rendezvous. It indicates that the HTV is remotely docked to the ISS, entirely by ground control. Whichever method is eventually employed, once the HTV is attached to the ISS, its payloads will be transferred to the ISS and disposables from the ISS will be transferred to it. Then the HTV will perform automatic departure from the ISS and destructive re-entry into the Earth's atmosphere.

For Yamanaka's scenario, the HTV GNC system employs GPS receivers for relative and absolute GPS, rendezvous laser radar, inertial reference units, accelerometers, and Earth sensor assemblies.[46] There is a GNC computer as well as an abort control unit for aborting rendezvous in case of an emergency. Absolute and relative GPS are used for navigation down to 500 m from the ISS. After that, the rendezvous laser radar is utilized.

To verify the rendezvous and robotic technologies required by the HTV, the HOPE, and advanced vehicles such as the OSV, the Japanese began a $\$2.6 \times 10^8$ project in 1990 called Engineering

Test Satellite-VII (ETS-VII).[44] The ETS-VII spacecraft consists of a chase vehicle and a target vehicle that are launched together on an H-II launch vehicle into a 550-km circular Earth orbit. Launch occurred on Nov. 28, 1997. Seven ARD tests are scheduled over an 18-month period that was to begin in February 1998. The basic goal is to demonstrate ground-controlleddocking maneuvers of uncrewed vehicles using a combination of GPS navigation and radar. However, the tests were also to include simulated equipment and component changeout using an RMS on the chase vehicle that is ground controlled.

A typical scenario for the rendezvous-and-docking tests is as follows. Upon orbit insertion, the chase vehicle separates from the target vehicle and backs off to a maximum distance of 10 km from the target vehicle. It then automatically approaches the target vehicle using relative GPS and Clohessy–Wiltshire guidance.[46] Relative range is determined to an accuracy of 20 m. At a distance of 500 m from the target vehicle, navigation is switched from relative-GPS to rendezvous laser radar. Relative range and bearing are determined to an accuracy of 0.1 m and 0.05 deg, respectively. This is used for the approach until the chase vehicle is 2 m from the target vehicle. At this point, a proximity camera sensor is used to determine relative position and attitude by measuring a two-dimensional CCD image of a three-dimensional marker on the target vehicle.[46] This sensor measures relative range to 2 cm or better in each axis and relative attitude to 0.05 deg or better in each axis. Rate gyros and an Earth sensor are available for measuring chase vehicle angular velocity and attitude, respectively, throughout the whole process.

Since the ETS-VII mission began, it has been plagued with problems. Reference 49 reports: "Shortly after launch, the automatic Sun tracking function on its solar panel failed; engineers attributed the loss to a software problem that was fixed." Then a companion satellite, the communications and broadcasting engineering test satellite (COMETS), was launched into a nearly useless orbit on Feb. 21, 1998 (Ref. 50). COMETS has the task of relaying commands to the spacecraft from ground controllers for the rendezvous, docking, and robotic tests. The Japanese tried to salvage some use of the satellite through a series of orbit adjustment maneuvers that were scheduled for May 1998. As a backup, they can use the U.S. Tracking and Data Relay Satellite System.[51] Despite these and other problems, the first test was a success. Here, the chaser separated 2 m from the target vehicle and then redocked, thus verifying the close-in docking procedure.[52] However, the second test had problems. In this case, the chaser separated 500 m from the target vehicle but then mysteriously shutdown.[53]

Proposed ARC Systems for Meeting Future Needs

This paper has identified the need for ARC in the U.S. space program and has reviewed today's technology and ongoing technology efforts related to ARC. Now ARC systems will be proposed that satisfy the U.S. need for ARC and that utilize, where possible, today's ARC-related technology. The focus is on systems for ARC with cooperative target vehicles because this is the near-term need and logically precedes the development of ARC with noncooperative target vehicles. Two systems are proposed. Both utilize the TPDM for a true capture or zero-velocity dock.

One is a system designed for operation in LEO where GPS can be utilized for navigation. This limits its use to altitudes of around 15,000 km and below. Use of GPS above this altitude becomes more complicated and performance degrades. This is because the GPS satellites are at an altitude of around 20,000 km, and GPS was originally designed for navigation well below this. Therefore, this system uses GPS/INS for absolute and relative navigation. A good choice for a GPS/INS system is the SIGI that is being developed for applications such as the Space Shuttle, the ISS, and the CRV.[33,34] It generates position, linear velocity, inertial attitude, and angular velocity with absolute GPS. These are used to get the chase vehicle to within about 7 km of the target vehicle. At this point, relative GPS is employed to generate relative range, bearing, relative-range rate, relative attitude, and relative angular velocity. These are used to get the chase vehicle to within about 100 m of the target vehicle. The precise relative-GPS algorithms that have been developed and are being improved upon can be used very effectively here.[29] Close in GPS has problems with shadowing and multipath; hence, an optical sensor is needed for the terminal phase of rendezvous. Because of the coupling between relative position and relative attitude and the need to control these states close in, the optical sensor should generate both relative position and relative attitude. Tietz and Kelly[54] state this another way:

> A system that measures only the distance and direction to the target is adequate to approach within eight meters of the target. At this point, attitude information becomes vital because offsets among the docking aid, target docking fixture, and target center of mass become major contributors to alignment errors. The offsets among the camera, chase-vehicle center of mass, and chase-vehicle docking fixture make attitude information doubly important because chase vehicle attitude and position must be controlled. To further complicate the problem, the target may be coning and nutating, making it difficult to anticipate attitude changes.

The best choice for the optical sensor is the VGS because it measures both relative position and relative attitude.[22] For LEO ARC applications such as delivering uncrewed reusable launch vehicles to the ISS and autonomously assembling vehicles for a crewed mission to Mars, VGS power and mass are not critical parameters that need to be optimized. Also, for these applications, simplifying the target by avoiding the need to deliver power to it is highly desirable. This leads to the use of a passive target with corner-cube retroreflectors. The GPS/VGS system just described satisfies the need for ARC to autonomously deliver uncrewed reusable launch vehicles to the ISS for ISS resupply, deliver the CRV autonomously to the ISS, and allow uncrewed vehicles to be autonomously assembled in LEO for a subsequent human excursion to Mars. It would also satisfy the need for ARC in LEO to accomplish satellite-servicing missions with cooperative targets. It should also satisfy the need for ARC in assembling large power-generating systems in Earth orbit.

The second ARC system that is proposed is one that is designed to operate where GPS cannot be used for navigation, as in Mars orbit, lunar orbit, or Earth orbit above 15,000-km altitude. This system has IMUs and star trackers for inertial navigation and attitude determination, respectively. It uses the AFF sensor to estimate the range and attitude of the chase vehicle relative to the target vehicle from 1300 km down to 10-m separation.[35,36] Ground tracking should very easily be able to get the chase vehicle to within 1300 km of the target vehicle.[55] The TCS or the SRI sensor are possible alternatives to the AFF sensor. At 10-m separation and closer, the VGS is used for relative position and relative attitude. Because VGS power and mass are critical parameters in this application, an active target with laser diodes is needed. This eliminates the corner-cube retroreflectors on the target and the laser diodes on the VGS sensor head assembly. Microminiaturization of the electronics is imperative. With these changes, an order-of-magnitude reduction in power and mass is conceivable. For example, the VGS that was flown on STS-87 dissipated 65 W nominally. The sensor head assembly, including electronics and cables, had about 18-kg mass; the target had about 11-kg mass. With an active target and microminiaturized electronics, these numbers can potentially be reduced by a factor of 5–10. The AFF/VGS system just described can satisfy the need for ARC in Mars orbit for the uncrewed and crewed Mars missions. It can also satisfy any future need for ARC in lunar orbit or geosynchronous Earth orbit (GEO), should the need arise.

Technology Plan for ARC

The systems just described should satisfy all of the U.S. projected needs for ARC with a cooperative target vehicle. Eventually, the United States will want a capability in ARC with a noncooperative target vehicle for servicing and retrieving disabled satellites. A proposed technology roadmap for developing both capabilities is presented in Fig. 8. The development of ARC in LEO with a cooperative target vehicle and a passive target is shown concurrently in time with the development of ARC outside LEO with a cooperative target vehicle and an active target. Both are shown to begin in the fourth quarter of 1998. This is unlikely to happen because of funding constraints, and so one or the other will in all likelihood slide to the right of the chart with time. Hence, the system with the more

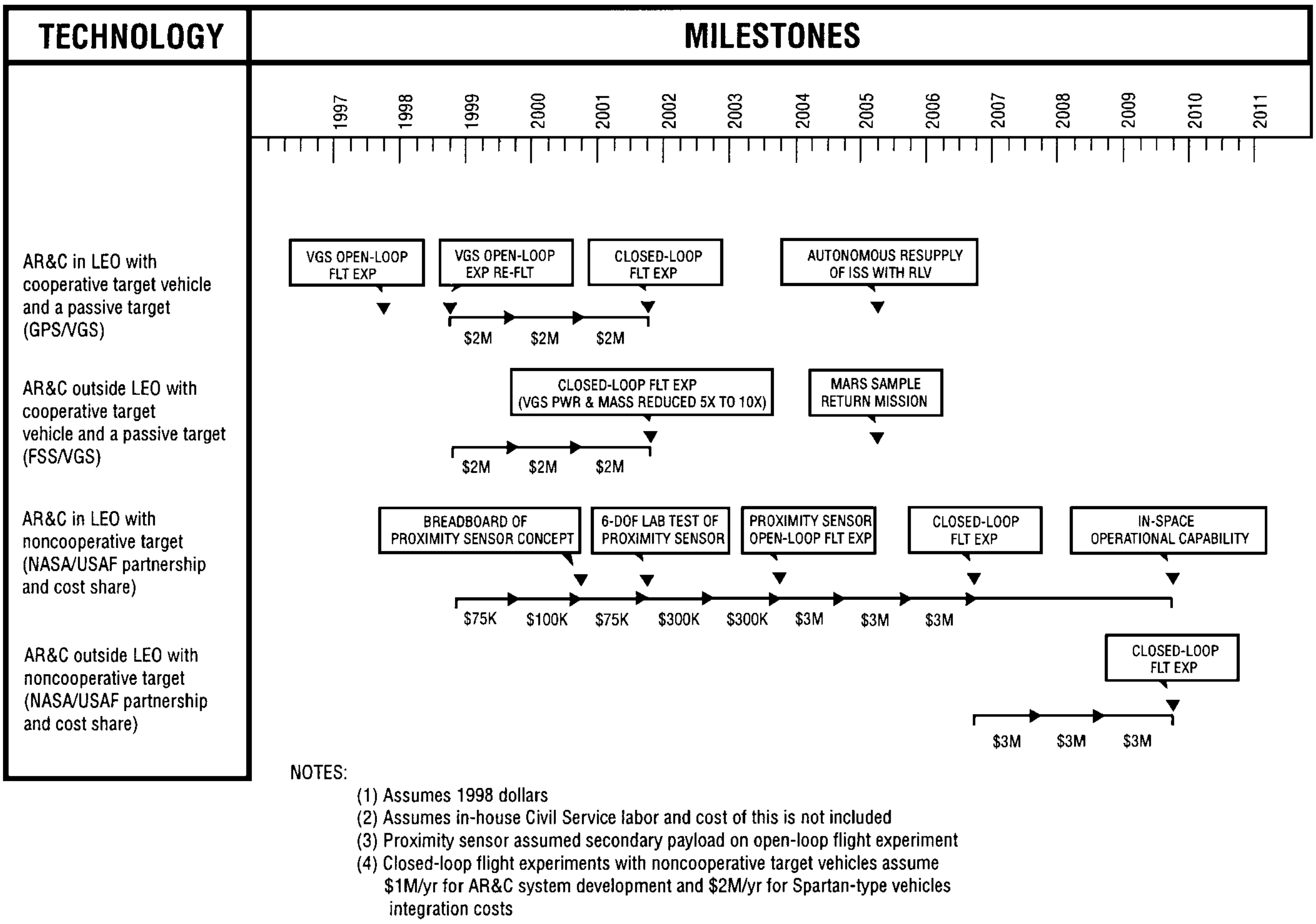


Fig. 8 Proposed technology roadmap for automated rendezvous and capture.

pressing need will be developed first, and the other will benefit from it. Note that separate ARC closed-loop flight experiments are proposed for testing the two systems described in the preceding section. It is unlikely that one flight experiment could be used to test both systems because of their uniqueness. Also, the complete systems need to be verified, not just the components in them. There are two attractive approaches to a closed-loop flight experiment for either system. These will be discussed in the remainder of this section for the system in LEO with a cooperative target vehicle and a passive target. Plans for the other system should be similar, with similar schedules and cost numbers.

The first approach to an ARC closed-loop flight experiment in LEO with a cooperative target vehicle and a passive target uses surplus and expendable manned maneuvering unit (MMU) hardware that is carried to orbit by the Space Shuttle Orbiter and deployed with the RMS. The MMU is a proven free-flying platform that can operate in either a pilot-monitored or uncrewed mode. The recommendation here is to fly it in an uncrewed mode. The MMU becomes the chase vehicle. A target and docking port for it is placed on the end of a mast, as shown in Fig. 9. This approach has been estimated to cost about $\$6 \times 10^6$ and take three years to complete.

An alternative approach uses two Spartan spacecraft, which are also carried to orbit by the Space Shuttle Orbiter and deployed with the RMS. One Spartan is modified to become the chase vehicle; the other is the target vehicle. This approach has also been estimated to cost about $\$6 \times 10^6$ and take three years to complete. Flying two Spartans on one Space Shuttle flight can be a challenge with regard to Orbiter center-of-gravity constraints and any manifest issues because two Spartans take up about one-fourth of the Orbiter's payload

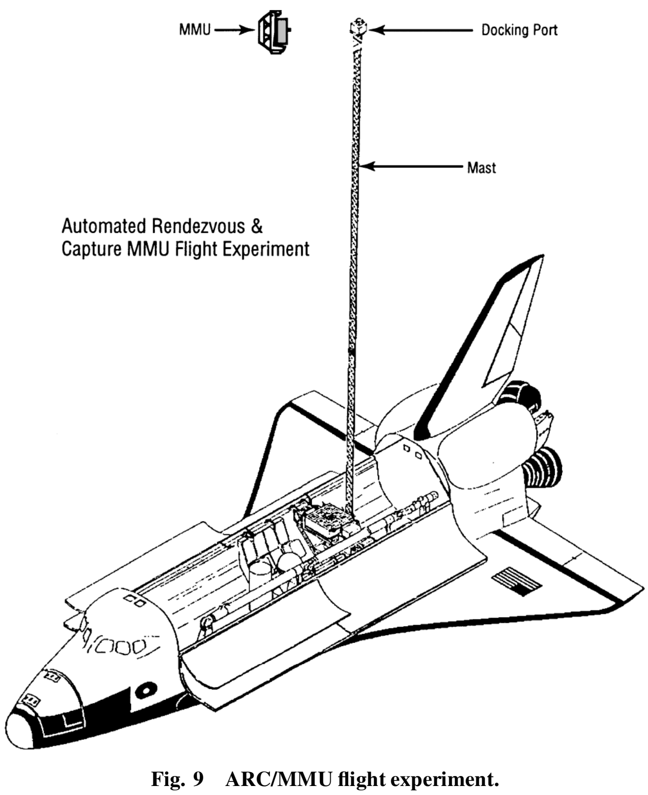


Fig. 9 ARC/MMU flight experiment.

bay. For this reason, the MMU approach is considered the more attractive option at this point. See Ref. 56 for more information on these options and a discussion of some other possibilities.

Conclusions

This paper has presented the results of a study into the technology of ARC in space. The main conclusion is that new ARC technology is needed for the United States to execute some future missions that are on the horizon. Two new ARC systems need to be developed. One is a system for ARC in LEO with a cooperative target vehicle and a passive target. This is needed to autonomously deliver the future U.S. RLV to the ISS. It is also needed for autonomously assembling uncrewed vehicles in LEO to execute a crewed mission to Mars. It can be used to autonomously deliver the CRV to the ISS. It can also be used for autonomously and cheaply assembling large power-generating systems in Earth orbit, one piece at a time. This system uses the SIGI GPS/INS for absolute and relative navigation during the phasing and proximity-operations segments of rendezvous. It uses the VGS with a passive target for the terminal phase. A closed-loop flight experiment to test this system in space using surplus MMU hardware costs about $\$6 \times 10^6$ and takes three years to complete.

Another new ARC system needs to be developed for ARC in Mars orbit to execute the return leg of an uncrewed Mars sample-return mission or a crewed mission to Mars. Once developed, this system can also be used for ARC in lunar orbit or GEO. This system has IMUs and star trackers for inertial navigation and attitude determination, respectively. It uses the AFF sensor for navigation during the phasing and proximity-operations segments of rendezvous. It, too, uses the VGS, but with an active target and microminiaturized electronics for the terminal phase. A closed-loop flight experiment to test this system using surplus MMU hardware also costs about $\$6 \times 10^6$ and takes three years to complete.

Eventually, the United States will want the capability for ARC with a noncooperative target vehicle both in LEO where GPS can be used and outside LEO where it cannot. This is for servicing and retrieving disabled satellites. A development program that parallels the one for ARC with a cooperative target should cost about $\$1.9 \times 10^7$ and take about 11 years to complete.

References

[1]"NASA Automated Rendezvous and Capture Review, Executive Summary," NASA TM-108597, Nov. 1991, pp. 2–4, 35, 36.

[2]Chiarrappa, D., "Analysis and Design of Space Vehicle Flight Control Systems, Volume VIII—Rendezvous and Docking," NASA CR-827, July 1967, pp. 3, 31, 46, 47.

[3]Burton, J. R., "Rendezvous with the Gemini Spacecraft," *Proceedings of the Symposium on Space Rendezvous, Rescue, and Recovery*, Vol. 16, Pt. 2, American Astronautical Society, Springfield, VA, 1963, pp. 173–176.

[4]Ballard, H. J., "Agena Target Vehicle for Gemini," *Proceedings of the Symposium on Space Rendezvous, Rescue, and Recovery*, Vol. 16, Pt. 2, American Astronautical Society, Springfield, VA, 1963, pp. 177–187.

[5]Quigley, W. W., "Gemini Rendezvous Radar," AIAA Paper 63-650, Aug. 1963.

[6]Felleman, P. G., "Rendezvous Requirements for the Apollo Mission," *Proceedings of the Symposium on Space Rendezvous, Rescue, and Recovery*, Vol. 16, Pt. 2, American Astronautical Society, Springfield, VA, 1963, pp. 133–135.

[7]"National Space Transportation System Reference, Vol. 1: Systems and Facilities," NASA TM-101877, June 1988, pp. 588–591, 681–791.

[8]"Space Shuttle Mission STS-86, Shuttle MIR Mission-07 (S/MM-07): Press Kit," NASA, Sept. 1997.

[9]"MIR 1 SPACE STATION: A Technical Overview," edited by G. A. Girian, NPO Energia, Alexandria, VA, 1995, p. 48.

[10]Legostayev, V. P., and Raushenbakh, B. V., "Automatic Rendezvous in Space," Foreign Technology Div., FTD-HT-23-21346-68, Dec. 1968.

[11]Lenorovitz, J. M., "Soviets Upgrading Space Transports to Support Manned Station Activities," *Aviation Week and Space Technology*, Oct. 23, 1989, p. 49.

[12]Kidger, N., "MIR MISSION REPORT: Progress M-7 Catastrophe Avoided," *Spaceflight*, Vol. 33, British Interplanetary Society, London, 1991, pp. 192, 193.

[13]Dickey, B., "Crash in Space Leaves Mir's Future Uncertain," *Final Frontier*, ATG Media, Hollywood, CA, Sept./Oct. 1997, pp. 10, 11.

[14]Littleton, F. C., "A Description of Hardware and Mission Planning for the Apollo–Soyuz Test Project," *Proceedings of the Twelfth Space Congress* (Cocoa Beach, FL), Canaveral Council of Technical Societies, Cocoa Beach, FL, 1975, pp. 4-7-4-14.

[15]Gonzalez, R., and Culp, R. D., "Investigation of an Rbar/Cusp Approach for Russian Rendezvous to Space Station Alpha," American Astronautical Society, AAS Paper 95-123, Feb. 1995, pp. 979–998.

[16]Wilson, A. (ed.), *Jane's Space Directory, Twelfth Edition, 1996–1997*, Jane's Information Group, Inc., Alexandria, VA, 1996, pp. 36, 121–124.

[17]"NASA Automated Rendezvous and Capture Review," NASA TM-108598, Nov. 1991.

[18]"Automated Rendezvous and Capture Demonstration Study Final Report," NASA CR-192587, June 1993.

[19]Polites, M. E., and Bartlow, B. E., "United States Control Module Guidance, Navigation, and Control Subsystem Design Concept," NASA TP-3677, March 1997.

[20]Scofield, W. T., and Mellin, J. R., "A Feasibility Study of Unmanned Rendezvous and Docking in Mars Orbit," Martin Marietta, MCR 74-244, Denver, CO, Sept. 1974.

[21]Vandenberg, F. A., "Autonomous Rendezvous and Docking in Mars Orbit," American Astronautical Society, AAS Paper 75-070, July 1975.

[22]Howard, R. T., and Book, M. L., "Video Guidance Sensor for Automated Capture," AIAA Paper 92-1389, March 1992.

[23]Locke, J. W., and Casey, L. D., "Rendezvous Radar for Orbital Vehicles," AIAA Paper 92-1689, March 1992.

[24]Backes, P. G., "Supervised Autonomous Control, Shared Control, and Teleoperation for Space Servicing," *Proceedings of the Sixth Annual Workshop on Space Operations, Applications, and Research* (Houston, TX), Vol. 2, NASA Headquarters, Washington, DC, 1992, pp. 720–731.

[25]Landzettel, K., Brunner, B., and Hirzinger, G., "The Telerobotic Concepts for Experimental Servicing Satellite (ESS)," IARP Workshop on Space Robotics (Montreal, PQ, Canada), July 1994.

[26]Scott, W. B., "U.S. Defines Missions for Military Spaceplane," *Aviation Week and Space Technology*, Jan. 13, 1997, pp. 362, 363.

[27]Scott, W. B., "USAF Set to Fly Mini-Spaceplane," *Aviation Week and Space Technology*, Aug. 4, 1997, pp. 20, 21.

[28]Mankins, J. C., "The Space Solar Power Option," *Aerospace America*, May 1997, pp. 30–36.

[29]Upadhyay, T. N., Cotterill, S., and Deaton, A. W., "Autonomous Reconfigurable GPS/INS Navigation and Pointing System for Rendezvous and Docking," AIAA Paper 92-1390, March 1992.

[30]"User's Manual for the Trajectory Control Sensor Command and Display User Interface, 6/5/97, Revision 5.8," NASA Johnson Space Center, JSC 27870, Houston, TX, June 1997, p. 4.

[31]Williams, T., and Tanygin, S., "On-Orbit Engineering Tests of the AERCam Sprint Robotic Camera Vehicle," American Astronautical Society, AAS Paper 98-171, Feb. 1998.

[32]"DTO 700-11: Orbiter Space Vision System Flight Testing," Flight Test and Supplementary Objectives Doc., Rev. U, NASA Johnson Space Center, NSTS 16725, Houston, TX, Feb. 1998.

[33]Gelderloos, H. C., Sheikh, S. I., and Schipper, B. W., "GPS Attitude System Integrated with an Inertial System for Space Applications," *Proceedings of the AIAA/IEEE 16th Digital Avionics Systems Conference*, Vol. 2, AIAA, Reston, VA, 1997, pp. 8.1-1–8.1-10.

[34]Bunn, R. L., "GPS/INS Autonomous Navigation Systems for Space Applications," American Astronautical Society, AAS Paper 98-016, Feb. 1998.

[35]Lau, K., Lichten, S., Young, L., and Haines, B., "An Innovative Deep Space Application of GPS Technology for Formation Flying Spacecraft," AIAA Paper 96-3819, July 1996.

[36]Lichten, S., Purcell, G., Wu, S., Kuang, D., Srinivasan, J., Young, L., and Lau, K., "Autonomous Formation Flyer (AFF) Sensor Technology Development," American Astronautical Society, AAS Paper 98-062, Feb. 1998.

[37]Fehse, W., Tobias, A., Champetier, C., Pairot, J. M., and Pauvert, C., "A Guidance, Navigation and Control System for Rendezvous and Proximity Operations Between Hermes, The Columbus Free Flyer and The Space Station Freedom," American Astronautical Society, AAS Paper 91-063, Feb. 1991, pp. 385–409.

[38]Cirillo, W. M., Saucillo, R. J., Hendershot, J. E., and Green, J. L., "Design Reference Mission Specifications for the ESA Automated Transfer Vehicle," International Astronautical Federation, IAF Paper 93-V.2.609, Oct. 1993.

[39]Pairot, J. M., Frezet, M., Tailhades, J., Fehse, W., Tobias, A., and Getzschmann, A., "European Rendezvous and Docking System," International Astronautical Federation, IAF Paper 91-328, Oct. 1991.

[40]De Selding, P. B., "Negotiations Place ATV Back on Track," *Space News*, Vol. 9, No. 14, 1998, pp. 1, 34.

[41]Ress, R., and Puech, P., "The Automated Transfer Vehicle ATV-Central Element of Europe's New Manned Space Transportation Programme," International Astronautical Federation, IAF Paper 94-V.2.531, Oct. 1993.

[42]Clar, P. G., Theillier, F., and Triou, H., "Analysis of Control Accuracy in the Final Phase of the Rendezvous Between the Automated Transfer

Vehicle and the Space Station Freedom," American Astronautical Society, AAS Paper 93-342, 1993, pp. 1295–1306.

[43]Cislaghi, M., Lellouch, M., and Pairot, J. M., "The ATV Rendezvous Pre-development (ARP) Project," *ESA Bulletin*, Vol. 39, Feb. 1997, pp. 60–67.

[44]Sekigawa, E., and Mecham, M., "NASDA Plans November Launch for ETS-7, TRMM," *Aviation Week and Space Technology*, April 14, 1997, pp. 61, 62.

[45]Akimoto, T., Tetsuichi, I., and Miyaba, H., "Concept of H-II Orbiting Plane, HOPE," *Proceedings of the Sixteenth International Symposium on Space Technology and Science, Volume II*, 1988, pp. 1501–1506.

[46]Yamanaka, K., "Rendezvous Strategy of the Japanese Logistics Support Vehicle to the International Space Station," *Proceedings of the 3rd ESA International Conference on Spacecraft Guidance, Navigation and Control Systems*, ESA SP-381, European Space Agency, Noordwijk, The Netherlands, 1997, pp. 103–108.

[47]Sekigawa, E., "Japan Forced to Cut HOPE, Other Programs," *Aviation Week and Space Technology*, Aug. 4, 1997, pp. 28, 29.

[48]Takagi, T., Tsuya, N., and Obara, H., "Orbital Servicing Vehicle," *Proceedings of the Sixteenth International Symposium on Space Technology and Science*, Vol. 2, 1988, pp. 1591–1596.

[49]*Aviation Week and Space Technology*, Dec. 15, 1997, p. 19.

[50]Anselmo, J. C., "Moving," *Aviation Week and Space Technology*, March 9, 1998, p. 17.

[51]Proctor, P. (ed.), "Arm Wrestling," *Aviation Week and Space Technology*, March 30, 1998, p. 13.

[52]Kallendar, P., "ETS-7 Docking Marks Triumph for Troubled Japanese Satellite," *Space News*, July 13–19, 1998, p. 32.

[53]*Aviation Week and Space Technology*, Aug. 17, 1998, p. 25.

[54]Tietz, J. C., and Kelly, J. H., "Development of an Autonomous Video Rendezvous and Docking System," Martin Marietta, MCR-82-569, Denver, CO, June 1982, p. III-3.

[55]Pollmeier, V. M., and Kallemeyn, P. H., "Galileo Orbit Determination from Launch Through the First Earth Flyby," *The Institute of Navigation: Proceedings of the Forty-Seventh Annual Meeting* (Williamsburg, VA), Inst. of Navigation, Washington, DC, 1991, pp. 9–16.

[56]Polites, M. E., "An Assessment of the Technology of Automated Rendezvous and Capture in Space," NASA TP-1998-208528, July 1998.

J. A. Martin
Associate Editor