NASA Technical Memorandum 102163

Pathfinder Autonomous Rendezvous and Docking Project Annual Report

Fiscal 1989

Engineering Directorate
Navigation, Control, and Aeronautics Division
Lyndon B. Johnson Space Center

National Aeronautics and Space Administration
Lyndon B. Johnson Space Center
Houston, Texas

August 1990

Contents

PRECEDING PAGE BLANK NOT FILMED

Contents (Continued)

Contents (Concluded)

List of Tables

List of Figures

Acronyms and Abbreviations

AR&D	Autonomous Rendezvous and Docking
ASTP	Apollo Soyuz Test Project
BVL	Battin Vaughan Lambert
CG	Center of Gravity
CSDL	Charles Stark Draper Laboratories
CSM	Command Service Module
CW	Clohessey Wiltshire
dps	degrees per second
DM	Docking Module
DOF	Degree of Freedom
DRM	Design Reference Mission
ERV	Earth Return Vehicle
EVA	Extravehicular Activity
FMEA	Failure Modes and Effect Analysis
fps	feet per second
FTS	Flight Telerobotic Servicer
GEO	Geosynchronous Earth Orbit
GN&C	Guidance, Navigation and Control
GPS	Global Positioning Satellite
GSE	Ground Support Equipment
HST	Hubble Space Telescope
IDM	International Docking Mechanism
JSC	Johnson Space Center
JPL	Jet Propulsion Laboratory
LDS	Laser Docking Sensor
LEO	Low Earth Orbit
LESC	Lockheed Engineering & Sciences Company
LM	Lunar Module
LOSF	Lunar Orbit Support Facility
LVLH	Local Vertical Local Horizontal
MCO	Mapping/Communication Orbiter
MAV	Mars Ascent Vehicle
MIPS	Million Instructions Per Second
MLM	Manned Lunar Module
MRSR	Mars Rover Sample Return
MSFC	Marshall Space Flight Center
NSTS	Nation Space Transportation System
OAST	Office of Aeronautics and Space Technology
OMV	Orbital Maneuvering Vehicle
ONCC	Optimized Nominal Corrective Combination
ORU	Orbital Replaceable Unit
OSC	Operations Support Center
OTV	Orbital Transfer Vehicle
POS	Proximity Operations Sensor
PROX OPS	Proximity Operations
R-bar	Radius Vector
RCS	Reaction Control System

Acronyms and Abbreviations (concluded)

RDM	Rendezvous and Docking Module
RGDM	RMS Grapple Docking Mechanism
RISC	Reduced Instruction Set Computing
RMS	Remote Manipulator System
SCA	Sample Canister Assembly
SE&I	Systems Engineering and Integration
SOR	Stable Orbit Rendezvous
SRC	Sample Return Canister
SRO	Sample Return Orbiter
SSF	Space Station Freedom
SSFP	Space Station Freedom Program
SSS	Satellite Targeting Algorithms for Rendezvous
SPRINT	Simulation Program for Rendezvous Integrating Navigation and Targeting
TN	Transportation Node
TPDM	Three-point Docking Mechanism
V-bar	Velocity Vector
WBS	Work Breakdown Structure
WP	Work Package

SECTION 1
INTRODUCTION

1.1 PATHFINDER AUTONOMOUS RENDEZVOUS AND DOCKING (AR&D) ELEMENT

The AR&D Project will develop and demonstrate capabilities to support manned and unmanned vehicle operations in lunar and planetary orbits. In this initial phase of the project, primary emphasis was placed on definition of the system requirements for candidate Pathfinder mission applications and correlation of these system-level requirements with specific technology requirements.

A near-term ground demonstration of AR&D capabilities is planned, incorporating existing or emerging technologies to verify proof of concept. A far-term demonstration will be developed to provide proof of concept for advanced sensors. The corresponding guidance, navigation, and control (GN&C) algorithms and trajectory control techniques will be developed and the integrated system will be tested on the ground.

1.2 PATHFINDER

The Pathfinder Program is a NASA initiative to develop capabilities critical to the future of the civil space program. Through Pathfinder, the Office of Aeronautics and Space Technology (OAST) will develop a variety of high-leverage technologies that enable a wide range of potential future missions. Pathfinder is organized into four principal areas:

(1) Exploration
(2) Operations [Includes AR&D]
(3) Humans-in-Space
(4) Transfer Vehicles.

1.3 TECHNOLOGY ASSESSMENT

AR&D implementation requires development of several critical technologies: (1) sensors for long and short ranges; (2) guidance, navigation, and control (GN&C) algorithms and trajectory control techniques; and (3) docking mechanisms suited to autonomous operation.

1.4 MANAGEMENT STRUCTURE

The Pathfinder AR&D project is structured as depicted in figure 1-1.

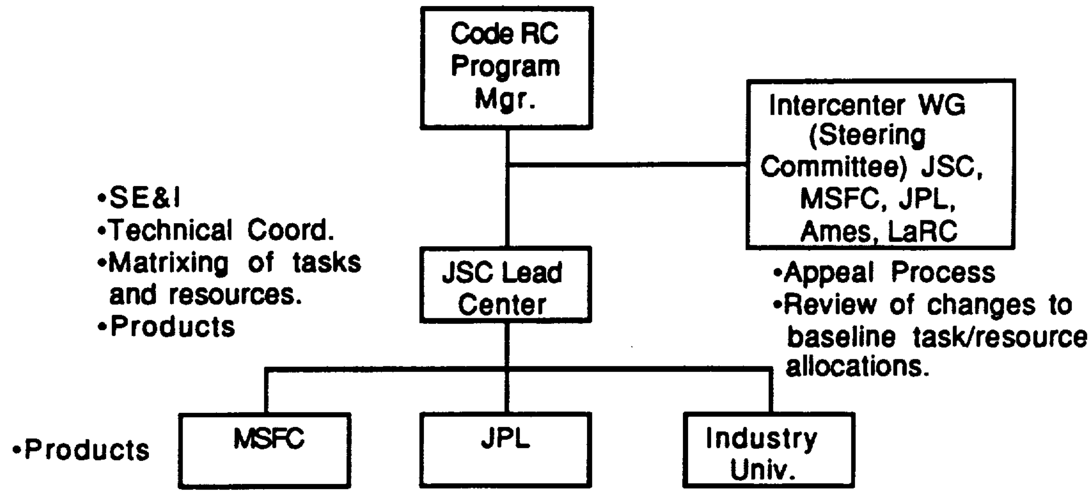


Figure 1-1.- Management Structure for Pathfinder AR&D Project.

1.5 WORK BREAKDOWN STRUCTURE (WBS)

The Autonomous Rendezvous and Docking Project WBS is shown in figure 1-2.

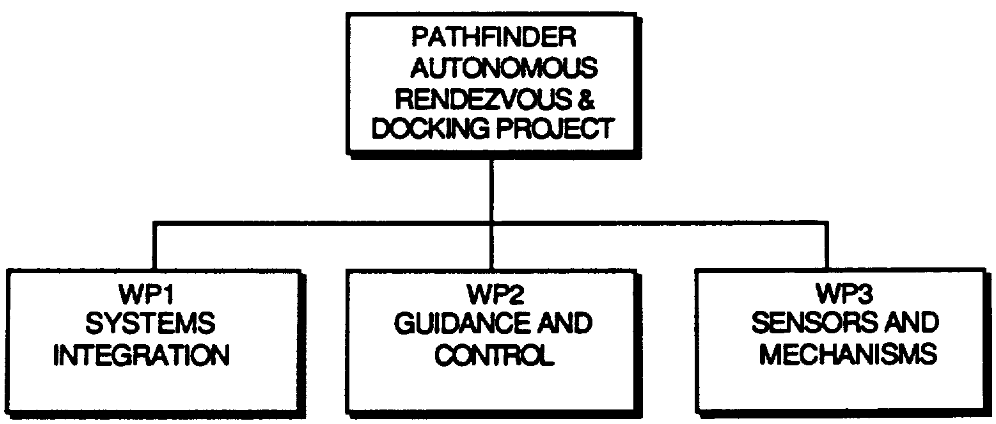


Figure 1-2.- Pathfinder AR&D Work Breakdown Structure (WBS).

1.6 SYSTEMS INTEGRATION

The objectives of work package one (WP1) are performance of several tasks including (1) program planning support, (2) systems and mission analyses, (3) trajectory control analyses, and (4) GN&C system integration.

Systems integration provides program planning and control support, systems engineering and integration of hardware and software technology developments, and continuing coordination of the program element's activities with Pathfinder Program directions. The AR&D Project involves the development of a number of technologies, with various options within each technology area. A focused systems integration effort will establish appropriate priorities, budgets, and systems-level assessments and direction to provide timely and cost-effective fulfillment of objectives.

Systems-level studies are being performed to provide direction and focus to the detailed technology development studies. Various options to meet the technology requirements will be assessed to select favorable candidates. The capabilities for near-term and far-term demonstrations will be defined based on the mission requirements and projections of technology development schedules.

AR&D technology components will be integrated in high-fidelity GN&C simulations to ensure that the individual technologies result in a viable and effective system design. Simulation will also be used to compare technology options.

When the AR&D technologies mature to prototype development, WP1 will include establishment of the requirements for proof of concept demonstrations. For ground demonstrations, the test plans and facility usage plans will be developed and coordinated. For flight demonstrations, sponsoring organizations will be sought and support for the manifesting process provided.

1.7 GUIDANCE, NAVIGATION, & CONTROL (GN&C)

The purpose of work package two (WP2) is to support the identification, development, and assessment of GN&C technologies, algorithms, and techniques applicable to AR&D. It is envisioned that autonomous operations will be employed as a part of unmanned missions, precursor missions, and manned explorations of the Moon and Mars.

Areas of interest include

- Optimization of rendezvous trajectories
- Cooperative control methodologies
- Trajectory control requirements, techniques, and boundary conditions
- Mass properties identification and compensation
- Guidance and control aspects of maneuver sequence abort criteria definition and implementation of abort sequences
- Detection and avoidance of hazardous conditions
- GN&C system failure detection, fault tolerance, redundancy requirements, and techniques for automatic system reconfiguration
- Application of expert systems technology to GN&C tasks
- Development of general purpose rendezvous guidance schemes

The basic approach to GN&C technology development is to proceed from a point design for a GN&C system to one optimized for a candidate mission and vehicle. The first step in this process is to identify integrated GN&C functional requirements. Proposed AR&D GN&C technologies will then be evaluated in terms of compliance with these requirements.

The assessment and evaluation of the candidate GN&C algorithms will rely heavily on simulations of vehicle operations in Earth, lunar, and planetary orbits. The AR&D simulations for GN&C will, whenever possible, build upon existing simulations and incorporate models of vehicles, sensors, and algorithms.

1.8 SENSORS AND MECHANISMS

Work package three (WP3) will provide sensors and mechanisms that satisfy the requirements of AR&D in Earth, lunar, and Mars orbits. Generic requirements for all devices include low power consumption, weight, and form factor; reliable operations in varying and hostile environments; and operation in these environments after protracted periods of dormancy. When practical, existing devices and techniques for fabrication and operation of these devices will be adopted.

Sensor development begins with definition of the tracking requirements for AR&D. These requirements are derived from a Design Reference Mission (DRM) for the specific program to be supported and from the trajectory control requirements specified by WP1. Final sensor requirements are a product of the interaction between all of the AR&D elements, such as the GN&C system, chase and target vehicle characteristics, and the mechanisms employed.

An early goal of WP3 is to perform and analyze the results of a trade study to identify those sensor technology development areas that are cost effective and to improve performance and reduce development and schedule risks. These technologies include optical and radio frequency tracking and active versus passive detection. The technologies for laser and radio frequency tracking sensors are sufficiently mature that they are candidates for a near-term demonstration of AR&D.

Reduction in system size and weight and in total power requirements, coupled with prolonged exposure to hostile environments, are problems to be attacked in developing the mechanisms. To meet these challenges, the following course was adopted:

- Identification and definition of requirements for high reliability, light weight, low power docking components such as latches and load attenuators
- Investigation of basic technologies in the areas of lubrication and seal materials and techniques for long-term exposure to hostile environments
- Assessment of mechanism requirements for anticipated Lunar and planetary auto-docking systems
- Performance of trade studies to identify preferred approaches to mechanism development

As is the case with sensor development, priority is being placed on selecting a single approach early to avoid dilution of resources among many potential avenues of exploration.

SECTION 2
SUMMARY OF FISCAL YEAR 1989 ACTIVITIES

Fiscal Year (FY) 1989 AR&D activities are best characterized as "foundation-building" within each of the work packages. The lion's share of effort has been dedicated to assessing the current state of the art, identifying desired elaborations and expansions to this state of the art, and charting a course that will realize the desired objectives in the future. A major effort in FY '89 across all work packages was in developing tools and facilities that will be used to test, refine, and validate basic AR&D elements, both in terms of hardware and software.

2.1 WORK PACKAGE ONE (WP1)

In WP1 (Systems Integration), a significant technical accomplishment was the production of a System Requirements Document. (See Section 3.) This document defines component parts and supporting elements of the AR&D system and establishes performance requirements for each of these. It is important to note that in the context of the document, "element" is not restricted to hardware, but rather can be interpreted as being synonymous with "function" or "functional area." In the course of deriving the requirements documented in this work, developments in arenas such as the Satellite Servicer System and the Mars Rover Sample Return mission were tracked closely. TRW and Lockheed Engineering and Sciences Company (LESC) had primary responsibilities in support of WP1 goals.

An ongoing function of WP1 was project planning and control in the face of shifting funding in the current year as well as in forecasts for future years. During FY '89 a set of comprehensive program and project plans were created and integrated as appropriate within the overall framework of the Pathfinder Program. Additionally, a number of status reports and other interim products were generated and a reporting structure was established.

As an investment in future systems engineering and integration support capabilities, two steps were taken in FY '89. Acquisition of a reduced instruction set computing-based (RISC) workstation to host dedicated systems engineering and analysis application software was initiated. A microcomputer-based data base (See Section 4) structure was also developed to facilitate the storage and accessing of information pertaining to system requirements and capabilities.

2.2 WORK PACKAGE TWO (WP2)

Efforts associated with WP2 (Guidance and Control) were oriented towards identification of improved guidance and targeting algorithms and concurrent development of analytical tools with which to evaluate their performance. For example, the Battin-Vaughan-Lambert (BVL) rendezvous algorithm has been proposed to supplant the more conventional Clohessey-Wilshire (CW) formulation. A significant effort has gone into developing the algorithm, quantifying its performance benefits, and developing the 6 and 12 degrees of freedom (DOF) simulations that will be used in evaluating it.

Development of algorithms controlling automatic proximity operations, docking, and concepts for multi-vehicle cooperative control began in FY '89. Collaterally, a detailed analysis of requirements governing the inclusion of artificial intelligence, expert systems, and related technologies was initiated. With respect to verification and evaluation tools and resources, the requirement for a significant capital expenditure has been avoided through development of plans to use existing resources, such as flat floor facilities, with relatively minor modifications to support AR&D development.

The objective of the BVL algorithm investigation is to develop guidance techniques that make it possible to execute, autonomously, highly fuel-efficient rendezvous in planetary orbit, on return to Earth, and between low Earth orbit (LEO) and geosynchronous orbit (GEO). Such rendezvous maneuvers involve transfers between circular, elliptical, parabolic, and hyperbolic trajectories. The guidance algorithm will use the BVL algorithm first published in 1984.

In FY '89, some basic WP2 components were completed, including a group of 10 general-purpose Ada packages (commissioned by the Space Station Freedom Software Support Environment and released in March) and an astrodynamics package (to be released in October). A new universal Kepler algorithm suitable for flight computers was developed using equations published in Battin's text and is included, with the Lambert algorithm, in the astrodynamics package. In FY '90, the guidance algorithm and a rendezvous simulation will be completed, and a report will be issued that compares the performance of the new BVL guidance to traditional guidance for Mars ascent and assesses the performance of BVL guidance for return to Earth and for LEO to GEO rendezvous.

An extensive review was performed of past rendezvous material from the Charles Stark Draper Laboratory (CSDL). This material was gathered from CSDL's involvement in Apollo as prime contractor to their current support of the Space Shuttle Program at the Johnson Space Center (JSC). This involvement includes Skylab and Apollo-Soyuz rendezvous analysis and development of proximity operations with the Space Station. A top level summary of integrated GN&C system requirements is being prepared from this extensive experience base and will be presented at JSC in early FY '90.

The collateral effort associated with WP2 involved participation in the development of a dual spacecraft 6-DOF simulation derived from the CSDL On-orbit Functional Simulator, with CSDL personnel involved in the integrated GN&C support task under the direction of the JSC Avionics Systems Division. This dual vehicle 6-DOF simulation is in the final stages of testing and will be used for high fidelity integrated GN&C rendezvous and proximity operations analysis. The dual vehicle simulation configuration allows flexibility in the types of vehicles to be used in the analysis. Current work involves the implementation of the Orbital Maneuvering Vehicle (OMV) GN&C software and vehicle models.

A three-axis rendezvous maneuver automated rendezvous targeting algorithm was developed by JSC that does not constrain the time of flight as do the traditional fixed transfer time three-axis rendezvous targeting methods. The formulation is based on the simultaneous solution of Kepler's equations for both the target and chase vehicle. The algorithm iterates on a target transfer angle that satisfies both time of flight equations, while controlling the differential altitude and phase angle between the two vehicles. Out-of-plane control is achieved by appropriately directing the required new chase vehicle velocity vector in three-dimensional space. The absence of the time of fight constraint requires the placement of a different constraint on the problem. While there are several possibilities, the terminal radial velocity of the chase vehicle was chosen for this study. The control of the terminal velocity constraint is expected to provide for better trajectory control than conventional Lambert targeting. In addition, the liberation of the time of flight enables the optimization of each maneuver, i.e. minimization of radial delta-velocity components. In this document this targeting method is referred to as Optimized Nominal Corrective Combination (ONCC) maneuver. (See Section 5.)

The ONCC algorithm was tested using Monte Carlo simulation. The stable orbit rendezvous (SOR) profile and Space Station Freedom rendezvous reference profile were successfully flown using ONCC targeting. For these profiles the reduction in delta-velocity requirements compared to the baseline was not significant, as these profiles are not designed to take advantage of the ONCC maneuver capabilities. However, a different profile was designed as a candidate for automated rendezvous applications suitable for ONCC targeting.

It was observed that if either two consecutive 240 degree or two 120 degree maneuvers are performed, the altitude of the second maneuver lies exactly half-way between the altitude of the first maneuver and the terminal altitude of the second maneuver, regardless of the actual altitudes. Moreover, if the first maneuver is executed at a relative apogee or perigee, the terminal point of the second maneuver is also at a relative apogee or perigee. By doing so, all maneuvers are nominally horizontal. These characteristics suggest the use of these sequences for automated rendezvous as the differential altitudes of the maneuver points can be algorithmically correlated. In addition, 120 degree or 240 degree transfer maneuvers provide for good phasing, altitude, and out-of-plane control and dispersion capability.

This observation, coupled with the development of the ONCC targeting method, enabled the formulation of an algorithm that allows determination of the target points (or I-Loads) for the impending rendezvous during the actual flight based on the latest measurements of the current position of the chase vehicle with respect to the target.

The proposed profile consists of two consecutive nominal 240 degree and two sets of two 120 degree transfer angles, requiring six primary three-axes maneuvers plus the velocity null at the V-bar line, as well as an optional midcourse correction prior to it. The transfer angle of the sixth maneuver may be altered to achieve a non-zero terminal radial velocity at the velocity null. This profile was flown using Monte Carlo simulations with both ONCC targeting and conventional Lambert targeting using the standard error ellipses and navigation. The nominal transfer angle of the sixth maneuver was set to 100 degree to achieve a 1.6 fps radial velocity component at the intercept point. The initial altitude was 21.6 nm (40km). The preliminary indication is that the proposed profile and ONCC targeting, because of its efficiency and flexibility, are well suited for automated rendezvous.

Participants in WP2 included the Jet Propulsion Laboratory (JPL), CSDL, Marshall Space Flight Center (MSFC), and JSC's Avionics Systems Division.

2.3 WORK PACKAGE THREE (WP3)

WP3 (Sensors and Mechanisms) is crucial to AR&D in that it encompasses development and acquisition of the long lead hardware items which will shape the basic performance envelope of the final implementation and will pace development of the entire project. Two key tasks have been initiated in WP3. A trade study of approaches to and requirements on sensors began in FY '89 and substantial progress has been made. The aim of this study is to characterize the types and numbers of sensors that will be required to support typical AR&D mission scenarios. Also a draft report on the status of basic mechanism research has been prepared. (See Section 7.) In this report, the basic approaches to docking, capture, and impact attenuation requirements have been identified, as is the interrelationship between attributes of the selected mechanisms and the docking algorithms. Miniaturization of existing systems is a major challenge to mechanism designers for AR&D.

An investigation was made into the suitability of a video-based sensor system comprising laser-diode illumination sources and a charge injection device imaging sensor mounted to the chase vehicle and a triad of retroreflectors on the target. (See Section 8.) Sophisticated image fusion and recognition algorithms would serve to couple the sensor data to the GN&C processing loop.

Contributors to these efforts for FY '89 were JSC's Structures and Mechanics Division, MSFC, Ames Research Center, and JSC's Tracking and Communications Division.

SECTION 3
AR&D SYSTEM REQUIREMENTS
(Prepared by TRW-Houston)

3.1 INTRODUCTION

3.1.1 Purpose

The purpose of this section of the report is to define the component parts of the AR&D System and to establish the requirements imposed upon the system and its component parts in order to meet the stated goals of the AR&D Project.

3.1.2 Scope

Currently, the AR&D Project is focusing on two general mission scenarios that require AR&D capability: the unmanned Mars Rover and Sample Return (MRSR) and the Space Station accommodation of a Lunar Base. These mission scenarios are described in TRW Report 89:W480.1-67, "Mission Scenario Assessment-Autonomous Rendezvous and Docking," 6 April 1989.

3.1.3 Autonomy Requirements

Autonomy is defined as the automatic operation of the vehicles without ground-based support and without manned intervention. All developments under the AR&D Project will be directed toward autonomous operations.

3.2 DESCRIPTION OF THE AR&D SYSTEM

3.2.1 System Architecture

Chase and target vehicles will be used as part of the AR&D Project. These vehicles will possess guidance, navigation, control, communications, tracking, propulsive, control effecter, and data processing/data management capabilities to support basic operations independent of the rendezvous and docking operations. The AR&D System will make maximum use of these existing capabilities and include only those elements necessary to implement autonomous rendezvous and docking capabilities. That is, the AR&D System is planned to be a complement to the basic avionics systems architecture of the chase and target vehicles.

The AR&D System is a functional assemblage of elements or components employed in the rendezvous and docking of spacecraft, which have a significant impact on the autonomy of the operation. The physical characteristics of the components will vary with mission and vehicle design. The requirements specified in this document may, therefore, be generally applicable to AR&D operations or they may conditionally apply to one or more specific applications.

3.2.2 Mission Characteristics

The AR&D System is an enabling technology for unmanned lunar and interplanetary space flights for which ground-based navigation and remote piloting are not adequate to achieve docking (i.e., the MRSR mission). The AR&D System is an enhancement technology in a number of Earth-orbital and manned-vehicle missions (i.e., the Space Station accommodation of a lunar base) as a result of the additional equipment and performance capabilities that it provides.

Rendezvous will be initialized from trajectories resulting from ascents, interplanetary transfers or elliptical orbits of any eccentricity.

3.2.3 AR&D System Elements

At this time, a number of options for AR&D System elements are being pursued in parallel. These options include various sensor types; guidance and control techniques; artificial intelligence or neural network implementation of the guidance, navigation and control techniques; and redundancy management techniques. A major effort will be required to coordinate selected options into an integrated system which meets the requirements of the proposed missions. The options chosen may vary with the specific mission.

3.2.3.1 Supporting Elements

Vehicles, systems and subsystems provided for the general performance of a mission are not included in the AR&D System elements if their design has no impact on AR&D performance, their design meets the AR&D requirements, or their design is unalterable to accommodate AR&D requirements. Systems supporting rendezvous and docking operations but not subject to these requirements may impose conditional requirements on the AR&D System which are specific to the mission or vehicle characteristics.

No unique vehicle design requirements are proposed. As noted previously, the supporting elements of the AR&D System include the guidance, navigation and control, communications, tracking, propulsive, control effecter, and data processing/data management capabilities.

3.2.3.2 Sensors

The sensor element comprises sensors that provide data on relative motion or relative orientation. Inertial measurement sensors are part of the chase vehicle and will not be considered a component of the sensor element unless specifically designed to support AR&D. Rendezvous radar, visual image trackers and cooperative target enhancements may be included in the AR&D sensor element.

3.2.3.3 Navigation

The AR&D navigation element is composed of computer software which maintains an estimate of the relative states of the chase and target vehicles. Navigated states include relative position, relative velocity, bearing, relative vehicle orientation and vehicle rates. Range, range rate and derived states providing look-angles and line-of-sight rates are provided to guidance and control users as required. Data required for the navigated states will be provided by the sensor element, inertial measurements from supporting systems, and from external sources through the communications and tracking element.

3.2.3.4 Guidance and Control

The guidance and control element consists of the computer software required to perform thrusting maneuvers at appropriate times during the rendezvous and docking operation. Targeting of future position and velocity is controlled by the maneuver times and burn durations based on navigated states.

Attitude control provides thrust vectoring, sensor visibility and docking orientation. It also provides the damping of docking transients and stabilization of the docked configuration of the chase and target vehicles. Gyros, rate gyros, thrust effecters and controllers are supporting systems of the chase vehicle.

3.2.3.5 Docking Mechanism

The docking mechanism element of a spacecraft comprises the structure and mechanical systems necessary to safely and reliably effect mating with another vehicle. The design of a docking mechanism is dependent on the characteristics of the vehicles involved, including mass properties, geometry, available sensors, guidance, and navigation and control parameters. These parameters vary from vehicle to vehicle, resulting in different limits for the expected contact conditions between the vehicles (i.e., closing rate, lateral velocity, relative attitude and attitude rate). These contact conditions are the primary driver in the design of a docking mechanism. The mechanism must be of sufficient strength and stiffness to handle the mass properties of the vehicles involved, for the expected contact conditions. Mechanical systems are to be provided on the docking mechanism for initial capture of the interfaces, attenuation of the relative motion and loads, and establishment of a rigid interface between the vehicles, which may be pressurized if necessary.

Different vehicles, programs, and mission scenarios, will place different requirements on the docking mechanism needed to safely and reliably accomplish docking and other mission requirements. Therefore, a docking mechanism with unique characteristics will be needed for different AR&D applications. As a result, the AR&D Project does not include development of a specific docking mechanism, but does include development of criteria and standards for design of a docking mechanism for AR&D applications. These criteria and standards will be based upon the requirements and characteristics of existing and planned docking mechanisms which are applicable to the proposed AR&D mission scenarios. In effect, the docking mechanisms for AR&D will be considered as a supporting element. The specific program applying AR&D technology will have the responsibility of verifying AR&D systems and docking mechanism compatibility to meet AR&D and program-specific docking requirements.

3.2.3.6 Communications and Tracking

The communications and tracking element consists of the equipment and software necessary to control the sensors used in tracking the target vehicle and to provide a data link between vehicles for transmitting navigated states from a cooperative target vehicle if available for AR&D.

3.2.3.7 Data Processing and Data Management

In general, the data processing, data management, and data bus capabilities are supporting elements of the chase and target vehicles. However, AR&D requirements may exceed the capabilities of the chase vehicle; in which case, the additional capabilities will be part of the AR&D system. Examples of areas where additional or unique capabilities may be required include implementation of artificial intelligence, neural networks, or computer-aided vision.

3.3 SYSTEM REQUIREMENTS

This section addresses the system requirements associated with the AR&D capability. These requirements will, in general, be highly dependent on the specific mission and associated supporting elements. It is expected that the AR&D Project will not be able to dictate control effecter designs (e.g., propulsion and reaction control system) or vehicle configurations to any appreciable degree. Therefore, AR&D System requirements will generally have to encompass the basic vehicle capabilities and identify the augmentation necessary for autonomous rendezvous and docking.

AR&D System requirements will incorporate the requirements of the supporting elements (e.g., relative navigation, guidance, and navigation and control capabilities for rendezvous and proximity operations, and the docking mechanism capabilities). Indeed, tradeoffs will be required to develop a viable allocation of performance requirements among the principal supporting elements. Moreover, the allocations may vary by the specific vehicle, mission, and/or docking mechanism.

3.3.1 Critical Parameters

The following paragraphs define parameters deemed critical to the AR&D capabilities and may be design drivers. Where available, numerical requirements for these critical parameters are defined. Because of the early state of AR&D development, many of the parameters are to be determined (TBD). There is a definite goal to develop numerical values for these TBDs, and this requirements document will serve as an evolving depository for their definition.

3.3.1.1 Target Acquisition Range

The minimum target acquisition range for the rendezvous tracking sensor is TBD. The minimum target range for acquiring angular sensor data is TBD. The numerical values will be dependent upon the specific sensors. These two acquisition ranges will determine the boundaries of Zones 1 and 2, which are addressed in Tables 3-I and

3-II. Zone 1 commences with the acquisition of the target and ends with the acquisition of angular data (e.g., azimuth and elevation angles). Zone 2 spans this acquisition of the angular data to the acquisition of relative attitude data (e.g., from the docking sensors). Zone 3 then spans the acquisition of the docking sensors to actual docking/berthing.

3.3.1.2 Navigation States/Sensor Measurements

The sensor element will supply measurement data to the navigation element which will perform the data processing required to supply the guidance and control users with state vectors in the form required by the users. The accuracy requirement for the critical parameters for the navigated states and sensor measurements are tabulated in Table 3-I. The column entitled "Sensor Measurement and Accuracy" represents desired or typical performance from relative navigation sensors. The column entitled "State Variable Name and Accuracy" represents the expected navigation performance for such sensors. Table 3-I is considered to be representative. The actual numerical values will vary depending upon the specific sensor or sensor set selected for AR&D. It is expected that several versions of this table will be created as the evaluations of candidate sensors mature.

Requirements for derivative states such as line-of-sight and docking speed and for non-critical parameters will be maintained in lower level requirements documentation.

3.3.1.3 Guidance and Control

The guidance system is required to assess the inertial and relative states and the vehicle orientation and to target the velocity or attitude corrections needed to perform the rendezvous and docking operations. After converting these corrections to the active vehicle's reference coordinates, the control system is commanded to generate the torques and forces in a form required for execution by the controller.

The requirements of the guidance and control element for controlling the critical parameters appear in Table 3-II. The commanded guidance corrections must maintain the actual parameter within the "boundary" values in Table 3-II in spite of the combined errors in "control" execution and state "estimation" listed in the other columns of the same table. That is, the column entitled "Parameter Name and Boundary Value" represents the total guidance and control system performance requirement, which can be represented by the sums of the control accuracy (Column 2) and the state estimation accuracy (column 3). The values of these boundaries will be driven primarily by hardware capabilities, such as the docking/berthing mechanisms for Zone 3 and acquisition capabilities of the sensors in Zones 1 and 2.

The parameters in the column entitled "Control Accuracy by G&C" represent the ability of the guidance and control system to effect a command and do not include the state estimation accuracy. The column entitled "State Estimation Accuracy" represents the state estimation performance, which when combined with the control accuracy will meet the boundary requirements.

Again, the values cited in Table 3-II should be treated as representative. The boundary values will vary depending upon the particular mission, vehicle, and mechanisms selected. Trade-offs can be made in the corresponding allocation of performance between the control accuracy and the state estimation accuracy. It is expected that several versions of Table 3-II will be developed as the guidance and control techniques mature.

It should be noted that the state estimation accuracies in Table 3-II do not necessarily equal the second column in Table 3-I. In this requirements document, the guidance and control capability is separated from the navigation capability. Table 3-II should be viewed as describing the performance parameters associated only with the guidance and control capability. Once a guidance and control approach has been selected, Table 3-I will provide the data on navigation sensors which would support the required states estimation accuracy.

3.3.1.4 Fault Tolerance/Reliability

All equipment and subsystems required for critical functions will be effectively two-fault tolerant with at least one-fault tolerance provided through physical redundancy or internal design.

The reliability of the overall AR&D System will be in excess of .9999.

3.3.1.5 Physical Size and Weight

The additional weight added to the spacecraft for the AR&D System is to be less than 35 kilograms. The additional volume added for AR&D is not to exceed 0.05 cubic meters.

3.3.1.6 Power Requirements

The peak power consumption for tracking is not to exceed 150 watts.

3.3.2 Other System Requirements

TBD.

Table 3-I- Navigation State/Sensor Measurement Parameters and Accuracy Requirements

	State Variable Name and Accuracy	Sensor Measurement and Accuracy
ZONE 1	Target Acquisition Range: 0.01R Range rate: $(2x10^{-6})R$	Range: 0.01R Range Rate: $0.0032R^{1/3}$
ZONE 2	Rendezvous/Prox Ops (LVLH) Relative Position: Downrange: 0.005R Crossrange: 0.002R Altitude: 0.002R Relative Velocity: Downrange: 0.02 m/sec + $Rx10^{-6}$ Crossrange: 0.02 m/sec + $Rx10^{-6}$ Altitude: 0.02 m/sec	Range: 0.01R Range Rate: $0.0032R^{1/3}$ Az and El: 0.9 mrad Bearing Rate: 0.52 mrad/sec **R is range in meters**
ZONE 3	Docking (Docking ref.) Relative Position: Axial: 0.01 m + .003R Lateral: $0.03R^{1/2}$ Relative Velocity: Axial: 0.01 m /sec Lateral: $0.003R^{1/2}$ Relative Attitude: TBD Relative Angular rate: TBD	Range: 0.01R or 0.005 m Range Rate: $0.0032R^{1/3}$ or 0.003 m/sec Az and El: 3.5 $R^{-1/3}$ mrad or 0.9 mrad Az and El Rates: $1.6R^{-1/2}$ mrad/sec or 0.52 mrad/sec Attitude: 0.02 deg Angular Rate: 0.003 deg/sec

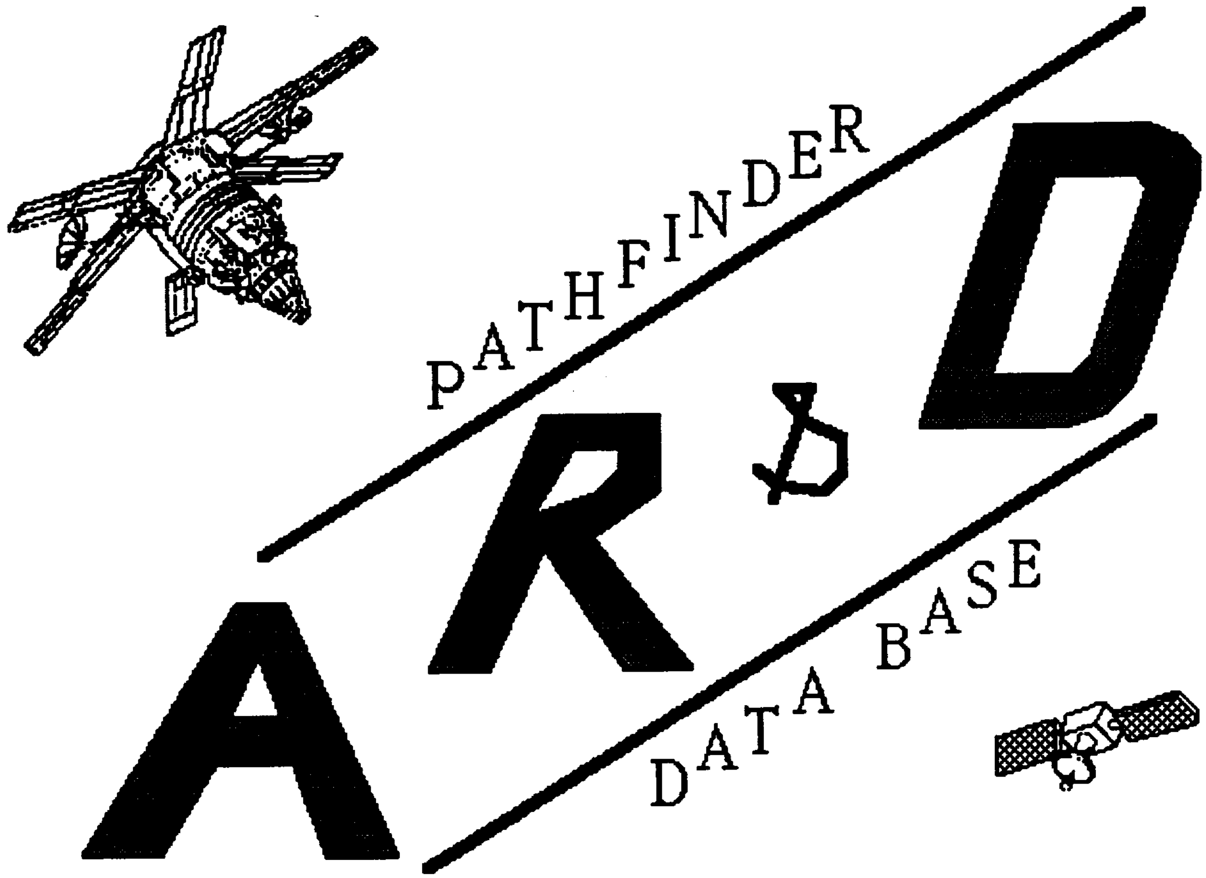
Table 3-II- Guidance and Control Requirements

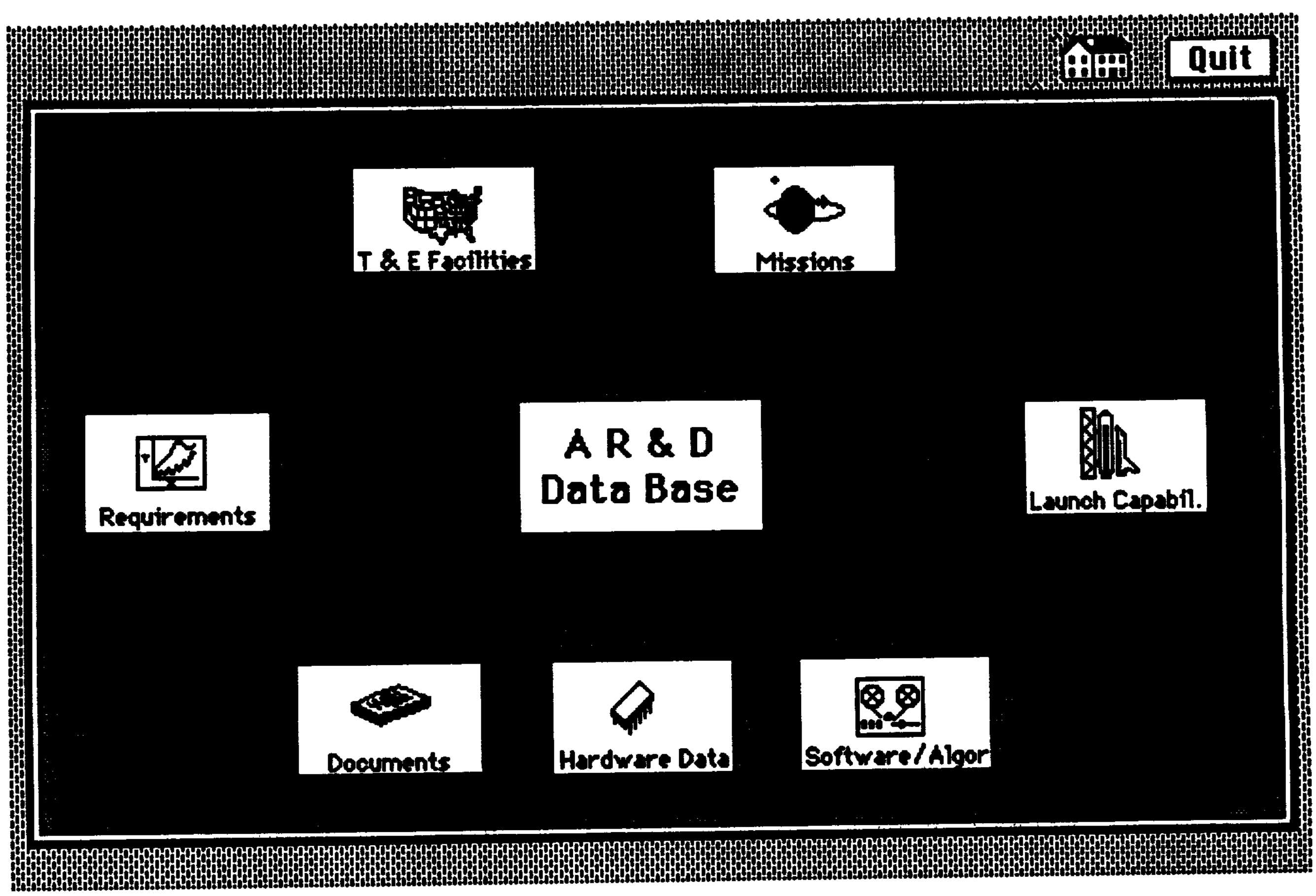
	Parameter Name and Boundary Value	Control Accuracy by G&C	State Estimation Accuracy
ZONE 1	N/A	N/A	N/A
ZONE 2	Rendezvous/Prox Ops (LVLH) Minimum Downrange Separation: TBD Maximum Crossrange: TBD	Relative Position: 3m Relative Velocity: 0.3 m/s **R is range in meters**	Relative Position: Downrange 0.005R Crossrange 0.002R Altitude 0.002R Relative Velocity: Downrange = 0.02 m/s + R x 10^{-6} Crossrange = 0.02 m/s + R x 10^{-6} Altitude = 0.02 m/sec
ZONE 3	Docking (Docking ref.) Maximum Docking Velocity: 0.015 m/sec Maximum Lateral Position: TBD Maximum Lateral Velocity: TBD Maximum Sensor Look Angle: TBD Maximum Body Rate: TBD	Closing Velocity: 0.003 m/sec Lateral Position Error: TBD Lateral Velocity Error: TBD Attitude Error: TBD Angular Rate Error: TBD	Relative Position: Axial 0.01 m + .003R Lateral $0.03R^{1/2}$ Relative Velocity: Axial 0.01 m/sec Lateral $0.003R^{1/2}$ Relative Attitude: Roll 3 deg Pitch/yaw 2 deg Relative Angular Rate: TBD

SECTION 4
AR&D REQUIREMENTS AND CAPABILITIES DATABASE
(Prepared by Lockheed Engineering & Science, Co-Houston)

4.1 INTRODUCTION

Figures 4-1 - 4-5 are representative samples of the types of data that can be stored and the query/reporting features of the data base shell structure. The dat base was developed by LESC.


PATHFINDER
A R & D
DATA BASE


Quit
T & E Facilities
Missions
Requirements
A R & D
Data Base
Launch Capabil.
Documents
Hardware Data
Software/Algor

HARDWARE REQUIREMENTS:

-List of documents-

- Pressure Sensor
- Shock Isolater Assembly

1st page | Systems | Hardware | Software
Go Back | Find...

PRESSURE SENSORS

Pressure sensors combine advanced piezoresistive sensor architecture with IC technology to offer a range of pressure sensing devices for automotive, biomedical, consumer and industrial applications.

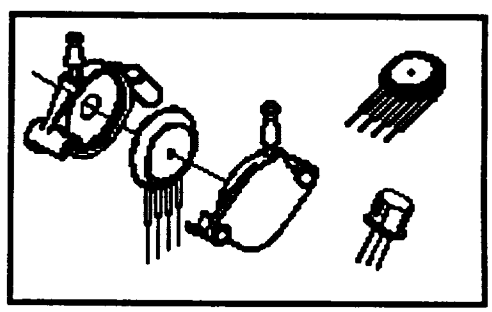
Specifications:

Pressure Ranges - 0.5 to 13, 0.8 to 16, and 0 to 45 PSI

Basic Measurements - Gage, Absolute, and Differential

Temperature Ranges - -20.0 to 100, 32 to 185, and 0 to 212 °F

Sensitivity - 3.4, 7.3, and 5.1 mV/PSI



Pressure Sensor

1st page | Systems | Hardware | Software
Go Back | Find...

DESCRIPTION: MSFC - Data Systems Test & Development Lab, System Simul.

NAME: Data Systems Test and Development laboratory, System Simulator

FECC IDENTIFIER: 4487-EB-29

ROOM NUMBER: B-246

S&E BENCH LABORATORY NUMBER: EB-42

STATUS: Operational

OVERALL CONDITION: Excellent

EQUIPMENT ACTIVATION DATE: 1982

ESTIMATED REPLACEMENT COST

Find...

DESCRIPTION: MSFC - Docking and Contact Dynamics Simulator

NAME: Docking and Contact Dynamics Simulator

FECC IDENTIFIER: 4663-EB-1

ROOM NUMBER: C-261, C-180, C-181, C-182, C-183

S&E BENCH LABORATORY NUMBER: EB-48

STATUS: Operational

OVERALL CONDITION: Excellent

EQUIPMENT ACTIVATION DATE: 1969, 1985

ESTIMATED REPLACEMENT COST OF EQUIPMENT: $2,978,000

Find...

DESCRIPTION: MSFC - Environmental Test Facility

NAME: Environmental Test Facility

FECC IDENTIFIER: 4476-ET-1

ROOM NUMBER: 102, 111, 113, 117, 119, 120, & 124

S&E BENCH LABORATORY NUMBER: ET-9

STATUS: Operational

OVERALL CONDITION: Good

EQUIPMENT ACTIVATION
DATE: 1965

ESTIMATED REPLACEMENT COST
OF EQUIPMENT: $800,000

Find...

DESCRIPTION: MSFC - Flight Simulation Laboratory

NAME: Flight Simulation Laboratory

FECC IDENTIFIER: 4487-EB-39

ROOM NUMBER: A-162, A-166, A-192, and A-191

S&E BENCH LABORATORY NUMBER: EB-45

STATUS: Being activated

OVERALL CONDITION: Good

EQUIPMENT ACTIVATION
DATE: Reactivation 1986

ESTIMATED REPLACEMENT COST
OF EQUIPMENT: $10,000,000

Find...

DESCRIPTION: MSFC - Teleoperator and Robotics Evaluation Facility

NAME: Teleoperator and Robotics Evaluation Facility

FECC IDENTIFIER: 4619-EB-1

ROOM NUMBER: 125

S&E BENCH LABORATORY NUMBER: EB-27

STATUS: Operational

OVERALL CONDITION: Good

EQUIPMENT ACTIVATION
DATE: 1983

ESTIMATED REPLACEMENT COST
OF EQUIPMENT: $2,500,000

Find...

CHOICE LIST SCREEN

States	California		
Towns	Boston		
Colors	Black		
Trees	Birch		
JSC-T&E	Air Bearing Floor		
JPL-T&E			
MSFC-T&E	Data Systems Test & Development		
GSFC-T&E			
AR&D	Autonomous Rendezvous and		
AR&D			
AR&D			

SECTION 5
DEFINITION OF DESIGN REFERENCE MISSIONS
(Prepared by Lockheed Engineering & Sciences Co-Houston)

5.1 INTRODUCTION

The work described in this section of the report was performed as part of the Pathfinder Project with the primary purpose of defining Design Reference Missions (DRMs) for potential AR&D applications. This information was released by LESC.

Previous rendezvous missions have only involved manned spacecraft. In all cases the rendezvous profiles were determined preflight and "hardwired" for the actual flights. For AR&D applications it is felt that profile design should be more flexible to the point of in-flight profile determination. Previously, rendezvous maneuvers either used transfers of integer multiples of one-half chase vehicle orbit revolutions or else rendezvous targeting formulations requiring fixed transfer times between maneuvers (Lambert targeting). Those formulations may be too constraining for AR&D missions. Some alternate targeting method that is not subject to these constraints is expected to be beneficial in that it can provide the necessary flexibility.

For that reason, the initial effort of this work was directed toward development of such a targeting algorithm. The result was a targeting algorithm that is based on the simultaneous solution of the time-of-flight equations of both the target and chase vehicle. The core algorithm is formulated using conic orbital mechanics that solves for the orbital elements of the chase vehicle simultaneously satisfying phasing and altitude requirements for a particular maneuver. When applied in a three-dimensional environment, full three-axis orbit control is achieved. As mentioned in the summary of activities associated with WP2 (section 2.1), the elimination of the time-of-flight constraint requires the inclusion of an alternate constraint to maintain a deterministic problem. While there are several possibilities, the terminal chase radial velocity was chosen for that constraint. This new algorithm has been labeled the ONCC maneuver. The mathematical formulation is described in detail in section 5.2 of this report.

The ONCC algorithm was implemented into the Satellite Targeting Algorithms for Rendezvous (STAR) program as well as in the Simulation Program for Rendezvous Integrating Navigation and Targeting (SPRINT) for testing and comparison with conventional targeting methods. Performance was compared using the NASA JSC Space Station Freedom reference profile. In addition, two rendezvous profiles were developed as candidate DRMs for AR&D. These two profiles are specifically designed utilizing the capabilities of ONCC targeting. A special small computer program was also developed that automatically generates a rendezvous profile from given chase vehicle differential altitudes with respect to the target and desired chase vehicle transfer angles between maneuvers. This program was used to generate the DRMs and to fly the second profile as a low Mars orbit rendezvous. Both profiles were tested using Monte Carlo simulation with SPRINT for Earth orbital rendezvous. The respective results are described in section 5.3 of this report.

5.2 DESCRIPTION OF ONCC TARGETING

5.2.1 Background

The primary concern of rendezvous targeting is the control of the phase angle as well as the differential altitude between the target and chaser spacecraft. Traditionally, the following maneuvers have been used to control phasing and altitude:

1. NC maneuver - controls phasing only in increments of full chase orbits.

2. NH maneuver - controls altitude only in increments of one-half orbit plus an integer number of complete chase vehicle orbits.

3. NCC maneuver - controls both phasing and altitude using fixed time-of-flight constraint (a Lambert maneuver).

While the first two maneuvers are single-axis maneuvers, (i.e., they apply delta-velocity in the local-horizontal direction only), the NCC maneuver is a three-axis maneuver, (i.e., it contains delta-velocity components in each local-vertical, local-horizontal axis). The transfer times for the NC and NH maneuvers are determined by the period of the corrected chase vehicle orbit, whereas the transfer times for the Lambert maneuver are typically determined before a mission and remain unaltered throughout the mission.

Although the NCC maneuver is capable of correcting both the altitude and phasing in one maneuver, the fixed transfer time constraint carries two disadvantages:

1. Transfer times must be individually determined for every rendezvous application and each maneuver within it.

2. Undesired radial components may be required to compensate for trajectory dispersions to meet the time-of-flight constraint.

It is therefore desirable to eliminate the time-of-flight constraint and replace it with some other constraint, with the goal of allowing for better trajectory shaping. The time-of-flight constraint for the NCC maneuver eliminates the need to include the relative motion between the target and chase vehicles in the orbit determination, as the initial and terminal chase vehicle positions remain inertially fixed. Thus, the relative motion must become a factor in the orbit determination process if the time of flight becomes dynamic. A reasonable constraint to replace the time-of-flight constraint may either be the initial or else the terminal velocity of the chase vehicle. By imposing velocity constraints to the solution, better trajectory shaping can be achieved. A logical initial velocity constraint would be to force the postmaneuver radial velocity component to equal the premaneuver radial velocity. Or alternately, the terminal radial velocity can be constrained, in which case, one can force the terminal point to be a particular point in the new orbit.

The subsequent sections describe a rendezvous maneuver targeting formulation that achieves simultaneous phasing and altitude control with one maneuver while imposing initial or terminal radial velocity constraints.

5.2.2 Approach

For targeting of such a rendezvous maneuver the following constraints must be satisfied:

1. phasing constraint:

$$(\theta_2-\theta_1)_C - (\theta_2-\theta_1)_T = \Delta\theta_C - \Delta\theta_T = \Delta\varphi \qquad (1)$$

where Δq_x = true anomaly difference for the Chase and Target vehicles, respectively,

and $\Delta\varphi = \varphi_2 - \varphi_1$

where φ_1 is the phase angle between chase and target vehicles at the time of the maneuver and φ_2 the phase angle at the target point.

2. altitude constraint: $r_{2T} - r_{2C} = \Delta h_2$ (2)

3. either an initial chase vehicle radial velocity constraint as

$$V_{r_1} = \frac{V_P}{\eta}\left(\frac{\lambda s}{r_1} - \lambda - x\eta\right) = V_{r_1\ old} \qquad (3a)$$

or else a terminal chase vehicle radial velocity constraint as

$$V_{r_2} = \frac{V_P}{\eta}\left(x\eta - \frac{\lambda s}{r_2} + \lambda\right) = V_{r\ desired} \qquad (3b)$$

where

s = semiperimeter of the transfer triangle defined by $\mathbf{r}_1, \mathbf{r}_2$ and the true anomaly difference between the two positions

$$\lambda = \frac{\sqrt{r_1 r_2}}{s}\cos\left(\frac{\theta_2-\theta_1}{2}\right)$$

$$\eta = \sqrt{1 - \lambda^2(1 - x^2)}$$

$$x = \text{normalized semimajor axis} = \sqrt{1 - \frac{s}{2a}}$$

$$V_p = \sqrt{\frac{2\mu}{s}}$$

The expressions for the radial velocity were derived by Battin (Battin, AIAA Journal, May 1977).

5.2.3 Maneuver Determination

$(\theta_2 - \theta_1)_T = \gamma_T$ is used as the independent variable. The function to be solved is the difference between the target and chase vehicles transfer time for a given set of transfer angles:

$$\Delta t_T - \Delta t_C = 0 \qquad (4)$$

The solution proceeds as follows:

1. An initial value for γ_T may be an already determined nominal value or else can be estimated using the following formulation:

$$\gamma_T = \frac{\Delta\varphi\ \omega_T}{\omega_C - \omega_T} \qquad (5)$$

where ω_T and ω_C are the orbital rate of the target and the new orbital rate of the chaser, respectively. A guess of the new chaser semimajor axis is necessary to compute a value for ω_C, which may be computed as

$$a_C = 0.5 * ((r_{1T} - \Delta H_2) + r_{1C})$$

r_{1T} is used here since the r_{2T} is not known at this time. r_{2T} cannot be computed until a value for the target transfer angle is determined. The familiar polar equation for elliptical orbits is used to obtain a value for r_{2T}:

$$r_{2_T} = \frac{p_T}{1 + e_T \cos(\theta_2)} \qquad (6)$$

where the following trigonometric identity is used to replace $e_T\cos(q_2)$:

$$e_T\cos(\theta_2) = e_T\cos(\theta_1+\gamma_T) = e_T\cos(\theta_1)\cos(\gamma_T) - e_T\sin(\theta_1)\sin(\gamma_T)$$

$e_T\cos(\theta_1)$ is also extracted from the polar equation and $e_T\sin(\theta_1)$ from $\sin^2(\theta_1) = 1 - \cos^2(\theta_1)$ or else from the relation:

$$e \sin(\theta) = \sqrt{\frac{p}{\mu}} V_{radial} \qquad (7)$$

2. With γ_C determined from equation (1) and r_{2C} from equation (2), equation (3) is solved using Newton-Raphson iteration to determine the new chaser semimajor axis. Subsequently both the target and the chaser transfer times are computed using the following time-of-flight equation, which is derived from the formulations described in references 1 and 2:.

$$\Delta t = \left(\frac{s}{V_p}\right) \frac{(n\pi + \tan^{-1}(f1/g1) - x\ w + x1\ y)}{w^3} \qquad (8)$$

where:

n = number of complete orbits
w = normalized mean orbital rate = $\sqrt{1 - x^2}$

$y = \eta - \lambda x$
$x1 = \lambda\ w$
$f1 = w\ y - x\ x1$
$g1 = x\ y + w\ x1$

3. The Secant method to find the roots of an equation is used to update the independent variable γ_T. This method requires a second initial guess for equation (5). A reasonable second initial value is found by subtracting $\Delta\varphi$ from the original value of γ_T.

4. The Secant method uses the latest two estimates of γ_T and the latest two Δt's to determine a new value for γ_T. The process is repeated until the two transfer times are sufficiently close (one thousand of the phasing requirement was found to be a good tolerance).

The new radial chase vehicle velocity is computed from equation (3a). The new horizontal chase vehicle velocity is computed from Battin (May 1977) as

$$V_h = \frac{V_p}{\eta}\sqrt{\frac{r_2}{r_1}}\sin\left(\frac{\theta_2-\theta_1}{2}\right) \tag{9}$$

Convergence problems may occur for transfer angles of close to 360° if the initial and terminal chase vehicle position magnitudes are different. Any other geometry including multirevolution transfers may be used. Nonetheless, the targeting altitude and phasing constraints must be physically reasonable to assure realistic transfer orbits.

A single iteration pass can only accommodate one boundary velocity constraint. However, if the dependent boundary value is not satisfactory, an additional iteration can be performed to adjust the maneuver time along the original chase vehicle orbit until the second constraint is satisfied. An additional iteration will be necessary to accommodate atmospheric and gravitational perturbations.

The described formulation does not use any assumptions other than the use of conic orbital motion.

5.2.4 Testing Of The ONCC Algorithm

A program was developed in Think Pascal on the Apple Macintosh IIx to test this algorithm. This program uses two-dimensional pure conic orbit propagation. To obtain meaningful test inputs, the capability to develop the geometry of an entire rendezvous profile from user input differential altitudes with respect to the target and desired chase vehicle transfer angles between the maneuver point and the target point was added to the program. The construction of the profile assures all maneuvers to be nominally horizontal. The program generates a spread sheet compatible data file to allow plotting of the resulting relative motion profile with conventional plotting programs. Following maneuver point determination the program "flies" the rendezvous trajectory computing the maneuvers using the formulation described above.

To stress test the formulation, a Monte Carlo capability was added to the program. Once the rendezvous profile is generated, the differential altitudes and ranges of the nominal maneuver points are dispersed using normally distributed random numbers. However, the altitude and range dispersions are not correlated. Furthermore, no velocity dispersions as such are given. This standard deviation of the dispersions is provided via user input as a percentage of the nominal values at each maneuver point. The terminal point is kept undispersed. Targeting is done from one dispersed maneuver point to the next dispersed maneuver point. The total number of cases is specified by the user.

It was found that better convergence on the maneuver solution is achieved when a terminal radial velocity constraint is given rather than an initial radial velocity constraint. Monte Carlo simulations were performed with and without iteration on the maneuver time to minimize radial delta-velocity of the maneuver. When the latter is minimized the iterations are terminated when the ratio of radial to horizontal DV component becomes less than 5%.

Typical results from this Monte Carlo simulation are shown in tables 5-I and tables 5-II, which show the maneuver summary for AR&D DRM 2, which is described in section 5.3 in this report, without and with maneuver optimization. For each set 100 cases were run using 5% 1s dispersions. Though fairly large dispersions were used, no convergence problems were encountered. It is interesting to note that the statistics of the optimizing simulation is significantly improved over the non-optimized statistics. The non-optimized simulation statistics show the trajectory dispersions as generated by the random numbers. The trajectory statistics of the optimized simulation reflect the adjustments in maneuver time. It must be emphasized again that for these Monte Carlo simulations the dispersions are not correlated and that no relative navigation is performed to reduce the dispersions. This capability was only added to provide some stress test capability for the ONCC targeting logic and rendezvous profile design.

Once the new maneuver determination formulation was found to be robust in a conic orbital environment, the logic was implemented into the STAR program to be tested in a simulated three-dimensional perturbed environment. No problems were encountered to accommodate gravitational and atmospheric dispersions in the maneuver solution process with the exception of transfers close to 180°. The problem here is the same as for Lambert targeting, in that two colinear position vectors do not define a unique orbit plane thus causing difficulties in determining the direction of the new velocity vector. Nonetheless, acceptable convergence was achieved at the expense of some accuracy.

While the Monte Carlo capability provides a crude means to stress test ONCC targeting, a better test is an actual Monte Carlo simulation where navigation measurements are simulated. For that reason, ONCC targeting was also implemented into SPRINT and tested by applying ONCC targeting to all maneuvers of the Space Station Freedom reference rendezvous profile with the exception of the NC 1 maneuver. This profile was designed by W. L. Jackson at JSC. A description of this profile can be found in Spehar and Clark (1988). The optimizing logic for ONCC targeting was set up to reduce radial delta-velocity components to less than 5% of the horizontal delta-velocity components. This caused convergence problems with nominally zero maneuvers (midcourse corrections). For that reason, the capability was added to optionally disable optimization for a particular maneuver. No convergence problems were encountered with optimization disabled. Tables 5-III shows the Monte Carlo maneuver summary for this profile using ONCC targeting and can be compared with the baseline data for this profile shown in table 5-IV. The ONCC targeted profile shows improved efficiency compared to the baseline. The baseline reference profile employs several single-axis maneuvers that do not lead themselves to be targeted efficiently using ONCC targeting. Additional performance improvement is expected for profiles that employ three-axes maneuvers only, as the DRM 1 described in the following section.

Table 5-I: Non-Optimized DRM 2 Monte Carlo:

Random number seed:	57648736
Number of cases:	100
Dispersion factor:	5.000 %
Target altitude:	400.00 km

Rendezvous maneuver 1:

Radial velocity:	-0.00	m/s		
Mean time:	0.00,	sigma:	0.00	min.
Mean downrange:	-334.89,	sigma:	17.46	km
Mean Delta-H:	34.23,	sigma:	1.90	km
Mean Delta-VH:	4.05,	sigma:	0.53	m/s
Mean Delta-VR:	0.24,	sigma:	3.31	m/s
Transfer angle:	240.61,	sigma:	23.56	deg.

Rendezvous maneuver 2:

Radial velocity:	-7.30	m/s		
Mean time:	61.52,	sigma:	46.62	min.
Mean downrange:	-177.02,	sigma:	9.83	km
Mean Delta-H:	23.10,	sigma:	1.08	km
Mean Delta-VH:	4.34,	sigma:	0.88	m/s
Mean Delta-VR:	-0.13,	sigma:	1.31	m/s
Transfer angle:	239.58,	sigma:	9.80	deg.

Rendezvous maneuver 3:

Radial velocity:	0.02	m/s		
Mean time:	122.83,	sigma:	41.65	min.
Mean downrange:	-42.20,	sigma:	2.16	km
Mean Delta-H:	12.10,	sigma:	0.63	km
Mean Delta-VH:	6.27,	sigma:	0.44	m/s
Mean Delta-VR:	-0.20,	sigma:	0.80	m/s
Transfer angle:	118.02,	sigma:	7.57	deg.

Rendezvous maneuver 4:

Radial velocity:	3.62	m/s		
Mean time:	153.11,	sigma:	44.36	min.
Mean downrange:	-10.20,	sigma:	0.48	km
Mean Delta-H:	6.52,	sigma:	0.30	km
Mean Delta-VH:	2.13,	sigma:	0.27	m/s
Mean Delta-VR:	-0.04,	sigma:	0.44	m/s
Transfer angle:	119.97,	sigma:	4.29	deg.

Rendezvous maneuver 5:

Radial velocity:	0.01	m/s		
Mean time:	183.94,	sigma:	44.61	min.
Mean downrange:	-2.04,	sigma:	0.10	km
Mean Delta-H:	0.93,	sigma:	0.05	km
Mean Delta-VH:	2.46,	sigma:	0.17	m/s
Mean Delta-VR:	0.00,	sigma:	0.06	m/s
Transfer angle:	120.45,	sigma:	3.48	deg.

Rendezvous maneuver 6:

Radial velocity:	0.60	m/s		
Mean time:	214.90,	sigma:	43.38	min.
Mean downrange:	0.00,	sigma:	0.00	km
Mean Delta-H:	0.00,	sigma:	0.00	km
Mean Delta-VH:	0.18,	sigma:	0.02	m/s
Mean Delta-VR:	0.60,	sigma:	0.00	m/s
Transfer angle:	0.00,	sigma:	0.00	deg.
Total Delta-V:	21.26,	sigma:	1.72	m/s

Table 5-II: Optimized DRM 2 Monte Carlo:

Random number seed:	57648736
Number of cases:	100
Dispersion factor:	5.000 %
Target altitude:	400.00 km

Rendezvous maneuver 1:

Radial velocity:	-0.00	m/s		
Mean time:	0.00,	sigma:	0.00	min.
Mean downrange:	-332.70,	sigma:	11.36	km
Mean Delta-H:	34.23,	sigma:	1.90	km
Mean Delta-VH:	4.27,	sigma:	0.42	m/s
Mean Delta-VR:	-0.01,	sigma:	0.02	m/s
Transfer angle:	241.54,	sigma:	11.34	deg.

Rendezvous maneuver 2:

Radial velocity:	-7.30	m/s		
Mean time:	61.77,	sigma:	22.55	min.
Mean downrange:	-176.93,	sigma:	5.54	km
Mean Delta-H:	23.12,	sigma:	1.59	km
Mean Delta-VH:	4.14,	sigma:	0.84	m/s
Mean Delta-VR:	-0.01,	sigma:	0.02	m/s
Transfer angle:	240.14,	sigma:	6.54	deg.

Rendezvous maneuver 3:

Radial velocity:	0.02	m/s		
Mean time:	123.22,	sigma:	20.41	min.
Mean downrange:	-42.48,	sigma:	1.62	km
Mean Delta-H:	12.12,	sigma:	0.64	km
Mean Delta-VH:	6.26,	sigma:	0.46	m/s
Mean Delta-VR:	-0.02,	sigma:	0.02	m/s
Transfer angle:	118.73,	sigma:	5.49	deg.

Rendezvous maneuver 4:

Radial velocity:	3.62	m/s		
Mean time:	153.68,	sigma:	23.72	min.
Mean downrange:	-10.42,	sigma:	1.09	km
Mean Delta-H:	6.57,	sigma:	0.69	km
Mean Delta-VH:	2.14,	sigma:	0.28	m/s
Mean Delta-VR:	-0.01,	sigma:	0.01	m/s
Transfer angle:	120.86,	sigma:	3.11	deg.

Rendezvous maneuver 5:

Radial velocity:	0.01	m/s		
Mean time:	184.73,	sigma:	23.09	min.
Mean downrange:	-2.04,	sigma:	0.10	km
Mean Delta-H:	0.93,	sigma:	0.05	km
Mean Delta-VH:	2.45,	sigma:	0.21	m/s
Mean Delta-VR:	0.00,	sigma:	0.01	m/s
Transfer angle:	120.18,	sigma:	3.62	deg.

Rendezvous maneuver 6:

Radial velocity:	0.60	m/s		
Mean time:	215.63,	sigma:	24.27	min.
Mean downrange:	0.00,	sigma:	0.00	km
Mean Delta-H:	0.00,	sigma:	0.00	km
Mean Delta-VH:	0.18,	sigma:	0.02	m/s
Mean Delta-VR:	0.60,	sigma:	0.00	m/s
Transfer angle:	0.00,	sigma:	0.00	deg.
Total Delta-V:	19.89,	sigma:	1.08	m/s

Table 5-III: Space Station Freedom Rendezvous Reference Mission Monte Carlo Maneuver Summary with ONCC Targeting

Maneuver:	ΔV - X:	ΔV - Y:	ΔV - Z:	ΔV-Total:
NH 1	26.65 ± 0.65	0.00 ± 0.00	0.57 ± 0.38	26.66 ± 0.66
NC 1	13.66 ± 0.81	0.00 ± 0.00	0.00 ± 0.00	13.66 ± 0.81
NCC 0	0.81 ± 0.37	2.49 ± 1.82	1.01 ± 0.71	3.79 ± 1.37
NH 2	19.74 ± 0.59	2.06 ± 1.67	0.37 ± 0.35	19.92 ± 0.62
NCC 1	8.68 ± 0.28	0.61 ± 0.68	0.17 ± 0.17	8.73 ± 0.28
NCC 2	1.15 ± 0.28	0.43 ± 0.39	0.10 ± 0.18	1.68 ± 0.45
Total ΔV:				74.43 ± 1.86

(All maneuver ΔVs are in fps)

Table 5-IV: Space Station Freedom Rendezvous Reference Mission Monte Carlo Maneuver Summary with Baseline Targeting.

Maneuver:	ΔV - X:	ΔV - Y:	ΔV - Z:	ΔV-Total:
NH 1	26.95 ± 0.55	0.00 ± 0.00	0.00 ± 0.00	26.95 ± 0.55
NC 1	14.15 ± 2.14	0.00 ± 0.00	0.00 ± 0.00	14.15 ± 2.14
NCC 0	0.28 ± 2.73	2.73 ± 2.12	0.91 ± 0.78	3.41 ± 1.85
NH 2	19.52 ± 1.66	0.00 ± 0.00	2.59 ± 2.07	19.79 ± 1.08
NCC 1	8.61 ± 0.16	2.95 ± 2.46	2.07 ± 1.56	9.72 ± 1.08
NCC 2	1.11 ± 0.40	0.68 ± 0.60	0.43 ± 0.34	2.20 ± 0.77
Total ΔV:				76.22 ± 3.10

(All maneuver ΔVs are in fps)

An accurate value for the sigma on the total ΔV was not available, but was estimated from available data.

5.3 AR&D DESIGN REFERENCE MISSIONS

5.3.1 Design Philosophy

For AR&D missions it was felt that the number of maneuvers to be performed during the rendezvous should be kept to a minimum. Nonetheless, several maneuvers must be performed to correct trajectory dispersions after processing relative navigation measurements. Standard Shuttle star tracking and rendezvous radar measurements were considered as the only relative navigation means for the development of the DRMs.

Profile design was based on the observation that two consecutive 240° chase vehicle transfer angles or two consecutive 120° transfer angle maneuvers result in the terminal chase vehicle radial velocity of the second maneuver to be the same as the radial velocity component of the chase vehicle following the application of the first maneuver, provided that the altitude of the second maneuver falls half way between the altitude of the first maneuver and the terminal altitude of the second maneuver. For example, if the chase vehicle is at an apoapsis or periapsis relative to the target orbit after completion of the first maneuver, then the terminal point of the second maneuver is also a relative apoapsis or periapsis, respectively. Moreover, the second maneuver of this two-maneuver set is always nominally horizontal. If these conditions are met, then, if the first maneuver is applied at an apoapsis or else at a periapsis, both maneuvers are horizontal.

In addition to the observed characteristics, 240° and 120° transfers provide good simultaneous phasing, altitude and out-of-plane control capability.

5.3.2 Candidate Design Reference Missions

Two candidate DRMs for AR&D applications have been developed with the aid of the ONCC targeting and the Pascal program described in section 5.2. Both DRMs make use of the characteristics described in the previous section.

The first DRM (DRM 1) comprises one set of two 240° transfers and two sets of two 120° transfers, with a total of seven maneuvers including a velocity null at V-bar intercept. The target altitude of the last maneuver is placed above the V-bar line such that the line is crossed with about 0.5 m/s (1.6 fps) radial velocity. The resulting transfer angle to V-bar is about 100°. The respective relative motion profile is shown in figure 5.1a and a list of the nominal maneuvers for DRM 1 is shown in table 5-V. In this particular profile, the differential altitudes between the two subsequent sets of those two impulse maneuver sets are not correlated. The first star-tracking period occurs prior to the first maneuver. If the second star-tracking period occurs 30-10 minutes prior to the second maneuver, then a third star-tracking period occurs between the third and the fourth maneuver. Rendezvous radar range is reached well before the fourth maneuver such that this maneuver is supported by range and range rate measurements. The profile as shown is targeted for direct intercept; in reality, one would target for some V-bar offset. This will shift all downrange numbers by the appropriate amount.

The second DRM (DRM 2), shown in figure 5-1b, is very similar to DRM 1. However, it consists of one set of two 240° transfers, one set of two 120° transfers, plus one 120° terminal phase, comprising a total of six maneuvers including a velocity null at the V-bar intercept. The terminal phase maneuver causes the chase vehicle to cross the V-bar line with 0.6 m/s (2.0 fps) radial velocity. For this profile the differential altitudes with respect to the target are correlated by the following algorithm:

Let $\Delta H_{TI} = \Delta H_5$ be the desired differential altitude of the terminal phase (fifth maneuver) and ΔH_1 be the differential altitude of the first maneuver then

$$\Delta H_3 = \Delta H_{TI} + (\Delta H_1 - \Delta H_{TI})/3 \qquad \text{and}$$
$$\Delta H_5 = \Delta H_{TI} + (\Delta H_3 - \Delta H_{TI})/3 \qquad \text{and}$$

$$\Delta H_2 = (\Delta H_3 + \Delta H_1)/2 \qquad \text{and}$$
$$\Delta H_4 = (\Delta H_5 + \Delta H_3)/2 \quad .$$

With the use of this algorithm, the given nominal transfer angles between maneuvers, plus constraining all be horizontal, the nominal downrange positions as well as the nominal transfer times can be determined. This is done in the Pascal program mentioned earlier. This method provides a simple means to determine the rendezvous profile in flight, based on the latest measurement of the chase vehicle altitude prior to commencing the rendezvous. The algorithm that performs the radial velocity constraint generates the required chase vehicle transfer orbital elements and is already an essential part of ONCC targeting. The remaining code to determine the profile is very simple and small in size.

The same sensor periods apply for DRM 2 as for DRM 1, with the exception that the third maneuver marginally falls into the standard rendezvous radar range and may possibly be supported. A list of the nominal DRM 2 maneuvers is found in table 5-VI.

Both profiles were simulated with STAR and with Monte Carlo simulation using SPRINT employing ONCC targeting exclusively. For comparison, DRM 1 was also simulated in SPRINT with Lambert targeting only. The results of the Monte Carlo simulations are listed in tables 5-VII-5-IX. For DRM 1, a midcourse correction maneuver was added halfway between the velocity null and the previous maneuver. DRM 2 was simulated without a midcourse correction. The third maneuver was supported with range and range rate measurements in the SPRINT simulations. It should mentioned that all cases used the same profile geometry instead of determining a new geometry for every case. Additional improvement in performance is expected if the geometry is newly determined based on the latest sensor measurements for every case.

DRM 1 shows less overall delta-velocity requirements when flown using ONCC targeting than when flown with Lambert targeting. Furthermore, the accuracy of arriving at the terminal point of the profile is improved over Lambert targeting. On the other hand, the time of arrival varies when ONCC targeting is used and deviated in some cases by more than 15 minutes from the nominal. Most of the time variations occur after processing range and rate measurements, so that the star tracking periods were not significantly affected.

DRM 2 showed very similar results to DRM 1. The dispersions about the terminal point were significantly larger than DRM 1, which is primarily attributed to the lack of the midcourse correction prior to the velocity null.

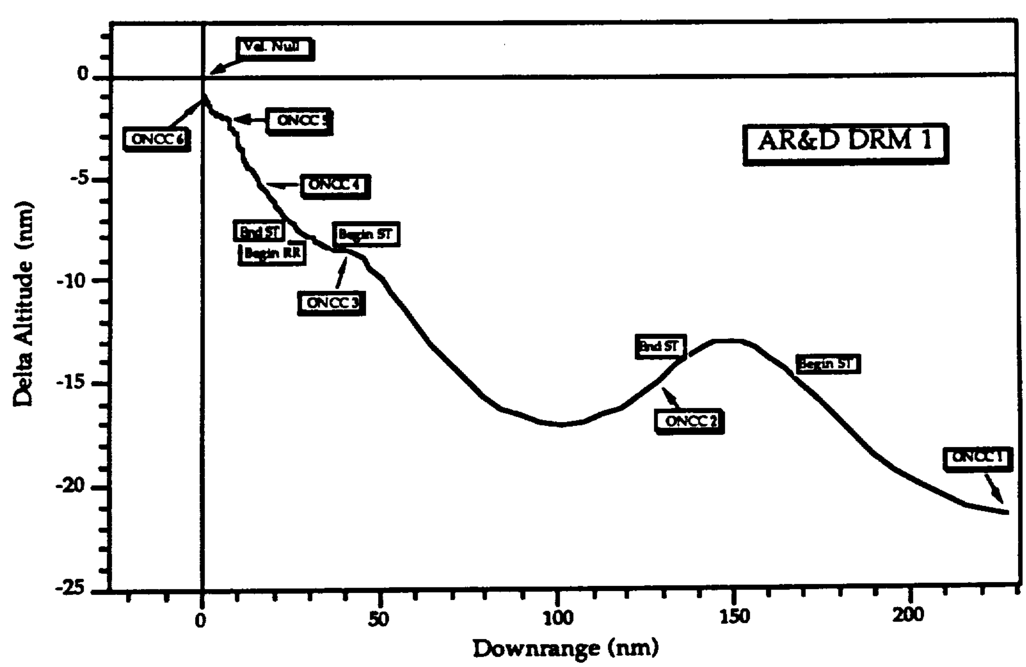


Figure 5-1a. AR&D DRM 1 Relative Motion Profile.

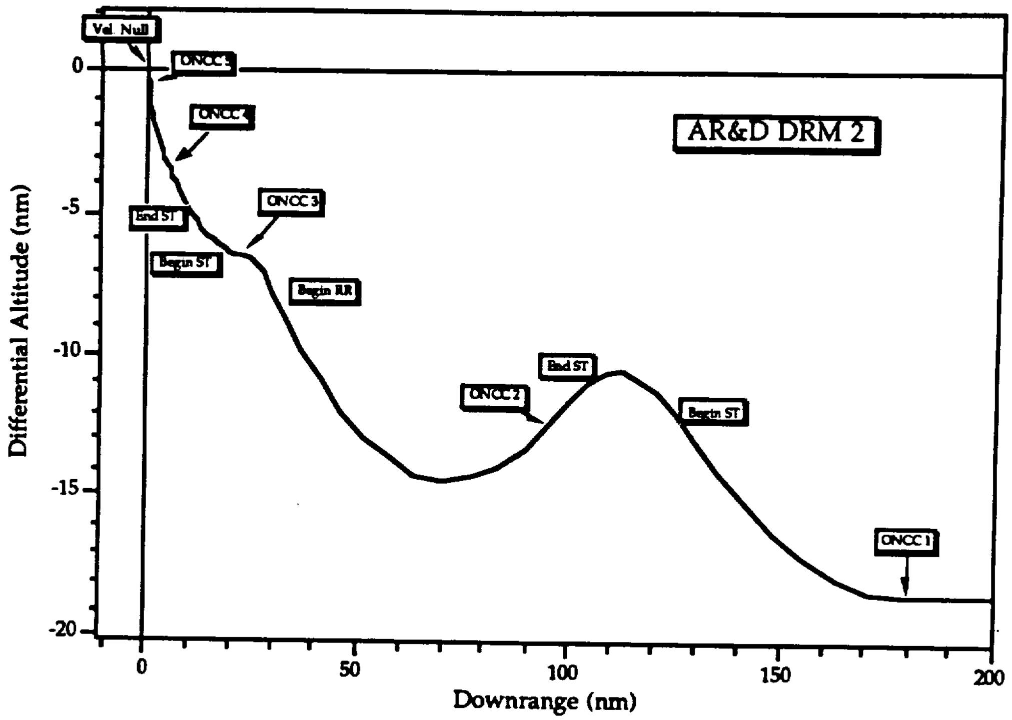


Figure 5-1b. AR&D DRM 2 Relative Motion Profile

Table 5-V. AR&D DRM 1 for Low Earth Orbit

Maneuver Summary For Target at 400.00 km (216 nm):

Rendezvous maneuver 1:

Radial velocity:	-0.11	m/s	(0.36 fps)
Mean time:	0.00	min.	
Mean downrange:	-427.12	km	(-230.63 nm)
Mean Delta-H:	40.00	km	(21.60 nm)
Mean Delta-VH:	4.53	m/s	(14.86 fps)
Mean Delta-VR:	0.11	m/s	(0.36 fps)
Transfer angle:	240.00	deg.	

Rendezvous maneuver 2:

Radial velocity:	-7.79	m/s	(-25.26 fps)
Mean time:	61.30	min.	
Mean downrange:	-239.43	km	(-129.28 nm)
Mean Delta-H:	28.00	km	(15.11 nm)
Mean Delta-VH:	4.61	m/s	(15.12 fps)
Mean Delta-VR:	0.00	m/s	
Transfer angle:	240.00	deg.	

Rendezvous maneuver 3:

Radial velocity:	-0.09	m/s	(-0.30 fps)
Mean time:	122.65	min.	
Mean downrange:	-75.56	km	(-40.80 nm)
Mean Delta-H:	16.00	km	(8.64 nm)
Mean Delta-VH:	6.82	m/s	(22.38 fps)
Mean Delta-VR:	0.00	m/s	
Transfer angle:	120.00	deg.	

Rendezvous maneuver 4:

Radial velocity:	4.02	m/s	(13.19 fps)
Mean time:	153.41	min.	
Mean downrange:	-30.79	km	(-16.63 nm)
Mean Delta-H:	10.00	km	(5.40 nm)
Mean Delta-VH:	2.22	m/s	(7.28 fps)
Mean Delta-VR:	-0.00	m/s	
Transfer angle:	120.00	deg.	

Rendezvous maneuver 5:

Radial velocity	-0.09	m/s	(-0.30 fps)
Mean time:	184.23	min.	
Mean downrange:	-12.76	km	(-6.89 nm)
Mean Delta-H:	4.00	km	(2.16 nm)
Mean Delta-VH:	3.07	m/s	(10.07 fps)
Mean Delta-VR:	-0.00	m/s	
Transfer angle:	120.00	deg.	

Rendezvous maneuver 6:

Radial velocity:	1.40	m/s	(4.59 fps)
Mean time:	215.06	min.	
Mean downrange:	-1.90	km	(-1.03 nm)
Mean Delta-H:	2.00	km	(1.08 nm)
Mean Delta-VH:	0.73	m/s	(2.40 fps)
Mean Delta-VR:	-0.00	m/s	
Transfer angle:	100.00	deg.	

Rendezvous maneuver 7 (Vel. Null):

Radial velocity:	0.50	m/s	(1.64 fps)
Mean time:	240.77	min.	
Mean downrange:	0.00	km	
Mean Delta-H:	0.00	km	
Mean Delta-VH:	0.75	m/s	(2.46 fps)
Mean Delta-VR:	0.50	m/s	(1.64 fps)
Transfer angle:	0.00	deg.	
Total Delta-V:	22.88	m/s	(75.07 fps)

Table 5-VI. AR&D DRM 2 for Low Earth Orbit

Maneuver Summary For Target at 400.00 km (216 nm):

Rendezvous maneuver 1:

Radial velocity:	-0.01	m/s	(-0.03 fps)
Mean time:	0.00	min.	
Mean downrange:	-333.46	km	(-180.0 nm)
Mean Delta-H:	34.26	km	(18.5 nm)
Mean Delta-VH:	4.22	m/s	(13.85 fps)
Mean Delta-VR:	0.01	m/s	(0.03 fps)
Transfer angle:	240.00	deg.	

Rendezvous maneuver 2:

Radial velocity:	-7.30	m/s	(-23.95 fps)
Mean time:	61.37	min.	
Mean downrange:	-177.67	km	(-95.9 nm)
Mean Delta-H:	23.15	km	(12.5 nm)
Mean Delta-VH:	4.21	m/s	(13.81 fps)
Mean Delta-VR:	0.00	m/s	(0.00 fps)
Transfer angle:	240.00	deg.	

Rendezvous maneuver 3:

Radial velocity:	0.00	m/s	
Mean time:	122.79	min.	
Mean downrange:	-42.73	km	(-23.1 nm)
Mean Delta-H:	12.04	km	(6.5 nm)
Mean Delta-VH:	6.31	m/s	(20.70 fps)
Mean Delta-VR:	0.00	m/s	(0.00 fps)
Transfer angle:	120.00	deg.	

Rendezvous maneuver 4:

Radial velocity:	3.63	m/s	(11.91 fps)
Mean time:	153.57	min.	
Mean downrange:	-10.16	km	(-5.5 nm)
Mean Delta-H:	6.48	km	(3.5 nm)
Mean Delta-VH:	2.10	m/s	(6.89 fps)
Mean Delta-VR:	-0.00	m/s	(0.00 fps)
Transfer angle:	120.00	deg.	

Rendezvous maneuver 5:

Radial velocity:	0.00	m/s	
Mean time:	184.41	min.	
Mean downrange:	-2.03	km	(-1.1 nm)
Mean Delta-H:	0.93	km	(0.5 nm)
Mean Delta-VH:	2.44	m/s	(8.00 fps)
Mean Delta-VR:	-0.00	m/s	(0.00 fps)
Transfer angle:	120.00	deg.	

Rendezvous maneuver 6:

Radial velocity:	0.60	m/s	(1.97 fps)
Mean time:	215.26	min.	
Mean downrange:	0.00	km	
Mean Delta-H:	0.00	km	
Mean Delta-VH:	0.18	m/s	(0.59 fps)
Mean Delta-VR:	0.60	m/s	(1.97 fps)
Transfer angle:	0.00	deg.	
Total Delta-V:	19.90	m/s	(65.29 fps)

Table 5-VII. AR&D DRM 1 Monte Carlo Maneuver Summary with ONCC Targeting

Maneuver:	ΔV - X:	ΔV - Y:	ΔV - Z:	ΔV-Total:
NCC 1	14.72 ± 0.39	1.47 ± 1.19	0.27 ± 0.22	14.84 ± 0.39
NCC 2	14.62 ± 0.83	1.84 ± 1.16	0.16 ± 0.19	14.78 ± 0.82
NCC 3	21.95 ± 0.40	0.65 ± 0.51	0.40 ± 0.31	21.97 ± 0.22
NCC 4	7.09 ± 0.68	0.19 ± 0.14	0.17 ± 0.10	7.09 ± 0.68
NCC 5	9.84 ± 0.62	0.08 ± 0.05	0.20 ± 0.17	9.84 ± 0.62
NCC 6	2.34 ± 0.35	0.03 ± 0.03	0.06 ± 0.32	2.41 ± 0.37
NCC 7	0.12 ± 0.13	0.04 ± 0.04	0.28 ± 0.26	0.44 ± 0.37
Vel. Null	2.22 ± 0.78	0.17 ± 0.12	1.55 ± 0.43	3.34 ± 1.00
Total ΔV				74.71 ± 1.67

(all maneuvers are in fps, NCC 7 is a midcourse correction)

Environment position at velocity null maneuver:
LVLH-X: -29 ± 141, LVLH-Y: 2 ± 31, LVLH-Z: -22 ± 80 feet.

Table 5-VIII. AR&D DRM 1 Monte Carlo Maneuver Summary with Lambert Targeting

Maneuver:	ΔV - X:	ΔV - Y:	ΔV - Z:	ΔV-Total:
NCC 1	14.75 ± 0.73	1.46 ± 1.17	0.77 ± 0.58	14.90 ± 0.71
NCC 2	14.57 ± 0.72	1.88 ± 1.16	2.38 ± 1.94	15.04 ± 0.92
NCC 3	22.00 ± 0.45	0.63 ± 0.49	2.32 ± 2.01	22.23 ± 0.61
NCC 4	7.05 ± 0.95	0.20 ± 0.15	1.76 ± 1.45	7.40 ± 1.02
NCC 5	9.97 ± 0.56	0.07 ± 0.06	1.69 ± 1.84	10.27 ± 0.71
NCC 6	2.22 ± 0.30	0.02 ± 0.02	1.07 ± 1.01	3.09 ± 0.68
NCC 7	0.09 ± 0.09	0.05 ± 0.04	0.25 ± 0.20	0.39 ± 0.25
Vel. Null	2.50 ± 0.52	0.20 ± 0.14	1.94 ± 0.61	3.46 ± 0.44
Total ΔV				75.77 ± 2.59

(all maneuvers are in fps, NCC 7 is a midcourse correction)

Environment Position at velocity null Maneuver:
LVLH-X: -37±176, LVLH-Y: 2±38, LVLH-Z: -38±95 feet.

Table 5-IX. AR&D DRM 2 Monte Carlo Maneuver Summary with ONCC Targeting

Maneuver:	ΔV - X:	ΔV - Y:	ΔV - Z:	ΔV-Total:
NCC 1	13.85 ± 0.46	1.73 ± 1.40	0.21±0.20	14.03 ± 0.46
NCC 2	13.85 ± 0.88	1.71 ± 1.05	0.19±0.19	14.00 ± 0.88
NCC 3	20.58 ± 0.43	0.29 ± 0.27	0.58±2.07	20.68 ± 0.79
NCC 4	7.11 ± 0.38	0.33 ± 0.24	0.17±0.11	7.12 ± 0.38
NCC 5	7.87 ± 0.31	0.33 ± 0.23	0.15±.013	7.88 ± 0.31
Vel. Null	0.65 ± 0.35	0.06 ± 0.05	1.98±0.04	2.69 ± 0.34
Total ΔV				66.40 ± 1.50

Environment Position at velocity null Maneuver:
LVLH-X: 974 ± 787, LVLH-Y: 11 ± 58, LVLH-Z: -13 ± 243 feet.

5.3.3 Non-Low Earth Orbit Rendezvous

The DRM 2 algorithm was also used to generate a reference profile for a low Mars orbit with the Pascal program by using Mars parameters rather than Earth parameters. Table 5-X shows the maneuvers for DRM 1 applied to a Mars rendezvous. The identical differential altitudes as for DRM 1 in low Earth Orbit (LEO) were used. It is interesting to note that the downrange distances of the chase vehicle at each maneuver point are practically identical to the LEO case. The resulting relative motion profile, when drawn on the same scale, overlays completely with the respective LEO profile. Therefore, a relative motion plot is not included in this report. The desired radial V-bar intercept velocity had to be adjusted to 0.45 m/s (1.48 fps) to preserve a 120° terminal phase. Differences are observed in maneuver delta-velocities and transfer times. There was not enough time to change the STAR or the SPRINT program to simulate the Mars environment and as a result no three-DOF or Monte Carlo simulation could be performed. Nonetheless, very similar performance to the LEO case is expected because of the virtually identical profile geometry.

Based on observation, the DRM 1 rendezvous profile using the same differential altitudes was also produced for geosynchronous Earth orbit. As expected, the downrange numbers for this application were also identical to the LEO case, but with substantially different delta-velocity requirements and transfer times. Again, the V-bar radial intercept velocity had to be adjusted to preserve a 120° terminal phase. A maneuver summary table is not included in this report, because the total rendezvous time is about 65 hours, which is probably not practical for a rendezvous.

It appears that DRM 2 is a suitable candidate for various AR&D missions because of the algorithmic correlation between maneuver points and the insensitivity of the profile geometry to the orbit environment.

Table 5-X. AR&D DRM 2 for Low Mars Orbit

Maneuver Summary For Target at 500.00 km (270 nm)

Rendezvous maneuver 1:

Radial velocity:	-0.02	m/s	(0.06 fps)
Mean time:	0.00	min.	
Mean downrange:	-333.30	km	(-180.0 nm)
Mean Delta-H:	34.26	km	(18.5 nm)
Mean Delta-VH:	3.18	m/s	(10.43 fps)
Mean Delta-VR:	0.02	m/s	(0.00 fps)
Transfer angle:	240.00	deg.	

Rendezvous maneuver 2:

Radial velocity:	-5.50	m/s	(-18.04 fps)
Mean time:	81.30	min.	
Mean downrange:	-177.58	km	(-95.9 nm)
Mean Delta-H:	23.15	km	(12.5 nm)
Mean Delta-VH:	3.18	m/s	(10.43 fps)
Mean Delta-VR:	0.00	m/s	(0.00 fps)
Transfer angle:	240.00	deg.	

Rendezvous maneuver 3:

Radial velocity:	0.01	m/s	(0.03 fps)
Mean time:	162.71	min.	
Mean downrange:	-42.72	km	(-23.1 nm)
Mean Delta-H:	12.04	km	(6.5 nm)
Mean Delta-VH:	4.75	m/s	(15.58 fps)
Mean Delta-VR:	0.00	m/s	(0.00 fps)
Transfer angle:	120.00	deg.	

Rendezvous maneuver 4:

Radial velocity:	2.73	m/s	(8.96 fps)
Mean time:	203.58	min.	
Mean downrange:	-10.17	km	(-5.6 nm)
Mean Delta-H:	6.48	km	(3.5 nm)
Mean Delta-VH:	1.58	m/s	(5.18 fps)
Mean Delta-VR:	-0.00	m/s	(0.00 fps)
Transfer angle:	120.00	deg.	

Rendezvous maneuver 5:

Radial velocity:	0.00	m/s	
Mean time:	244.58	min.	
Mean downrange:	-2.03	km	(-1.1 nm)
Mean Delta-H:	0.93	km	(0.5 nm)
Mean Delta-VH:	1.84	m/s	(6.04 fps)
Mean Delta-VR:	-0.00	m/s	(0.00 fps)
Transfer angle:	120.00	deg.	

Rendezvous maneuver 6:

Radial velocity:	0.45	m/s	(1.48 fps)
Mean time:	285.62	min.	
Mean downrange:	0.00	km	
Mean Delta-H:	0.00	km	
Mean Delta-VH:	0.13	m/s	(0.43 fps)
Mean Delta-VR:	0.45	m/s	(1.48 fps)
Transfer angle:	0.00	deg.	
Total Delta-V:	15.00	m/s	(49.21 fps)

5.4 SUMMARY

A transfer time-independent formulation for the determination of a three-axes rendezvous maneuver was developed as a means to provide orbit targeting flexibility for AR&D missions. This formulation permits the use of radial velocity constraints to allow for better orbit shaping. The inclusion of velocity constraints can be used to assure the maneuvers to be horizontal. There is no restriction on the transfer angle between maneuvers with the exception of maneuvers of multiples of complete chase vehicle orbits. In the latter case, the initial and terminal altitudes must be the same to assure convergence. The formulation has been successfully tested when subjected to gravitational and atmospheric perturbations. Moreover, Monte Carlo tests indicate that this algorithm is quite robust and efficient for handling navigational and trajectory dispersions. In the future it may be worth investigating if constraints other than the terminal radial velocity constraint may provide additional benefits.

Two DRMs were developed as candidates for AR&D missions. Both are composed of sequences of 120° and 240° chase vehicle transfers and require nominal horizontal maneuvers. DRM 2 is designed using algorithmically correlated differential altitude permitting in-flight profile determination. Furthermore, the DRM 2 profile geometry appears to be insensitive to the orbit environment, making this profile well suited as a standard profile for AR&D.

5.5 REFERENCES

1. "A New Solution For Lambert's Problem," R. H. Battin, 19th Congress of the International Astronautical Federation, New York 1968.

2. "Lambert's Problem Revisited," R. H. Battin, AIAA Journal, May 1977.

3. "Validation Of Space Station Relative State Requirements," P.T. Spehar and F.D. Clark, LESC Corres. No. 94-88, 1988.

SECTION 6
AR&D PRELIMINARY TRAJECTORY CONTROL OPERATIONS REQUIREMENTS
(Prepared by Lin Com)

6.1 INTRODUCTION

As stated previously, development of AR&D technology is an enabling requirement for future space operations such as the MRSR mission and lunar base colonization. Also, responsibility for protecting the Nation's investment in satellites requires formulation of a comprehensive plan for on-orbit servicing, which is defined as "any activity performed on orbit to assemble, maintain, repair, resupply, upgrade, deploy, retrieve, or return various spacecraft and/or facilities." In addition, the Department of Defense has identified a number of situations which would benefit from appropriate on-orbit servicing capabilities, especially highly "autonomous" operations.

Autonomous operations for on-orbit servicing will

1. Permit servicing in locations not accessible by humans
2. Permit servicing of hazardous targets and cargos
3. Relieve the servicing load from systems like the Space Shuttle and make these systems available to perform other activities
4. Become increasingly cost effective as satellites increase in cost, complexity, size, mass and frequency of required servicing

A natural evolution of this technology will be an application to missions involving other planets. As AR&D technology develops, it can first be tested in LEO for satellite repair and servicing, with ground control assistance if needed. This technology can be progressively used in other environments like that of Mars and the Moon. Furthermore, this technology can play a significant role in initial and evolutionary development of Space Station Freedom.

6.2 SCOPE

Trajectory Control is one aspect of the AR&D Project that will feed overall system requirements. This section summarizes the results of a trajectory control requirements definition study. It also describes the inter-relationship among the subsystems as appropriate and identifies the control requirements at systems level. This information was released by LinCom Corporation

The objective of the trajectory control requirements study was to determine preliminary operational requirements for trajectory control during automated rendezvous and docking in Earth and Mars orbits. Preliminary requirements for launch windows, navigation (both absolute as well as relative), guidance, targeting, control,and sensor performance have been formulated.

For this study, the MRSR and the Satellite Servicer System (SSS) missions were selected as representative scenarios. The reasons for selecting the SSS missions were

1. SSS can be considered an incremental testbed for many AR&D technologies and activities. Hardware- and software-related items could be tested during these missions.

2. There may be several SSS missions before the MRSR or lunar base operations can be undertaken. Thus, the AR&D technology could be matured at a faster rate when applied to SSS missions.

3 Cost effectiveness and significant cost savings could be realized earlier due to higher frequency of servicing missions.

Efforts were not concentrated on lunar base operations for two reasons: 1) limited resources are available for this effort, and 2) most lunar AR&D requirements could be derived using MRSR and SSS AR&D results. At this time, no special requirements related to lunar base operations are envisioned.

6.3 MISSION SCENARIO

6.3.1 MRSR Missions

The MRSR entails four mission scenarios: Local D, Areal B, Areal D, and Areal B-Heavy, which are described briefly in the following paragraphs. AR&D system requirements are similar for all the scenarios.

The MRSR Local D reference mission consists of two flights (i.e., D1 and D2). Both flights will be launched by an upgraded Titan IV with the Centaur G' as an upper stage. Flight D1 will consist of the Rover and Mars Ascent Vehicle (MAV). Flight D2 consists of the Mapping/ Communication Orbiter (MCO) and Sample Return Orbiter (SRO). These flights are expected to be launched around the year 2000.

Upon arrival into the vicinity of Mars, both the MCO and SRO will be placed in the proper parking orbit. Flight D1 will de-orbit the MAV and Rover and place them on the Mars surface. Surface activity will include sample collection by the Rover, collection of a contingency sample and deployment of a meteorological/geophysical science package by the lander. After the surface activity is completed and samples are properly stowed, the MAV will launch and orbit the planet. The SRO will then rendezvous and dock with the MAV. Once docking is complete, the Sample Canister Assembly (SCA) will be transferred to the Sample Return Capsule (SRC) of the Earth Return Vehicle (ERV) portion of the SRO. The SRC will be separated from the ERV shortly before the ERV's closest approach to Earth. The SRC will be aero-captured into a circular orbit of the Earth, and the Space Shuttle will retrieve it at a later time.

The difference among the four mission scenarios is how the Rover and MAV reach the Mars surface. In terms of AR&D, once the Rover and MAV are on the Mars surface, the rest of the mission segments (i.e., sample collection, MAV launch, SRO rendezvous and docking and sample return to Earth) are identical. For each mission, the SRO and MCO remain in the martian parking orbit while the Rover and MAV land on the surface. MRSR missions and their launch configurations are summarized below:

Mission	Flight 1	Flight 2
Local D	Rover and MAV	MCO and SRO
Areal B	SRO and MAV	MCO and Rover
Areal D	Rover and MAV	MCO and SRO
Areal B-Heavy	SRO and MAV	MCO and Rover

There are two critical segments of the MRSR missions: launch of the MAV and rendezvous and docking of the SRO with the MAV. Assistance from the ground is not available in a timely manner. Therefore, the AR&D System must work in a fail-safe, fail-operational condition. Thus, the MRSR mission is very important to the derivation of AR&D System requirements.

6.3.2 Satellite Servicer System

The SSS will involve three missions: Flight Demonstration One, Two, and Three. Requirements for the SSS have been derived from the first flight demonstration. A brief description of each of the missions follows.

6.3.2.1 Flight Demonstration One (DF1)

The objective of DF1 will be to demonstrate the AR&D and proximity operations capability required of a servicing system that will provide a logistics resupply for Space Station Freedom. DF1 will consist of the OMV, a target and the Orbiter.

The Orbiter will deploy the OMV and target in docked configuration. The OMV and target will be positioned from the Orbiter at a safe distance. A fly-around the Orbiter will occur so that sensor evaluations can be performed. The OMV and target vehicle will perform proximity operations in radius of 500 meters in and out of plane. The Orbiter will be positioned in an observation attitude. The OMV will then release the target and move one mile away from the target. The OMV will then perform Terminal Phase Initial and Terminal Phase Final approaches and docking.

The OMV will maintain a safe distance from the Orbiter and release the target. The OMV will then move approximately 100 miles away from the target. From that position, the OMV will search the target, perform Terminal Phase Initial and Terminal Phase Final approaches and docking operations. The OMV will then bring the target to the Orbiter and perform a slow approach to the Orbiter payload bay. Station keeping will be performed at about 30 feet distance. The Orbiter will retrieve the OMV and target and stow them in the payload bay.

6.3.2.2 Flight Demonstration Two (DF2)

The second flight demonstration will be conducted in the Orbiter payload bay and will prove the capability to perform satellite servicing functions. Orbital Replaceable Unit (ORU) exchange will take place and the integrity of the fluid transfer interface will be verified. DF2 will involve the Orbiter, OMV and a target.

The initial demonstration activity will be an on-orbit supervised autonomous interfacing of the electrical and mechanical connections. A validation of servicing interfaces will be performed. Refueling activities will be performed as will mating and demating of fluid connectors with subsequent preservation and leak checking to validate the reliability of the interfaces. Several ORU's of different shapes and sizes will be exchanged.

6.3.2.3 Flight Demonstration Three (DF3)

DF3 will combine both DF1 and DF2 and will involve autonomous rendezvous and docking; ORU exchange; fluid transfer; and mating and demating of mechanical, electrical and fluid interfaces. This demonstration will include use of the Orbiter, a target vehicle and the OMV and related hardware.

The Orbiter will first deploy the OMV and move 1000 feet away and then deploy the target vehicle. The Orbiter will remain within 500 feet of the target to facilitate observations. The OMV will move away approximately 1 mile from the target and perform autonomous rendezvous and docking. The OMV will then undock the target and move 100 plus miles away from the target. Rendezvous and docking will be repeated.

The OMV will remain docked with the target. Four different sizes and types of ORUs will be exchanged. Refueling will be performed and hydrazine fuel will be transferred from the OMV to the target vehicle. The OMV will then release the target and move away from it. The Orbiter will retrieve the target vehicle and then the OMV and stow them in the payload bay.

6.3.3 Lunar Base Operations

As mentioned earlier, the AR&D requirements for lunar base operations can be derived from the MRSR and SSS missions. However, three Space Station-based scenarios are briefly described in the following paragraphs.

6.3.3.1 Mission Scenario One

This mission consists of two Orbit Transfer Vehicles (OTV), the Manned Lunar Module (MLM), an expendable lander and launcher. One of the OTVs will provide for translunar injection of the stack that consists of the other OTV, the MLM, and expendable lander and launcher. The second OTV will provide for lunar orbit insertion of the stack (i.e., MLM, lander and launcher) and will also allow the MLM to leave lunar orbit and re-establish Earth orbit on the return flight. The MLM and lander/launcher will descend to the lunar surface while the second OTV stays in orbit and waits for the MLM return. The MLM/launcher will ascend from the lunar surface, rendezvous and dock with the second OTV. The MLM and OTV will return to the Space Station.

6.3.3.2 Mission Scenario Two

The second mission scenario involves two OTV's, a cargo module, and an expendable lander/launcher. The first OTV will be used to provide translunar injection of the stack (second OTV, cargo module, expendable lander/launcher) and return to Space Station. The second OTV will insert the cargo module and the expendable lander/launcher into lunar orbit. The lander will place the cargo module on the lunar surface. The second OTV will return to Earth orbit, circularizing above the Space Station orbit, and perform rendezvous and docking with the OMV. The OTV will be returned to the Space Station by the OMV.

6.3.3.3 Mission Scenario Three

In this scenario, a Lunar Orbit Support Facility (LOSF) will be placed in lunar orbit in a manner similar to the Space Station's Earth orbit. An OTV will provide for the LOSF lunar orbit insertion. Another OTV will provide translunar injection to the stack (consisting of one OTV, the MLM, and hydrogen cargo) that will go to the LOSF. A reusable lunar ascent/descent stage will land on the lunar surface and will later ascend and dock with the LOSF. The LOSF will support exchange of the crew and the hydrogen cargo. For the Earth-bound crew, an OTV will provide trans-Earth injection of the OTV/MLM. The OTV/MLM will aero-break in Earth-orbit and circularize above the Space Station orbit. Later, the OTV/MLM will dock with the Space Station.

6.4 APPROACH

There are several studies and reviews in progress that will help develop various mission characteristics and AR&D technology. Our study focused on trajectory control requirements with activities in the areas of sensor performance requirements, navigation, guidance, control, targeting, docking performance and mission planning.

Our approach was divided into three segments: 1) to"brainstorm" the overall mission and establish guidelines, 2) to review and study past and present techniques including the baseline OMV, and 3) to study each area in detail.

We studied rendezvous and docking scenarios and techniques used for the Space Shuttle, Apollo, and OMV missions. We also reviewed the techniques used for Skylab and Soviet missions. Overall assessments of mission models were consolidated and failure scenarios and recovery techniques were discussed. New sensor development and upgrades were reported and incorporated in the guidelines. Reviews of operational procedures were conducted and accompanying rationales were studied in terms of requirements.

Advantage was taken of results from an MRSR Program Phase A midterm review which included discussion of all aspects of the mission. As part of this review, there were presentations on software as well as hardware components of the system and results from guidance and navigation studies. Laser docking sensor development status was presented and plans to use it in SSS missions were revealed. There were concerns about lack of available environmental data on the martian atmosphere, dust storms, rock sizes, and topography. There were presentations on various sensors in general and related measurement errors.

As part of our study, basic equations of nodal regressions, catch-up rates or phasing, orbital periods, out-of-plane delta-v, Hohmann transfers and related parameters were entered into a Lotus123 spreadsheet application to compute preliminary estimates of overall delta-v's for the missions. Reference altitude for the parking orbit was maintained constant and coast altitudes for the target vehicle were varied to optimize the timing in orbit and minimize the required delta-v's. Plots using Lotus123 were generated to provide insight about the interrelationships between in-plane phasing, nodal regression, parking orbit for the SRO and launch altitude for the MAV. This information was important in generating concepts for MAV launch windows that will minimize delta-v penalties.

CW relative motion equations were also implemented in Lotus and preliminary trajectories for SSS separation were generated. The target separation distances of 300 and 5000 feet as defined in the SSS mission were used. Optimal delta-v's were generated where the OMV would return to the release point in about 90 minutes.

Mars gravity was modeled in the 6-DOF guidance, navigation and control (GN&C) orbital operations analysis simulator and parking orbits were studied. How the behavior of eccentricity and other orbital parameters affected the rendezvous considerations was then studied.

Docking tolerances were calculated for the Three Point Docking Mechanism and Remote Manipulator System Grapple-fixture Docking Mechanism based on relative geometry and dimensions. The maximum and desirable velocities and angular rates were also calculated.

6.5 MRSR RESULTS

In this section, our results for the MRSR mission are described. The MAV's ascent and SRO's rendezvous and docking with the MAV are identical in each of four mission profiles as noted in section 6.3. Therefore, our results are applicable to all MRSR mission profiles. In section 6.5.1, our assumptions and mission guidelines regarding the vehicle characteristics are described. In section 6.5.2, preliminary results on parking orbit behavior are described. In section 6.5.3, major subsystem requirements are detailed and their interrelationships are specified as appropriate.

6.5.1 Mission Model Guidelines

The MAV will be a fully active vehicle during the ascent phase as well as the coasting phase where it will be waiting for the SRO to perform rendezvous and docking. There will be an on-board computer for data processing and GN&C algorithms. The MAV will be a cooperative target that helps relative navigation determination. The control system will have attitude hold and maneuver capability. Navigation software will propagate the inertial state using measurements from the Inertial Measurement Unit (IMU) or gyros and accelerometers. Various sensors, e.g., Sun and Earth sensors, will help to align the IMUs periodically. The MAV is assumed to have full or partial communication with the SRO and the MCO, and will be able to provide its inertial position and velocity at regular time intervals. The SRO will use this information to point its relative sensor in the "right" direction and find the MAV promptly. The MAV is assumed to have proper docking fixtures which are compatible with the SRO's docking mechanisms. Since the MAV is fully active, it can also perform final approach to a close distance in the event docking efforts by the SRO fail or the SRO is running low on fuel.

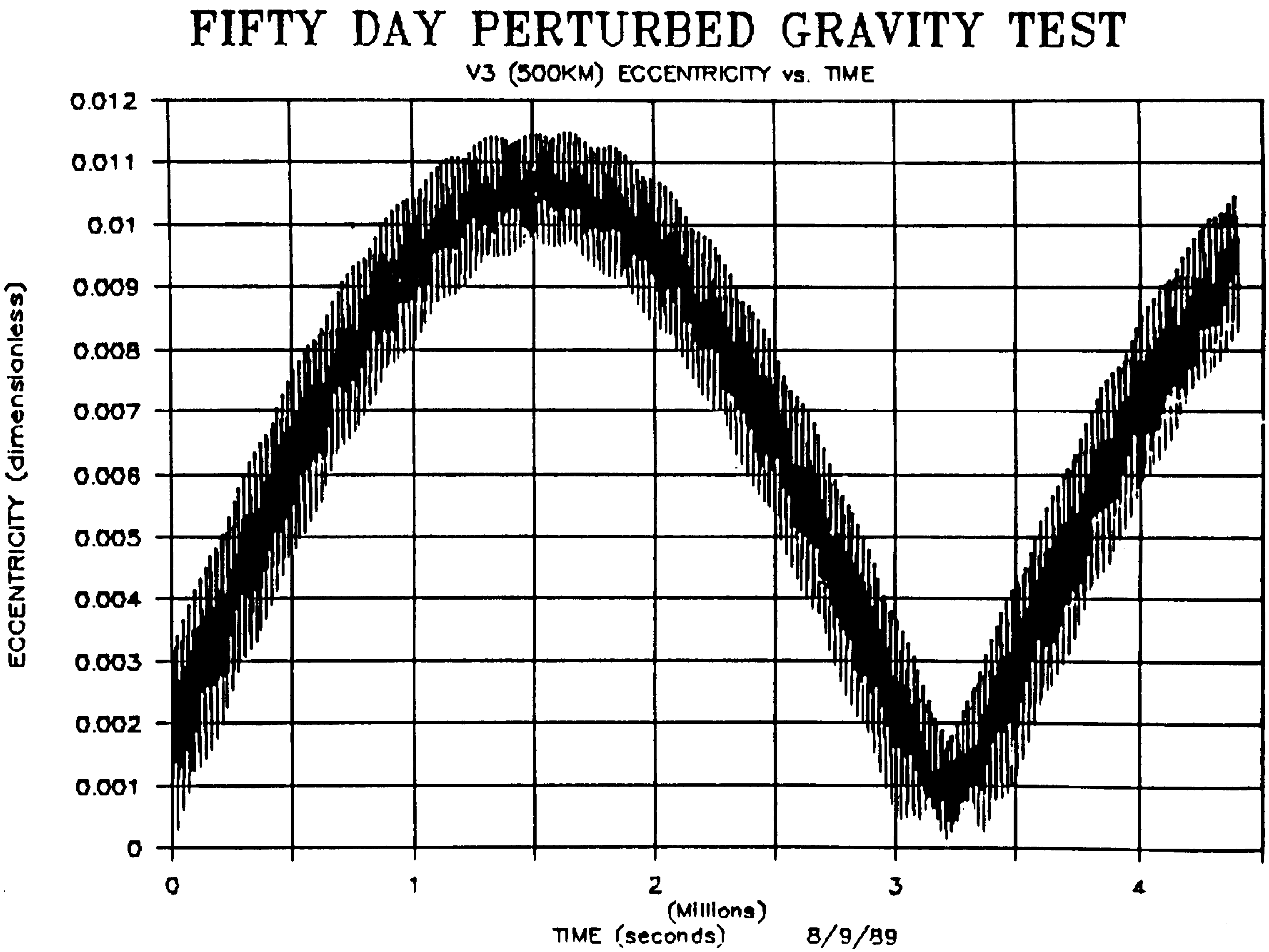
The SRO is assumed to have docking mechanisms compatible with the MAV's. It will also have full 6-DOF control and various sensors like IMU, Sun and Earth sensors, star tracker, and a laser docking sensor. The SRO will be equipped with full GN&C software as well as mission planning software. The SRO will continuously communicate with the MAV during its ascent as well as its coasting (or waiting) period. There will be a fully redundant propulsion system for long-range rendezvous or orbit transfer, terminal rendezvous, final approach and docking operations. The MCO's will be used to communicate with the ground when the SRO cannot communicate directly.

6.5.2 Trajectory Analysis for the MAV Ascent and Rendezvous Phasing

Since the SRO will be parked in a 500 km orbit prior to the MAV launch, it is necessary to find out the effects of Mars gravity on this parking orbit. Mars gravity with J2 and J3 perturbation terms was modeled in a 6-DOF orbital operations analysis simulator, and the SRO state in 500 km circular orbit was propagated for 50 days. The fourth order Runge-Kutta integrator was used with a time step of 10 seconds and the data were output at one-hour intervals. The SRO orbit exhibits oscillations in the eccentricity with a period of about 35 days and peak value of 0.011 as shown in figure 6-1. At the peak eccentricity, the orbit is nearly 530 x 450 km (fig. 6-2), thus making the SRO in-plane phasing very difficult. A detailed study is required to investigate the in-plane phasing with eccentricity variations and differential nodal regression effects.

An in-plane catch-up rate for a given delta height was calculated by assuming that the SRO is in 500 km and 450 km orbits,. As shown in figure 6-3, the MAV can reduce only certain amounts of the phase angle in a day. If the MAV has to coast for a longer time in its orbit to achieve optimal phasing for long-range rendezvous, differential nodal regression will force the two orbital planes to separate. The SRO will have to correct for this plane change. Currently, it is planned that the SRO will carry only 50 m/s delta-v as a reserve. As shown in figure. 6-4, the differential nodal rate is significant for large delta heights. At 500 km altitude, the SRO can correct for only 0.87 deg nodal angle (fig. 6-5 and 6-6) with the above reserve delta-v. The MAV launch must be properly timed in order to fulfill these constraints.

It is recommended that the SRO be maneuvered into a phase-repeating orbit prior to the MAV launch. A communication link should be established before launch and maintained till docking so that the SRO will not loose track of the MAV. Trajectory for ascent and long-range rendezvous should be tuned together as one segment. The SRO can then wait for a desired length of time before it performs terminal rendezvous and docking. The MAV has to launch in plane or be able to correct for any out-of-plane deviations. Further trajectory analysis is required to fully understand the impact of nodal regression, in-plane phasing and eccentricity variations.



FIFTY DAY PERTURBED GRAVITY TEST
V3 (500KM) ECCENTRICITY vs. TIME
ECCENTRICITY (dimensionless)
0.012
0.011
0.01
0.009
0.008
0.007
0.006
0.005
0.004
0.003
0.002
0.001
0
0
1
2
3
4
(Millions)
TIME (seconds)
8/9/89

Figure 6-1

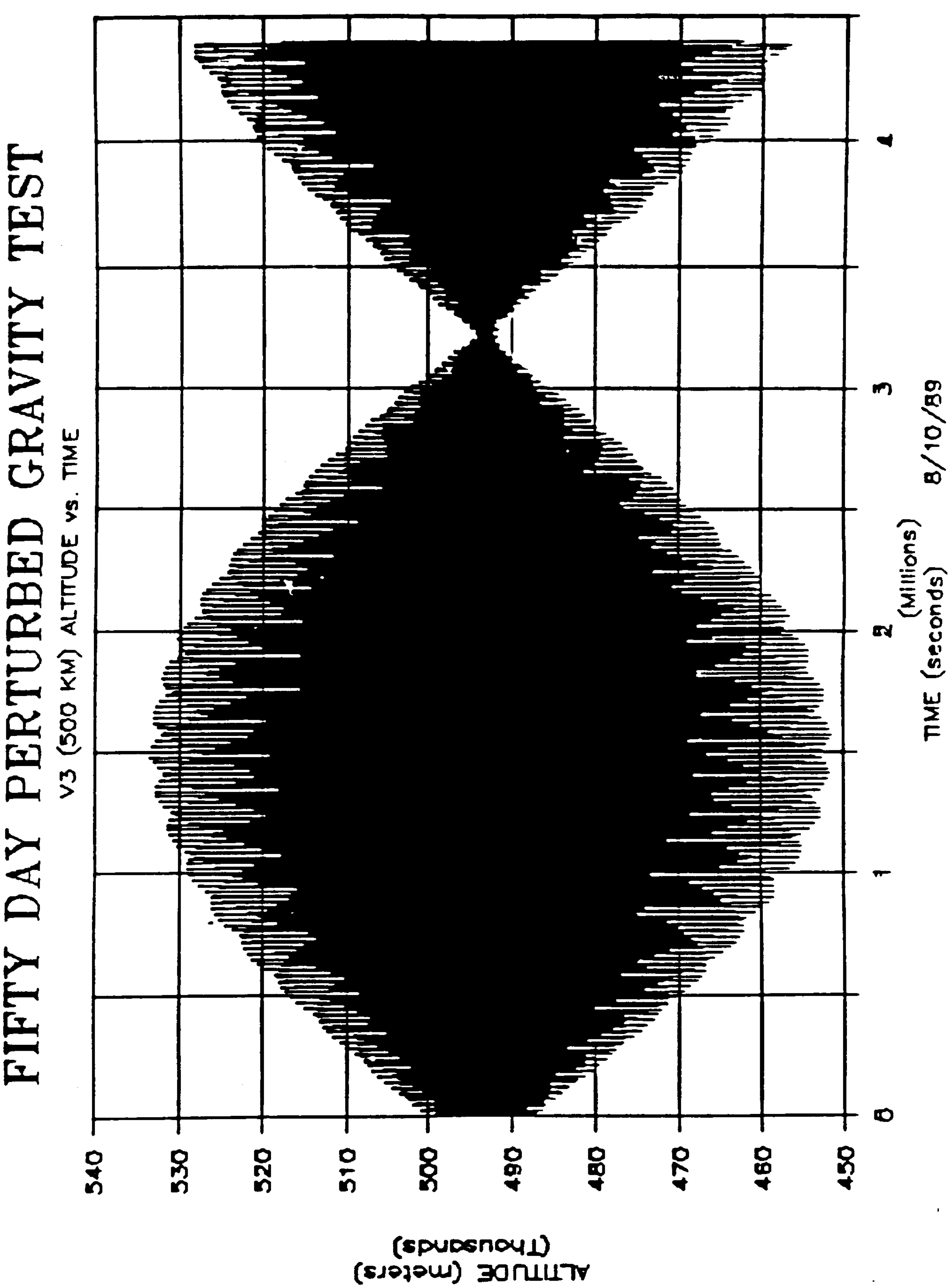
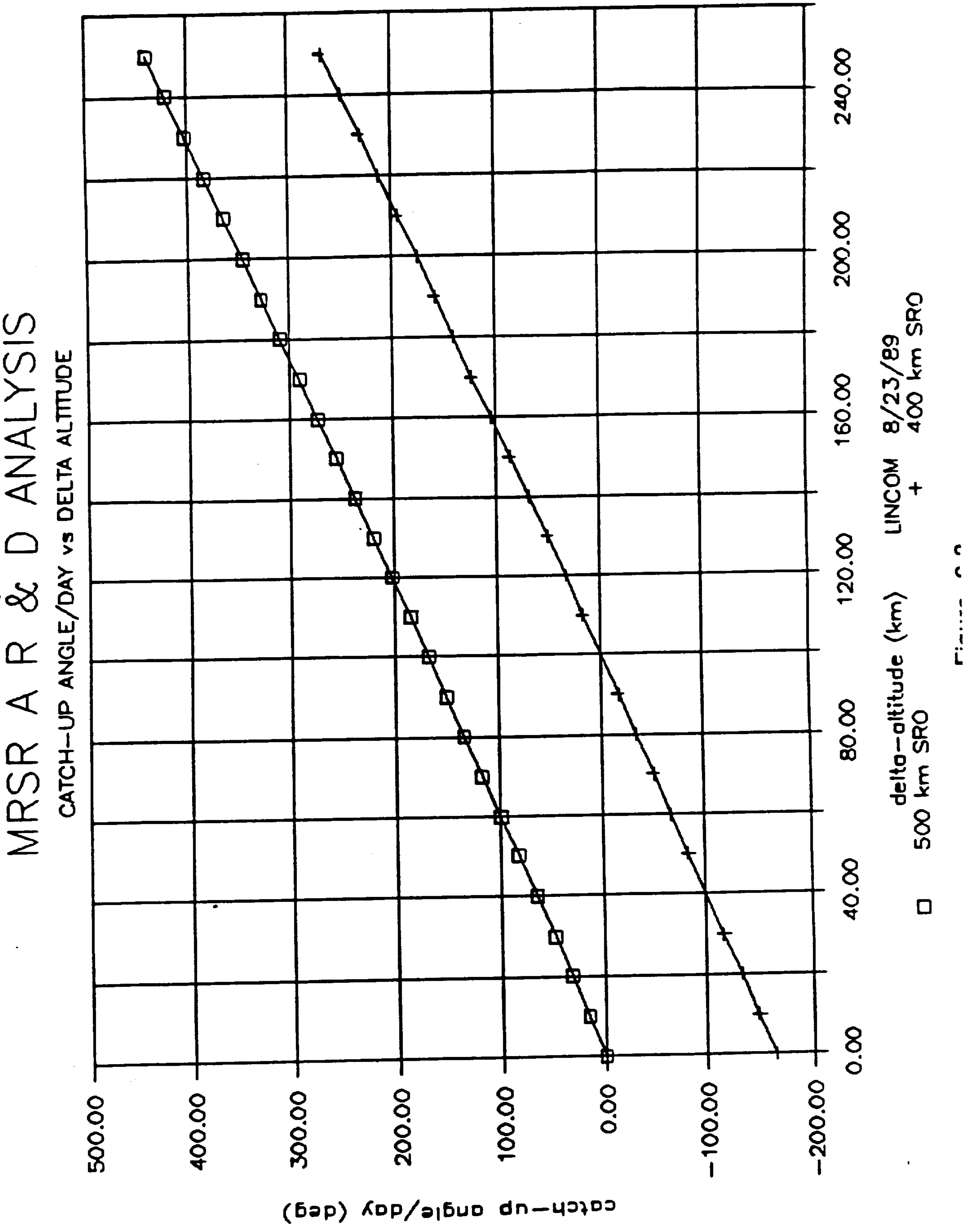
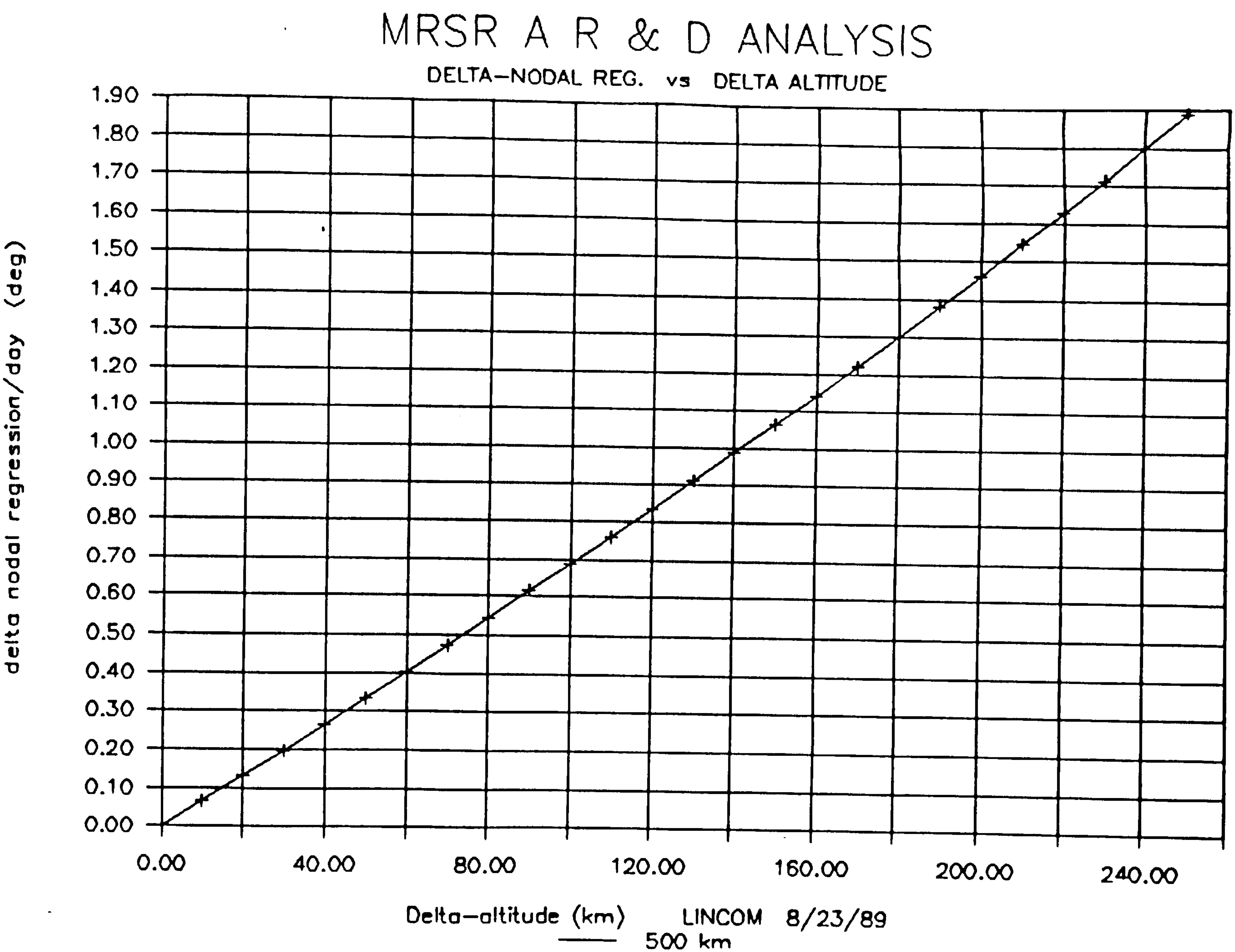

FIFTY DAY PERTURBED GRAVITY TEST
V3 (500 KM) ALTITUDE vs. TIME
ALTITUDE (meters)
(Thousands)
540
530
520
510
500
490
480
470
460
450
0
1
2
3
4
TIME (seconds)
(Millions)
8/10/89

Figure 6-2


MRSR A R & D ANALYSIS
CATCH-UP ANGLE/DAY vs DELTA ALTITUDE
LINCOM 8/23/89
delta-altitude (km)
□ 500 km SRO
+ 400 km SRO
catch-up angle/day (deg)
500.00
400.00
300.00
200.00
100.00
0.00
-100.00
-200.00
0.00
40.00
80.00
120.00
160.00
200.00
240.00



MRSR A R & D ANALYSIS
DELTA-NODAL REG. vs DELTA ALTITUDE
1.90
1.80
1.70
1.60
1.50
1.40
1.30
1.20
1.10
1.00
0.90
0.80
0.70
0.60
0.50
0.40
0.30
0.20
0.10
0.00
delta nodal regression/day (deg)
0.00
40.00
80.00
120.00
160.00
200.00
240.00
Delta-altitude (km)
LINCOM 8/23/89
500 km

Figure 6-4

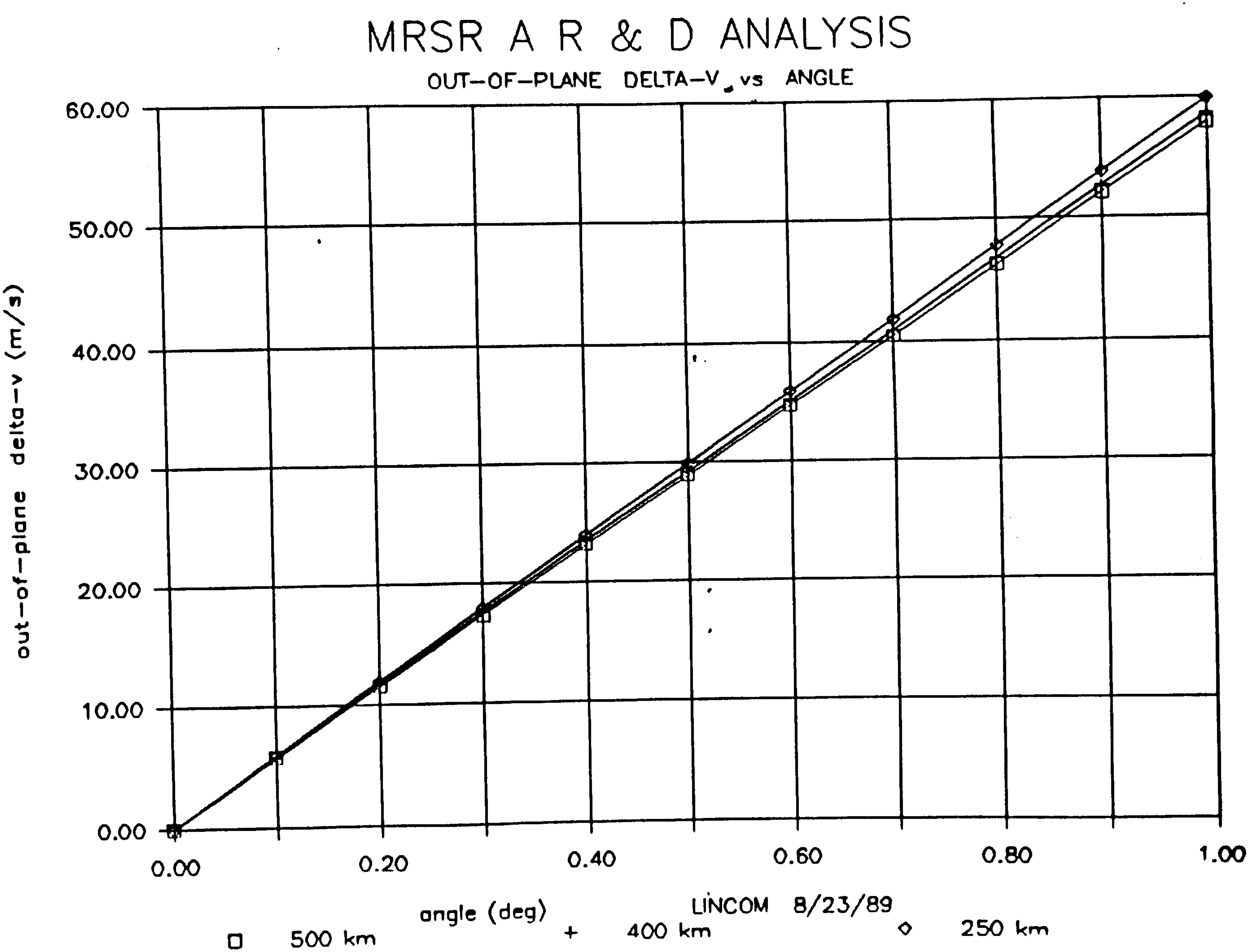
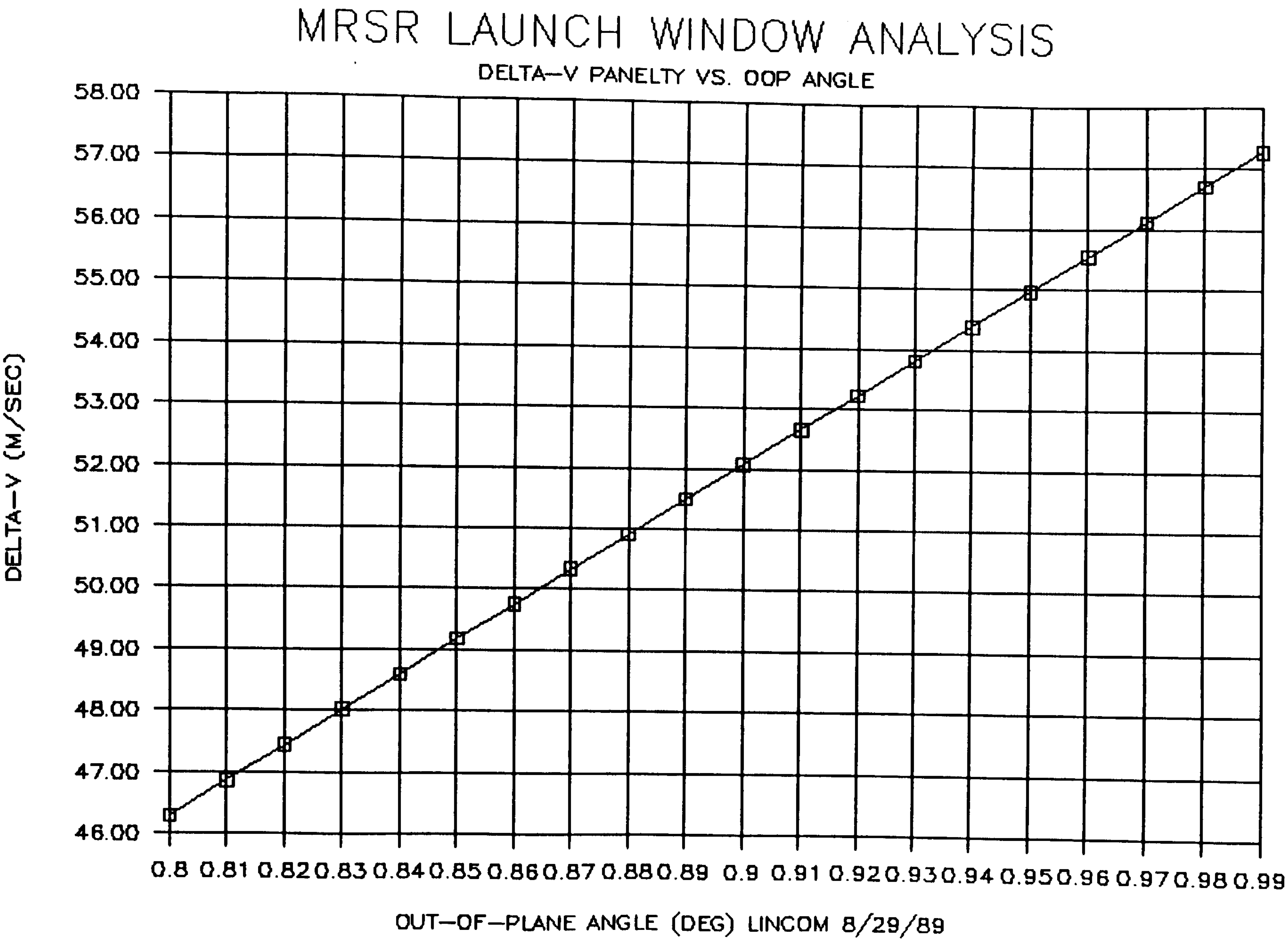

MRSR A R & D ANALYSIS
OUT-OF-PLANE DELTA-V vs ANGLE
60.00
50.00
40.00
30.00
20.00
10.00
0.00
out-of-plane delta-v (m/s)
0.00
0.20
0.40
0.60
0.80
1.00
angle (deg)
LINCOM 8/23/89
500 km
400 km
250 km

Figure 6-5



MRSR LAUNCH WINDOW ANALYSIS
DELTA–V PANELTY VS. OOP ANGLE
58.00
57.00
56.00
55.00
54.00
53.00
52.00
51.00
50.00
49.00
48.00
47.00
46.00
DELTA–V (M/SEC)
0.8 0.81 0.82 0.83 0.84 0.85 0.86 0.87 0.88 0.89 0.9 0.91 0.92 0.93 0.94 0.95 0.96 0.97 0.98 0.99
OUT–OF–PLANE ANGLE (DEG) LINCOM 8/29/89

Figure 6-6

6.5.3 <u>Study Areas</u>

6.5.3.1 Navigation and Sensor Performance

Sensors will provide measurement of non-gravitational acceleration and body rates. Absolute navigation software will propagate the inertial state and the attitude information. There will be a need to periodically incorporate measurements from the star tracker to realign IMU's and correct for drift effects. Other inertial measurements like GPS, land mark navigation, TDRSS tracking or MCO tracking will improve covariance data. If no such measurement is available, the state estimates will become inaccurate. The behavior of covariance data will require future study.

A long-range relative sensor will be required to improve our knowledge of the target state. Cooperative sensors like communication beacons are especially helpful for this purpose. For passive targets without communication capability, long-range radar will be useful. Long-range laser sensors, currently in development, can also be used if the target is equipped with retroreflectors. A medium-range relative sensor with much higher accuracy in measurements will be required to perform accurate rendezvous burns. Note that the target attitude information is not required for these burns. Pointing requirements are O.5x in attitude during burns. Accuracy in range measurement is 2 km at 150 km range, 0.3 km at 30 km range, and 0.5 m at 100 m range. A short-range (100 m to 0.005 m) sensor will be required for range, range rate and relative attitude determination. The laser docking sensor will provide such a capability. Determination of target attitude using a vision sensor has been demonstrated by several research groups including those at Rice Univeristy, Purdue, and the Jet Propulsion Laboratory. There are several algorithms available with varying degrees of accuracy.

6.5.3.2 Guidance and Control

The guidance system is required to assess inertial and relative states and the vehicle orientation. The targeting function is then used to generate the correction delta-v's for the future desired position provided by the mission planning element. Based on the correction delta-v's, pointing commands in terms of inertial and Local Vertical Local Horizontal (LVLH) attitude are generated. Delta-v's and attitude commands are provided to the control system for proper execution. The control system performance is monitored through sensors, and the navigated states are continuously checked.

If the long-range rendezvous (orbit transfer) trajectory is deviating from the desired/planned trajectory, then it will generate appropriate mid-course corrections via targeting software.

During proximity and docking operations, 6-DOF guidance will be required, and delta-v's will be generated using the sensor measurements directly (rather than through targeting). The guidance system must be capable of performing fly-around for inspection, final approach, and soft docking.
A 4-DOF guidance with proper pointing capability will be required during the rendezvous phase so that the relative sensor can continuously track (closed-loop

steering) the target and improve the relative state information. The guidance system will compute the attitude required for proper delta-v, and will also generate relative sensor orientation commands, such as pointing of the radar antenna, laser beam or high gain antenna.

The control system must be capable of maintaining attitude hold in the inertial as well as LVLH frame. Commanded angles and the reference frame will be specified by the guidance system. The flexibility of the control system enables it to receive the deadband values from the guidance system or generate these values based on the mode indicator. Typically, the control system is expected to maintain angles within a degree of accuracy and correct the pointing errors with a rate less then 0.1 deg/sec. The control system must also perform attitude maneuvers at a desired rate and should have the capability to maintain a rate within 0.01 deg/sec. The maximum rate for attitude maneuvers is expected to be within 2-5 deg/sec. There should be a rate-hold capability independent of maneuver rate capability.

The control system must be able to provide 0.03 m/s delta-v accuracy during the long-range as well as terminal rendezvous burns. These burns should be performed in 4-DOF and/or 6-DOF mode with proper commands to appropriate jets. The capability to perform these burns simultaneously with attitude hold is required.

During the final approach and docking operations, a full 6-DOF authority is required to translate and rotate the vehicle in all directions. Capability to correct the relative velocity within 0.003 m/s magnitude and maintain relative orientation within 0.5 deg is needed. For proper control authority relative attitude should be corrected with a less than 0.01 deg/sec torque impulse.

6.5.3.3 Targeting

Targeting software will be required to compute the correction delta-v's for all three zones; long-range rendezvous (orbit transfers), terminal rendezvous and the final approach. Since the target orbit could be elliptical during first zone (or the end-point from the ascent circularization burn), the targeting must be capable of generating delta-v's for co-elliptic transfer with a given time of flight. The most common algorithms, like Hohmann and Lambert, are based on the impulsive transfer, meaning instantaneous delta-v. Since all burns have finite time, there is an error in the delta-v which should be corrected using mid-course corrections. There are algorithms like BVL, CW, and iterative Lambert and CW algorithms that correct for the finite burn effects. The on-board targeting software will require some of these algorithms and utilities such as a precision extrapolator and phasing calculation.

6.5.3.4 Propulsion System

The trajectory is divided into three regions: long-range rendezvous (from 150+ km to 30 km), terminal rendezvous (from 30km to 100 m), and final approach and docking (from 100 m to docked state). The orbit maneuvers present a range of delta-v's for each region. The long-range rendezvous delta-v's are in the range of 15 - 20 m/s with a pointing accuracy within 1-2 degrees. During terminal rendezvous, the range of delta-v's is from 1 m/s to 15 m/s with a pointing accuracy of better than one degree.

The delta-v's for final approach and docking are in the range of 0.02 m/s to 1 m/s with a pointing accuracy of 0.5 degree in relative attitude.

To satisfy the above accuracy requirements, three types of thrusters are envisioned at this time. The first set, called large thrusters, have 500+ N thrusting capability. Nominal thrust value is determined by the mass of the vehicle. (Typically, the acceleration due to these thrusters is around 0.2 g's.) Minimum delta-v's due to these thrusters should be in the range of 0.05 m/s. This provides the minimum on-time requirements. These thrusters could be fixed or gimballed depending on the attitude hold capability of the second set of thrusters.

The second set of thrusters, called medium thrusters, have a thrusting capability in the range of 50 to 150 N. These thrusters provide the attitude hold capability (pointing accuracy) during long-range rendezvous maneuvers and delta-v's during the terminal rendezvous maneuvers. Minimum delta-v's should be in the range of 0.01 to 0.03 m/s and typical full thrust acceleration is around 0.1 g. The angular torque capability must conform to the control requirements. This set can also be used for attitude hold during terminal rendezvous. If necessary, these thrusters can be used simultaneously with large thrusters to achieve accurate delta-v's or as a backup to large thrusters.

The third set of thrusters, called small thrusters, have a thrusting capability in the range of 5 to 20 N. These thrusters are used for final approach and docking operations. Minimum impulse should be in the range of 0.001 to 0.003 m/s with a torque impulse capability in the range of 0.001 to 0.01 deg/sec. Thruster arrangement of this set must provide complete 6 DOF with minimum cross-coupling between translation and rotation pulsing. If required, these thrusters are also used for attitude hold during terminal rendezvous operations.

Thus, there are three types of thrusters providing a full range of capabilities as well as redundancy for the MRSR system. From an AR&D point of view, a propulsion system must have certain characteristics because of required accuracy in controlling relative delta-v's and attitudes. If an existing vehicle's propulsion system does not meet these requirements, it will be technically difficult and costly to upgrade that vehicle for autonomous rendezvous and docking.

6.5.3.5 Docking Performance

There are two docking systems in use at this time: a three-point docking mechanism (TPDM) and RMS grapple-fixture docking mechanism (RGDM). Each system has a mechanism which is an active part that closes for rigidization and a fixture on the target which is grappled by this mechanism. For simplicity, we refer to these two parts as mechanism and fixture. The accuracy of these two systems is described in section 6.6.2.6. A system like RGDM is proposed for MRSR because of its higher tolerances and flexibility in handling the operations.

6.5.3.6 Mission Planning Element

The MRSR system is required to have the mission planning element as part of all autonomous unmanned vehicles. The mission timeline will be planned by the mission planner to perform rendezvous and docking operations. This system will receive its input from absolute and relative navigation and will output the information for the guidance (and targeting) function which typically generates the correction delta-v's and pointing vector.

This element must plan the mission appropriately when estimates of the MAV's inertial state are provided by the absolute and relative navigation functions. It must compute 1) time of acquisition, 2) appropriate time to coast in the given orbit to null out the differential nodal regression and in-plane phasing, 3) time to perform the first orbit transfer burn, and 4) time to perform the final burn that achieves the orbit with the same eccentricity as that of the MAV. The last burn should put the SRO in the desired position for the terminal rendezvous initiation. The mission planning element must be capable of generating a mission timeline with two or more burns as required. The capability to replan the mission in case of failures, and/or change mission objectives is required. Methods to monitor the progress of the mission and current status of the vehicle are also needed.

One of the characteristics of this mission planning element is the ability to plan the terminal rendezvous, final approach and docking, and determine the time estimates of these events. Initially, the mission planning element may work from a standard rendezvous profile; however, as it evolves and matures, it must be capable of planning/replanning from start to end. The mission monitoring activity can be embedded here or can be a separate element. From a systems point of view, it is assumed that the same element will perform both functions.

6.5.3.7 Communications and Tracking

For the MRSR mission, active communication between the SRO and MAV is a must for two reasons: 1) to have a long-range navigation capability, and 2) because the SRO must command the MAV thruster to stop firing for attitude corrections while it is capturing the MAV. Otherwise, if the MAV thrusters fire during docking, the impulse will create a failed docking. The same situation applies during the SSS missions.

6.5.3.8 Data Processing and Data Management

The on-board data processing system will be required to have sufficient computing resources to 1) process measurements, 2) compute critical parameters and commands by executing all GN&C algorithms, 3) communicate with other systems without interruptions, and 4) handle the rule bases as well as knowledge bases used for this mission planning. At this time it is envisioned that the computing power in the range of 5-10 MIPS will be sufficient to fulfill all these requirements and generate all mission planning data. The mission data for long-range rendezvous (from 150+ km to 30 km) will be generated initially and checked once every 30 seconds. For the terminal rendezvous segment, the mission data will be checked every 2-3 seconds. During final approach and docking, mission monitoring will be required continuously, i.e., twice a second, and decisions will be made to proceed or abandon the activity based on the sensor data. Requirements for exact computing resources will be derived from this type of requirement for the mission planning element.

6.6 SSS SCENARIO

Three SSS scenarios are described in section 6.3. For AR&D requirements, we have used the first mission as it relates closely to rendezvous and docking. The second mission is essentially an autonomous ORU exchange procedure which is not incorporated in AR&D requirements at this time.

6.6.1 Mission Model Guidelines

Mission guidelines were generated for DF1 SSS mission. There are two active vehicles for this mission: the OMV, which will perform all activities, and the Orbiter, which will simply function as a supervisor. The target is a small payload. The results, as described below, mainly pertain to the OMV, target, new sensor and mission profile.

6.6.1.1 The OMV

OMV capabilities as they relate to the SSS flight demonstration are described here. The baseline OMV (fig. 6-7) has a propulsion system with three types of engines. There are IMU's, Sun sensor, horizon sensor, radar, video camera, HGA for communication, Global Positioning System (GPS) Receiver and other appropriate hardware and software. Orbit transfer maneuvers are performed by the OMV in an automated mode using the mission timeline provided from the ground-based and on-board software. The terminal rendezvous phase is monitored on the ground but is essentially in automated mode with a manual over-ride capability. From 1000 ft. to dock, the OMV is remotely piloted from the ground using video cameras and radar inputs.

Thus, the OMV in its current state is not capable of performing an AR&D mission as described in the first SSS scenario. Definite upgrades will be required as described in the following sections.

6.6.1.2 Target

The use of a a cooperative target with certain capabilities is assumed. The target will be a small cylinder or symmetric object with retroreflectors mounted properly on the surface so that the OMV docking sensor can use them in a relative attitude determination. The target will have a control system with attitude hold capability. There will be a communications beacon on the target to aid in long-range search by the OMV, and some communication with the Orbiter. The target will have the necessary docking fixture to match the docking mechanism on the OMV.

6.6.1.3 Laser Sensor

The laser docking sensor, under development in the JSC Engineering Directorate, is assumed to be available for automated docking operations. This sensor provides range, range rate, and bearing angles and rates fairly accurately. It also provides the relative attitude of the target within a 0.3 deg accuracy, and the relative rates within a 0.01 deg/sec. The attitude and angular rates are measured for a range of less than 30 meters. The laser sensor operates by sending out laser beams that sense the reflected beams. Four retroreflectors placed on the target will reflect four beams and the total time of flight for each of the four reflectors will be different. Using these travel times, the sensor computes the relative attitude of the target. However, it needs only one reflector to compute the range and range rate for the target.

6.6.1.4 Sequence of Operation

The overall SSS mission scenario for DF1 from the pre-proposal briefing is shown in fig. 6-8. There are several steps in the entire sequence of operations as described below.

1. The Orbiter will deploy both the OMV and target together with the axis on the Orbiter's V-bar. The Orbiter will then place itself at a safe (TBD) distance from the OMV and target.

2 When, in a docked configuration, the OMV and target will fly around the Orbiter at 100 feet distance to simulate proximity operations. Proper attitude is required for the inspection task and a full 6-DOF guidance capability will be maintained. Small jets will be used as appropriate.

3. The OMV and target will separate with a small relative velocity so that the target does not drift away from the Orbiter.

4. The OMV will use its small jets to move about 300 feet away from the target to evaluate its laser docking sensor. The impulse will be such that it will return back to the release point in about 90 minutes (fig. 6-9).

5. As the OMV moves away and back to the target it will evaluate its laser docking sensor. Then, it will dock with the target to complete the docking evaluation. It will perform proper station-keeping at a safe distance, if necessary, and use all its closed-loop guidance algorithms during its final approach and docking operations.

6. The OMV will release the target and slowly move about 1 mile away. There will be a station-keeping phase at this distance, utilizing its closed-loop guidance system. The terminal rendezvous profile will be planned by the mission planner. The OMV will use its targeting algorithms to generate the necessary delta-v's and also generate appropriate pointing commands. This will be relayed to the Orbiter and a confirmation will be requested. The algorithmic evaluation will then be complete.

7. The OMV will perform the terminal rendezvous with the target and demonstrate autonomous rendezvous techniques. Evaluation of its relative sensor, relative navigation software and mission monitoring capabilities will be performed. The OMV will then, continue its final approach and docking depending upon the real-time progress and status of the mission. This will demonstrate full AR&D capability.

8. The OMV will repeat steps 5 & 6 by moving 100 miles away from the target. This segment of the mission will demonstrate full AR&D capability from 100+ miles. The long-range relative sensor will be used to detect the target and improve knowledge of the target's relative state. Mission planning and mission monitoring capabilities will be demonstrated during this long-range rendezvous. Appropriate jets and 4-DOF guidance mode will be used.

9. After docking, the OMV will approach the Orbiter within a 30' distance of the payload bay. The Orbiter RMS will grapple the OMV and target stack, as it performs station-keeping at about a 30' distance.

10. The stack will then be properly stowed in the payload bay marking the end of the mission.

6.6.2 Study Areas

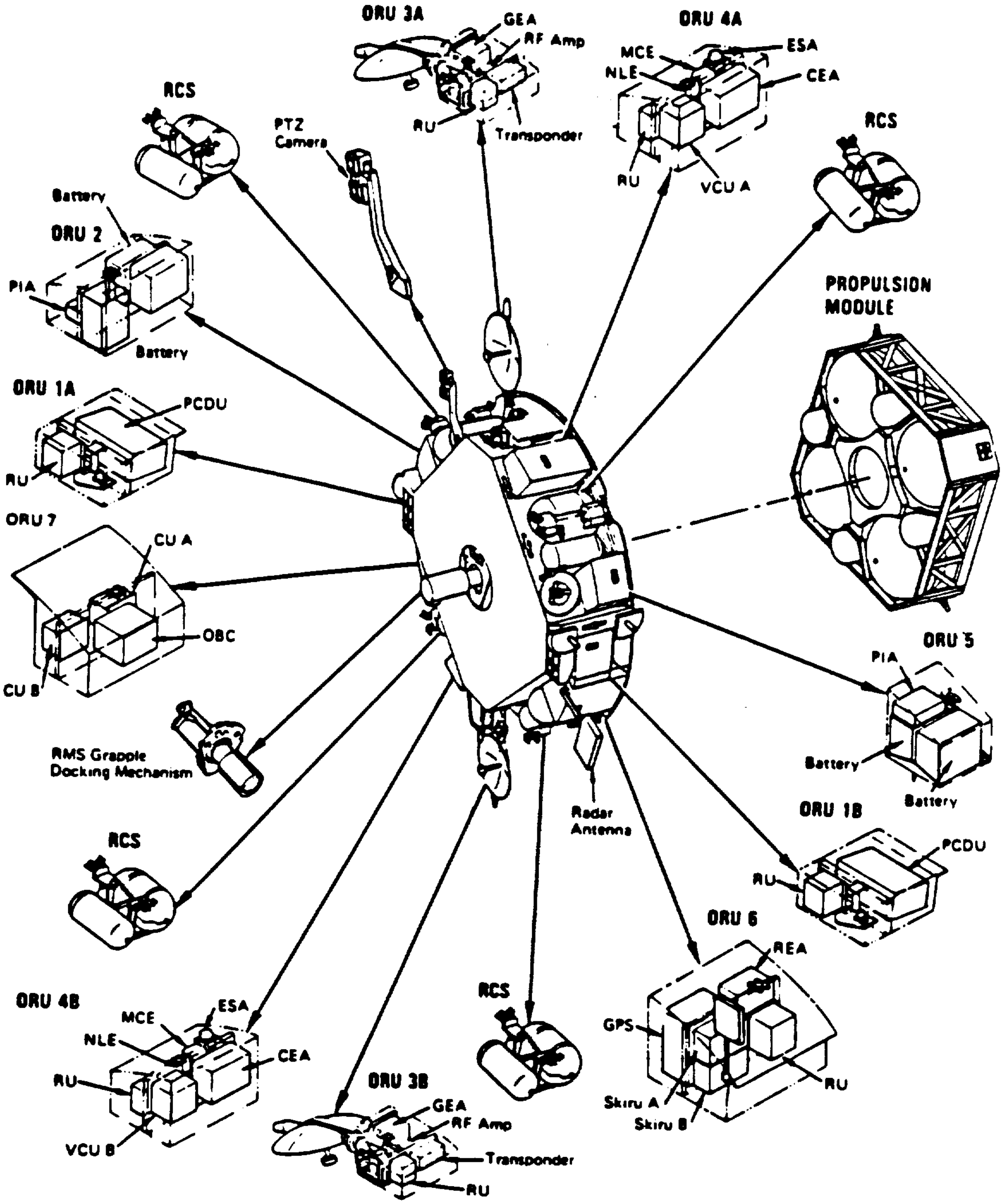
This section provides our results on several study areas. Each study area was investigated from the SSS point of view.

Emphasis was placed on the OMV capabilities required for an AR&D mission and how well the existing capabilities meet these requirements. We have identified the type of upgrade required in each area.

6.6.2.1 Absolute Navigation and Sensor Performance

There is a GPS receiver on-board the OMV that provides the inertial state from the GPS measurements of pseudo range and range rate. Accuracy of this derived inertial state is within 400 feet per axis in position and 0.9 ft/sec per axis in velocity. The absolute state propagator uses these measurements and incorporates them in the estimate of state.

For the SSS mission, these accuracies are sufficient and no other sensor will be required. The navigation software in the OMV uses gravity model with only J2 term and depends on highly accurate GPS and accelerometer measurements for its state propagation. It is recommended that this software be upgraded to include a high fidelity gravity model like 21 term Spline formulation.



CEA Control Electronics Assembly
CU Central Unit
ESA Earth Sensor Assembly
GEA Gimbal Electronics Assembly
MCE Mechanism Control Electronics
NLE Navigation Light Electronics
PCDU Power Control Distribution Unit
PIA Payload Interface Assembly
PTZ Pan, Tilt and Zoom
REA Radar Electronics Assembly
VCU Video Compression Unit

Figure 6-7 OMV Flight Vehicle System Elements

SATELLITE SERVICER SYSTEM FLIGHT DEMONSTRATION

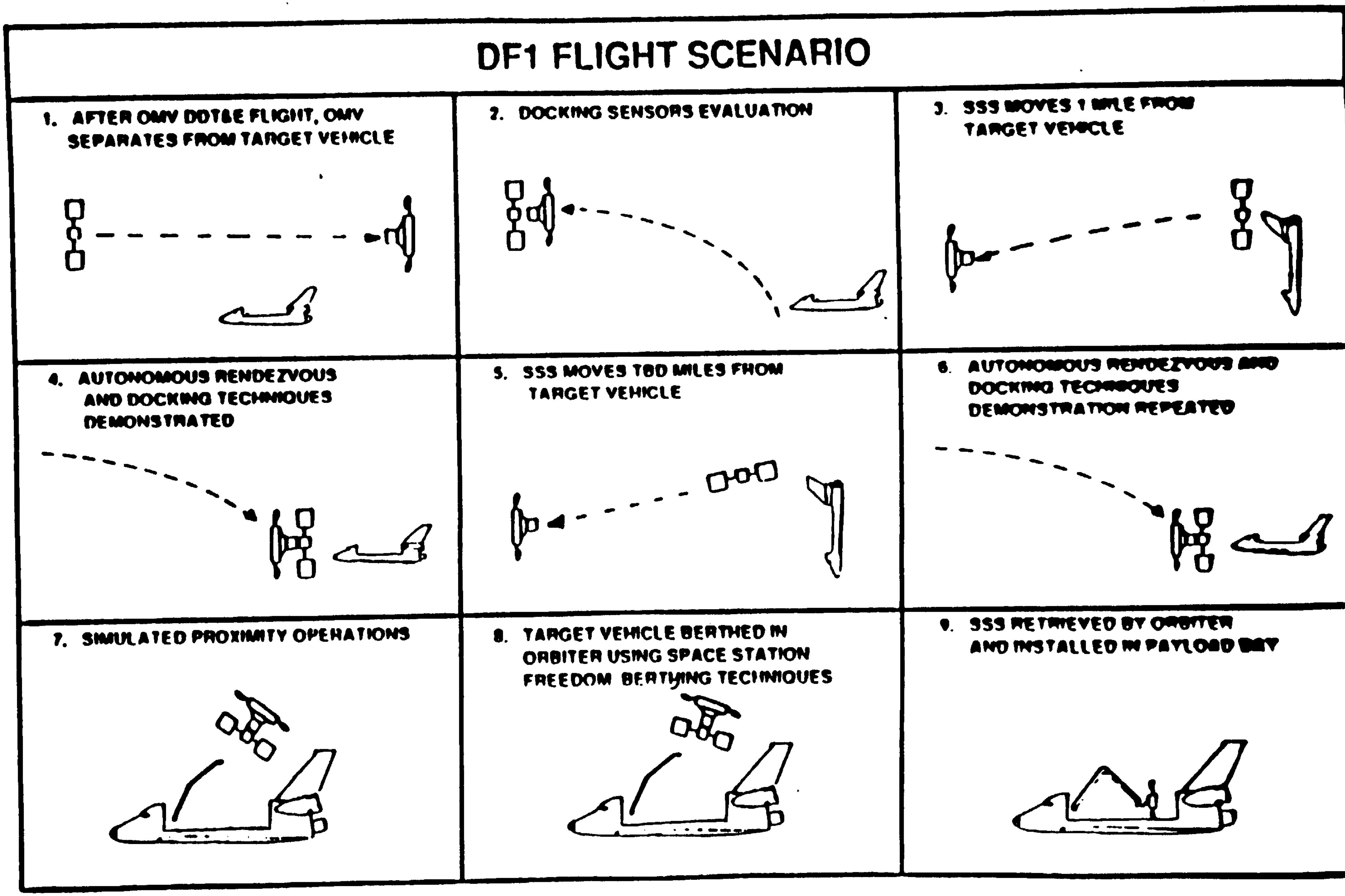


Figure 6-8

RELATIVE TRAJECTORY OF SSS WITH PAYLOAD

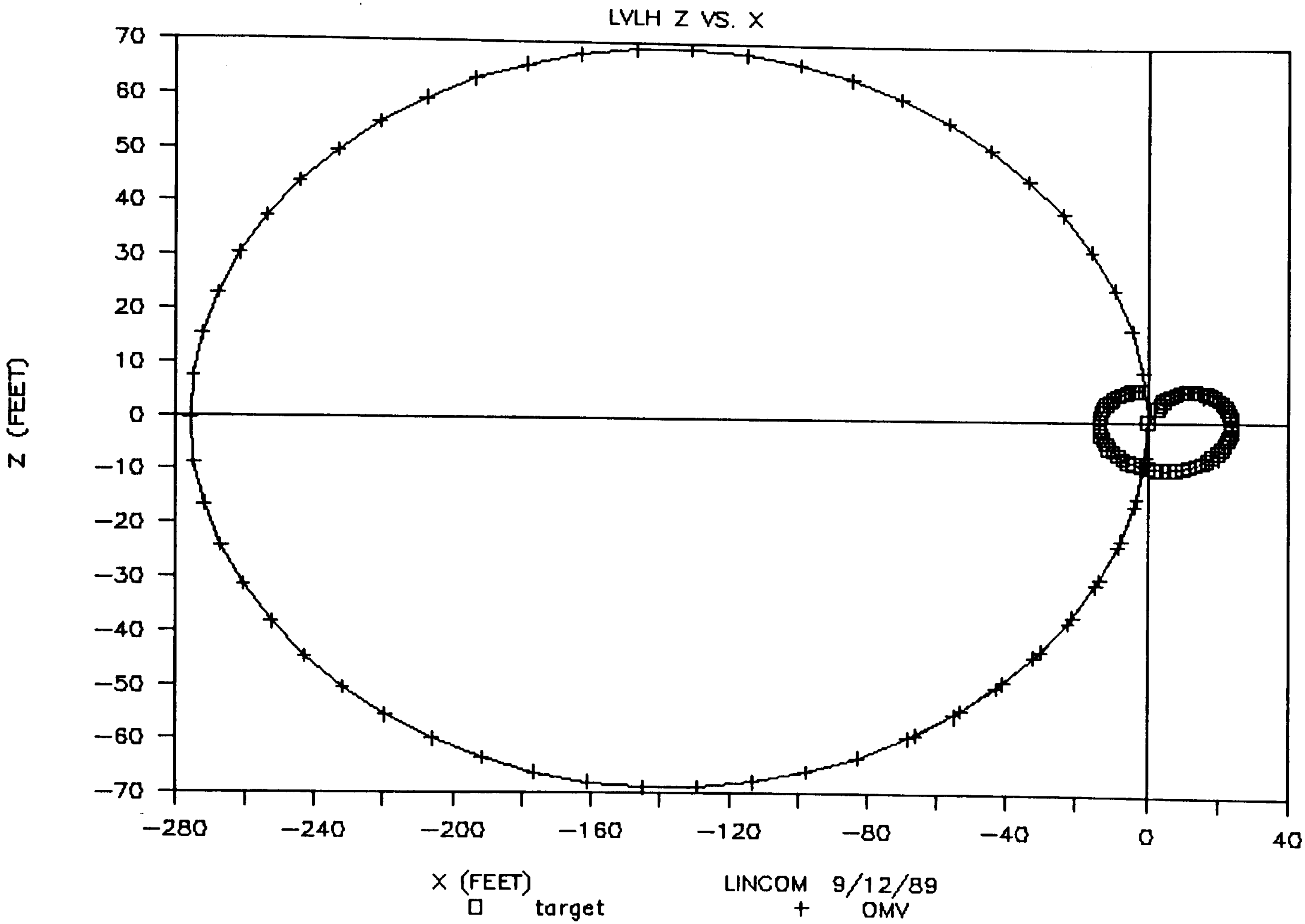


Figure 6-9

6.6.2.2 Relative Navigation and Sensor Performance

The relative navigation software incorporates sensor measurements to correct its propagated relative state (of which the basic form is that of CW equations in circular orbit). The sensor measurements are received from radar, laser docking sensors, IMU's (gyros and accelerometers), Sun sensors and Earth sensors. Performance of these sensors shows relatively small errors and is considered adequate. The measurement incorporation in the Kalman Filter helps to reduce errors in the state estimate. If the sensors fail or provide poor measurements, the state estimate will be very poor and the OMV will not be able to properly control its trajectory.

The range of the OMV's rendezvous radar is 4.5 nmi, with the following measurement accuracies.

Range: 20 feet or 2 % of range, or which ever is larger
Range Rate: 0.1 + 2.5E-5 * range

These accuracies are sufficient for rendezvous and docking from 4.5 nmi only. If the SSS objective is to perform rendezvous from 100+ nmi, the OMV will need a long-range relative sensor. There are several possibilities for this upgrade: 1) long-range radar with high power, 2) long-range laser which can also be used during docking, 3) star tracker with some co-operative reflectors on target, and 4) co-operative transponder on target for total communication.

The baseline OMV does not have a sensor that meets the requirements for final approach and docking. A short-range sensor and relative attitude sensor will be required for docking. It is assumed that the laser docking sensor described earlier will be available for OMV upgrade. This sensor will properly fulfill AR&D requirements.

6.6.2.3 Guidance and Control

The on-board software includes guidance for orbit transfer, rendezvous and station keeping at a 1000 ft distance. The OMV will need closed-loop guidance that can perform final approach from 1000 ft. and docking.

The control system of the OMV has 6-DOF and 4-DOF modes that provide the capability of maintaining a relative attitude with respect to the target and that properly execute the delta-v's. However, it is reported that this control system has some stability problems. The control system should be thoroughly examined for 6-DOF control requirements as outlined in section 6.5.3.2.

6.6.2.4 Targeting

Lambert and CW targeting is used for the OMV to perform orbit maneuvers. The CW targeting provides a linearized closed-form solution in a circular orbit and works for short distances (less than 20 miles). For longer distances, the Lambert solution is used to go from the initial position to a final position in a specified time. These algorithms, with some modifications, can provide almost optimal solution for finite burn effects.

6.6.2.5 Propulsion System

The baseline OMV propulsion system has three types of jets: 4 large variable thrust engines (13 to 130 lbs.) that use bipropellant fuel, 24 hydrazine Reaction Control System jets with 15 lbs. thrust, and 28 cold gas thrusters with 5 lbs thrust. These are mounted on the OMV in such a way that the system provides both 4-DOF and 6-DOF control modes during its operations. At this time, the OMV does not need any propulsion system upgrade.

6.6.2.6 Docking Performance

For the TPDM and RGDM, docking tolerances in relative position, velocity, relative attitude and angular rates are calculated based on the relative geometry and actual sizes of each mechanism as described below.

6.6.2.6.1 TPDM Docking Mechanism

TPDM is used to "rigidly" dock a chase vehicle (e.g., the OMV) with a target (e.g., a space telescope. Both, the chase and target vehicles have complementing docking mechanisms. The chase vehicle's docking mechanism has three attach points, 120 degrees apart, on a 36.25 in. radius circular plate (fig. 6-10 - 6-12). These attach points have two "catching arms" designed to grip towel-rack-like bars mounted on the target at matching locations.

The chase vehicle's docking mechanism is designed with larger spaces than required by the target's docking fixture. This space allows flexibility and sets upper bounds on the orientation error and relative position and velocity of the target at the time of docking. The OMV will not successfully dock if the target's translational and rotational motion is outside the bounds calculated here.

The Coordinate axis system for the docking mechanism is shown in figure 6-10. It can be observed that the capture envelope in the x, y, and z direction will be 4 in., 6 in. (7* cos 30x) and 3.5 in. (7* cos 60x). Delta displacements from the center of the envelope will be half these values.

6.6.2.6.2 RMS Grapple-fixture Docking Mechanism

The RGDM is used to perform docking between the chase vehicle (i.e., the Orbiter) and a target. Both, the chase and target vehicles have complementing docking mechanisms. The docking mechanism on the chase vehicle consists of an end effector which is connected to a robot arm. The end effector (fig. 6-13) is a hollow cylinder with 8.0 in. internal diameter and 21.0 in. length. A snare/wire (fig. 6-14) wraps around a nail-type grapple object. The docking fixture (fig. 6-15) on the target consists of a nail supported by three alignment cams and guide ramps, 120 degrees apart. When this nail goes into the end effector, snare wires wrap around it. Later-on rigidization is performed by pulling the nail inside the cylinder in such a way that the three cams mounted at the base of the nail fit exactly into the slots in the end effector.

The chase vehicle's docking mechanism is designed with more space than required by the target's docking fixture. This space allows flexibility and sets upper bounds on the orientation error and relative position and velocity. The Orbiter may not be able to successfully dock with the target if the translational and rotational motion exceed the upper bounds computed here.

Even though, the end effector is 21 in. long, the capture volume is given by the cylinder whose length is 4 in. and radius is also 4 in. This provides the position errors of 2 in. in each translational axis.

Position clearance (full range)

x = 4", y = 4", z = 4"

Angle accuracy

Roll angle = 15x
Pitch angle = 15x
Yaw angle = 15x

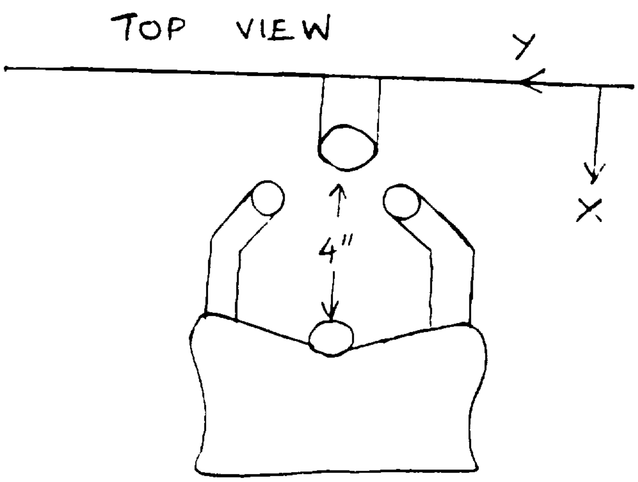
RMS capture time (or snaring time) is 1.1 sec and the rigidization time is 9.2 sec. Based on these values, maximum and desired velocity clearances and angular rate errors are calculated as follows: *Velocities Error* and *Angular Rates*

6.6.2.7 On-board Data Processing

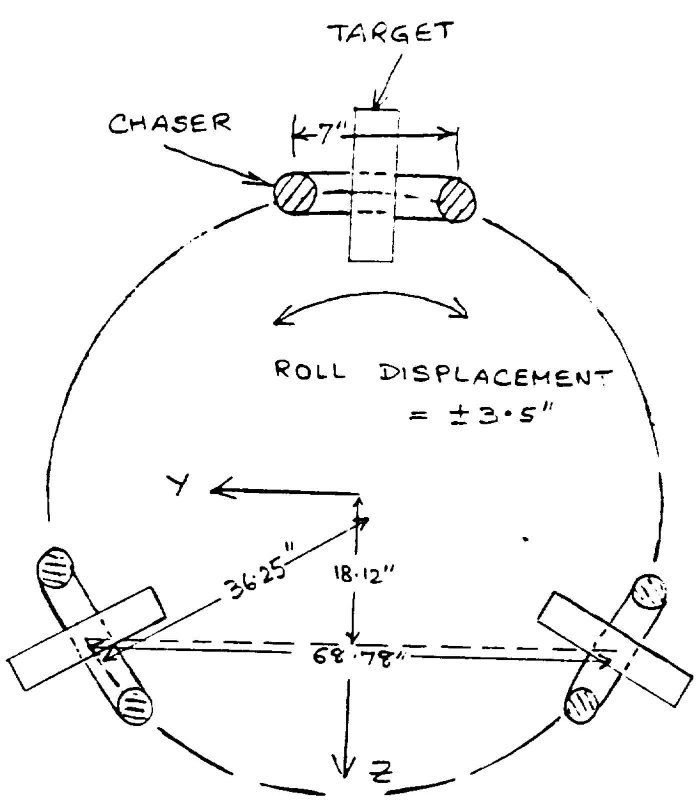
There is a large processing requirement for the AR&D System. It is expected that a computer with power in the range of 5 to 10 MIPS will be required. The on-board computer for the OMV does not have this capacity nor the memory to store all the software algorithms. The current data processing system will require a complete revision for the SSS missions.

6.6.2.8 Separation Requirements

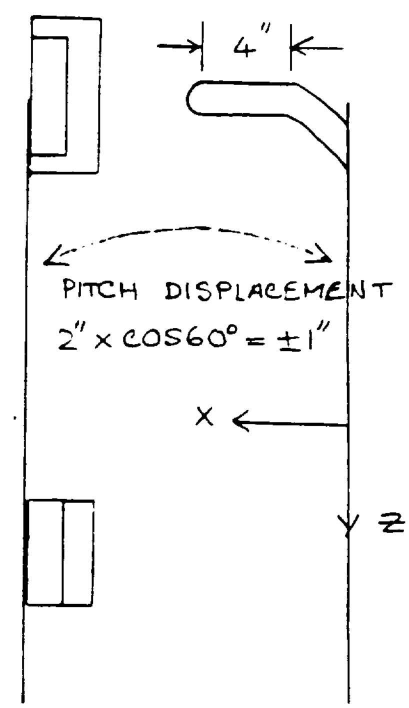
The current SSS profile requires the OMV to separate from the target and perform several operations including fly-around. The Orbiter will be used to simulate the Space Station Freedom and support all OMV activities. It will be necessary to place the target in such a trajectory that it does not drift away from the Orbiter and is not blocked by it. Simultaneously, the OMV trajectory should be such that there is no excessive delta-v penalty if it is to capture the target. A small study of relative trajectory was conducted to minimize the delta-v usage and find a suitable relative position for the Orbiter where it can observe without creating perturbations. The OMV and target were placed in a stable circular orbit. Several separation velocities were used to generate relative motion. After the release, the OMV was imparted a little more thrust to increase its relative distance to 300 feet. All trajectories generated were analyzed for the above constraints and conclusions were derived.



X 3



X1

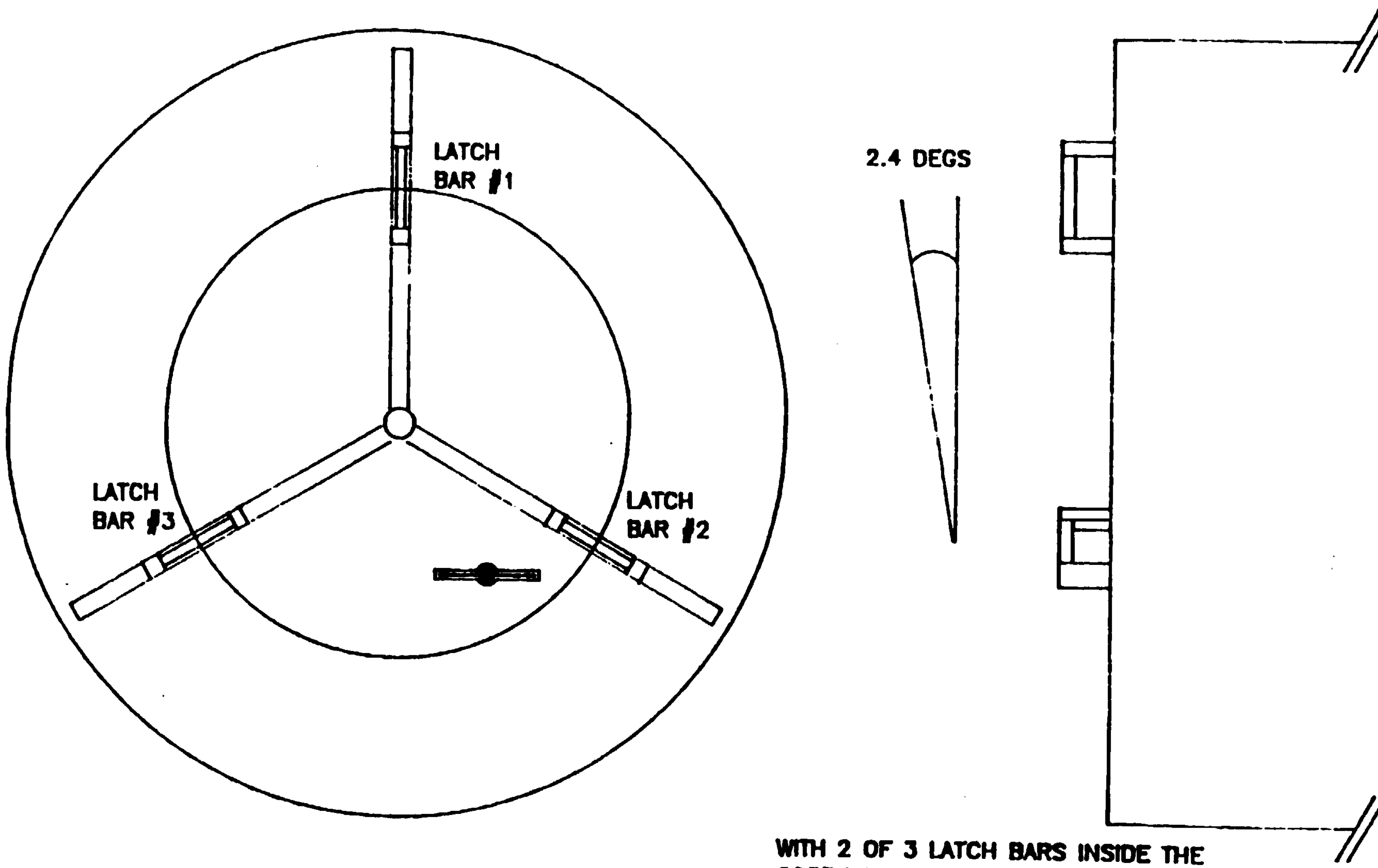


X2

THREE POINT DOCKING MECH.

(TPDM)

Figure 6-10



WITH 2 OF 3 LATCH BARS INSIDE THE CORRESPONDING LATCH CAPTURE RANGES, THE ANGULAR MISALIGNMENT CAN BE UP TO 2.4 DEGREES BEFORE THE THIRD LATCH IS PULLED AWAY FROM THE LATCH BAR.

TPDM LATCH POSITIONS

NOTE: POSITIONS AND ANGLES ARE NOT TO SCALE

Figure 6-11

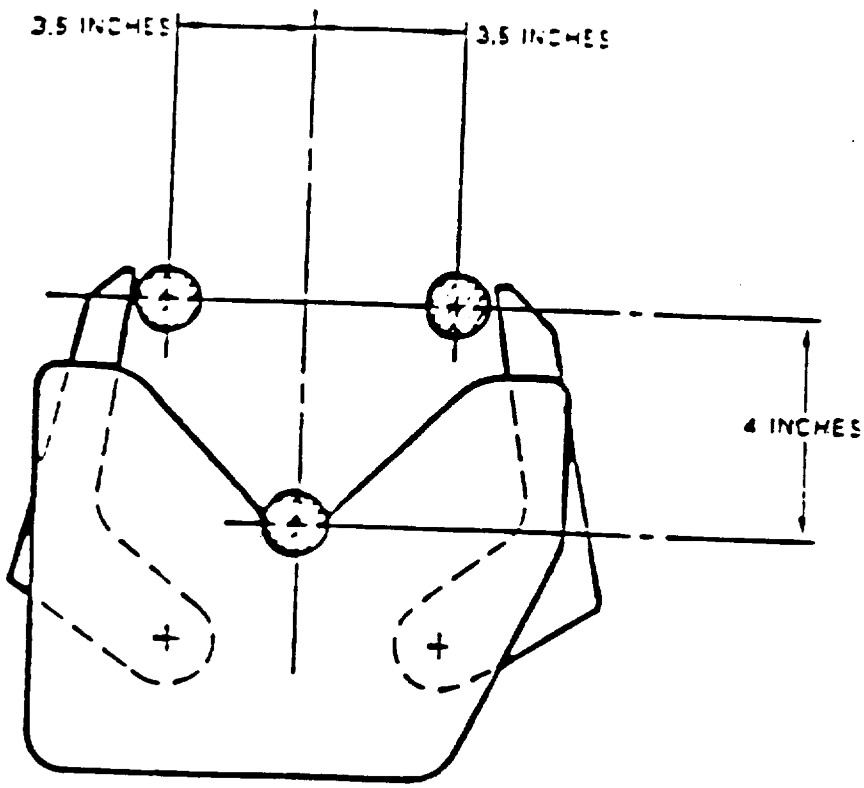


Figure 6-12. Latching Mechanism

NASA-S-78-3882 A

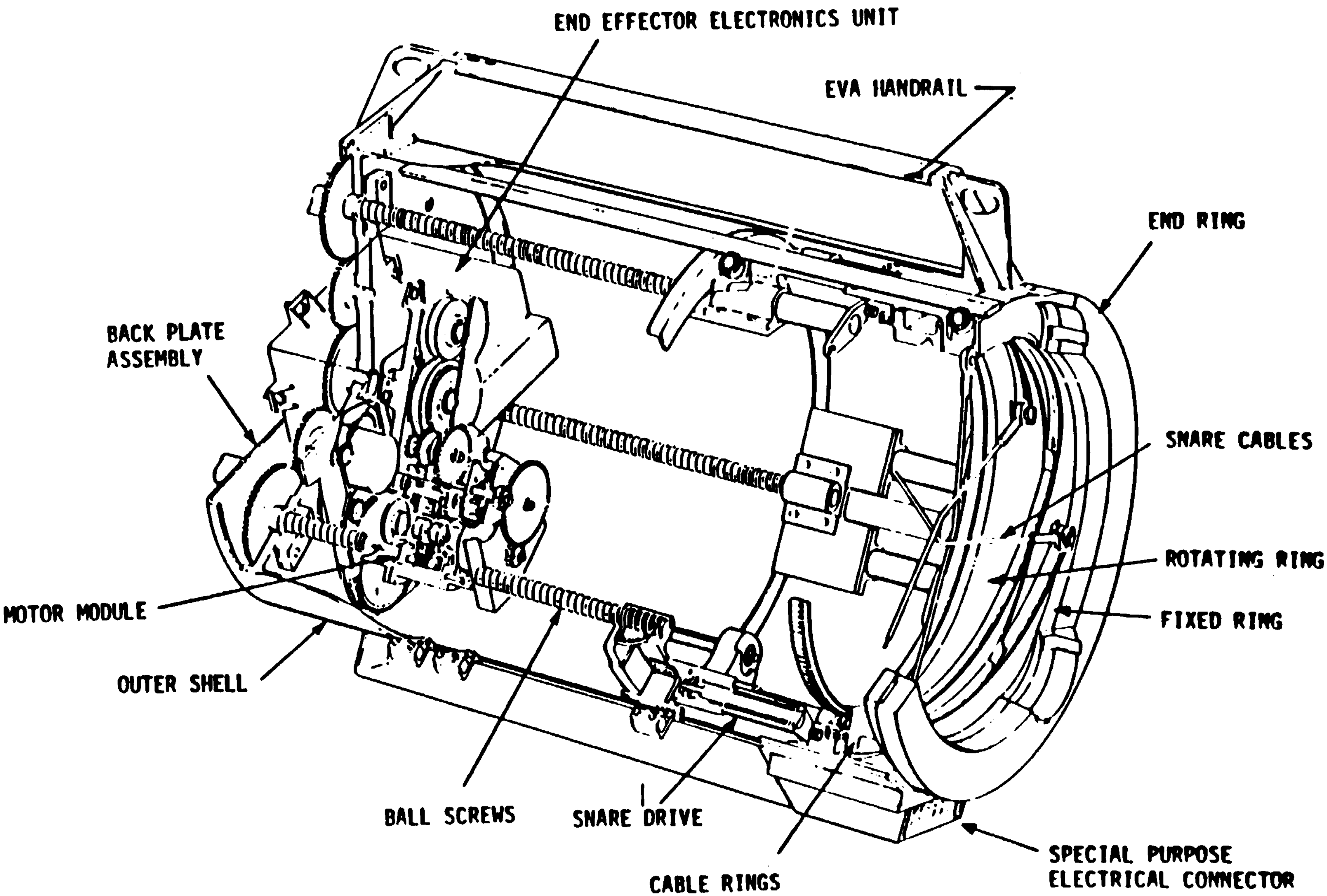


Figure 6-13. RMS End Effector

ORIGINAL PAGE IS OF POOR QUALITY

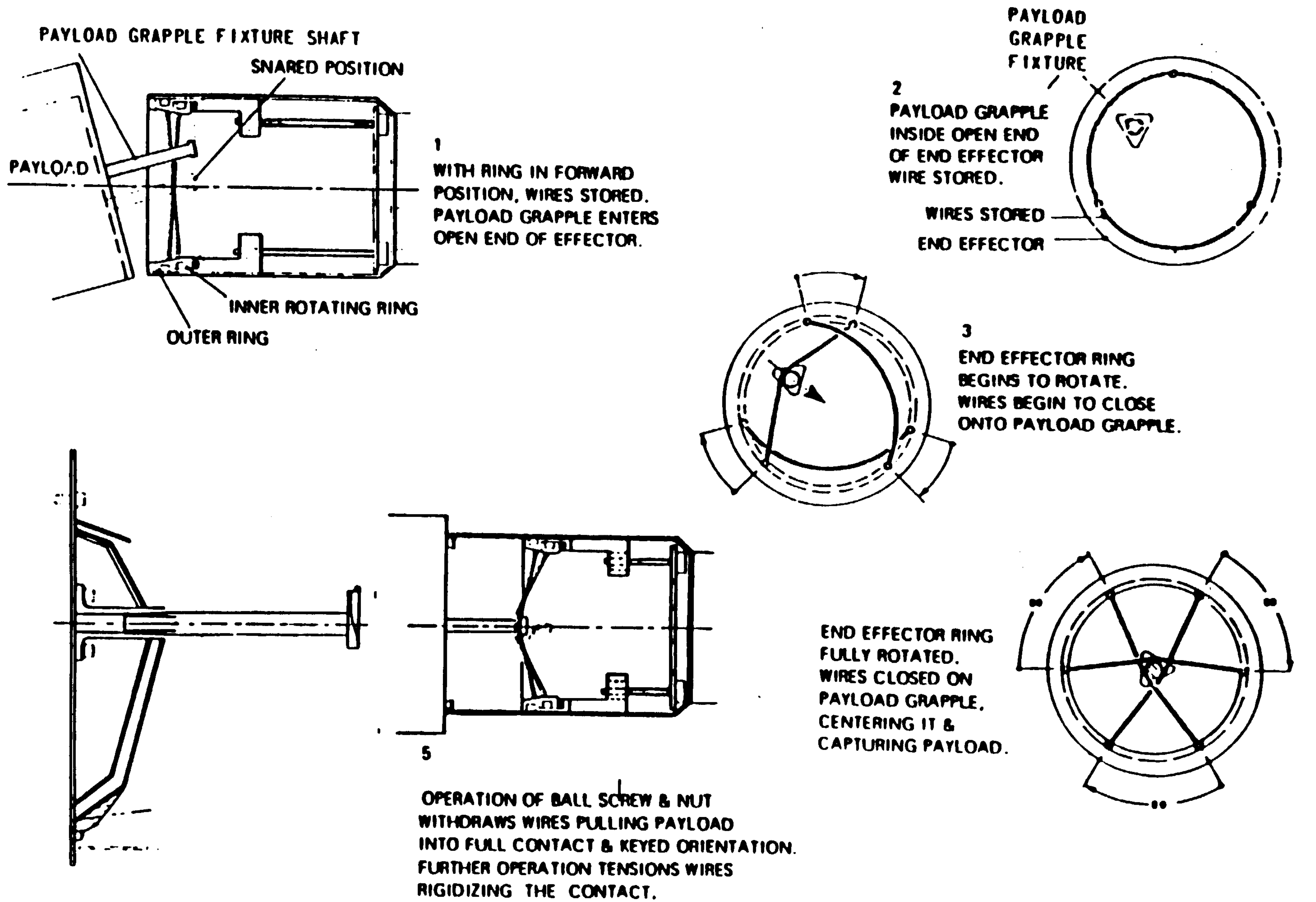
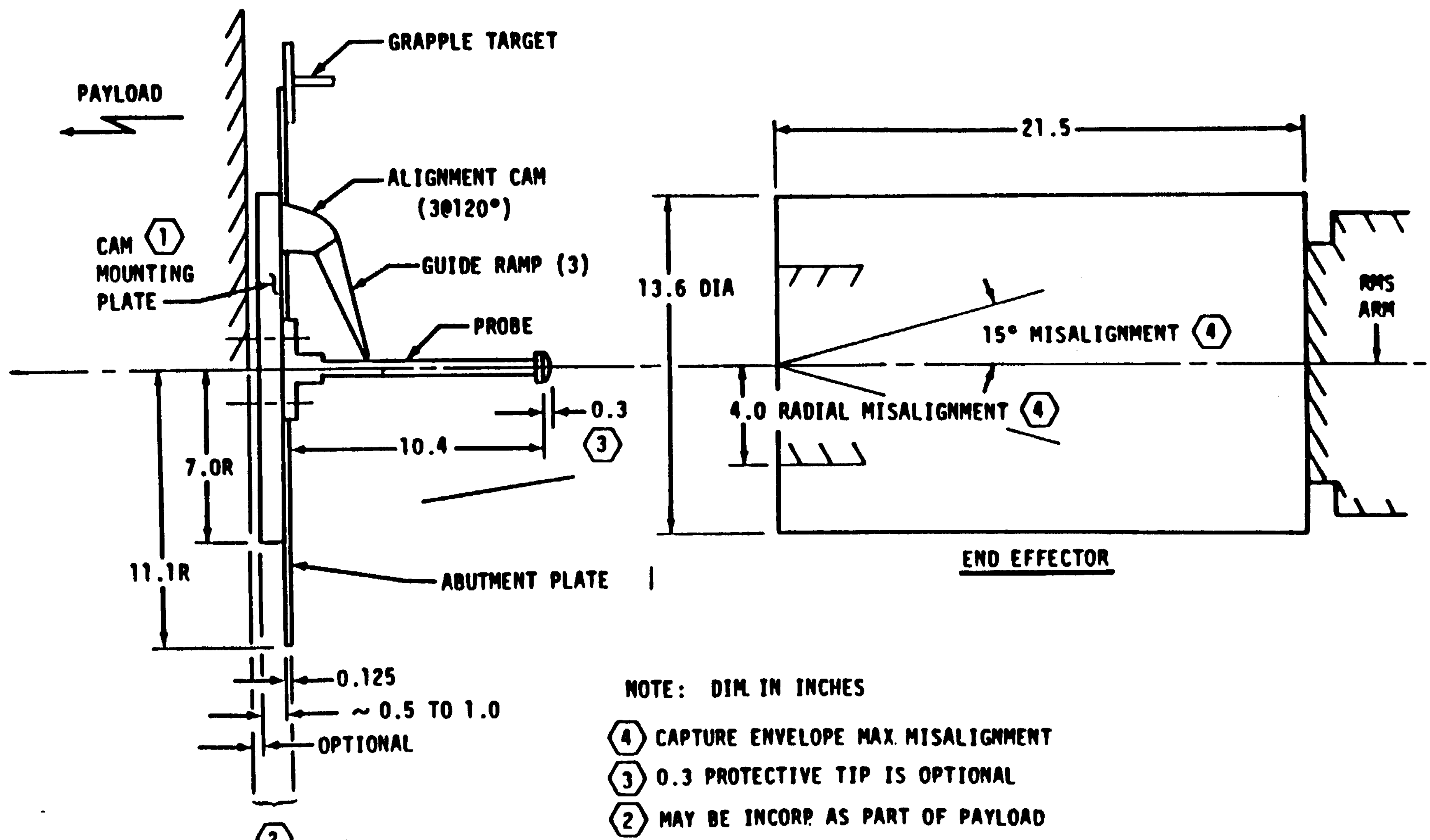


Figure 6-14. End Effector - Capture and Rigidize Sequence

NASA-S-78-2640 A

Figure 6-15. RMS Standard End Effector and Grapple Fixture Envelope Schematic



NOTE: DIM. IN INCHES

④ CAPTURE ENVELOPE MAX. MISALIGNMENT

③ 0.3 PROTECTIVE TIP IS OPTIONAL

② MAY BE INCORP. AS PART OF PAYLOAD

① STIFFNESS AND THERMAL CRITICAL

RELATIVE TRAJECTORY OF SSS WITH PAYLOAD

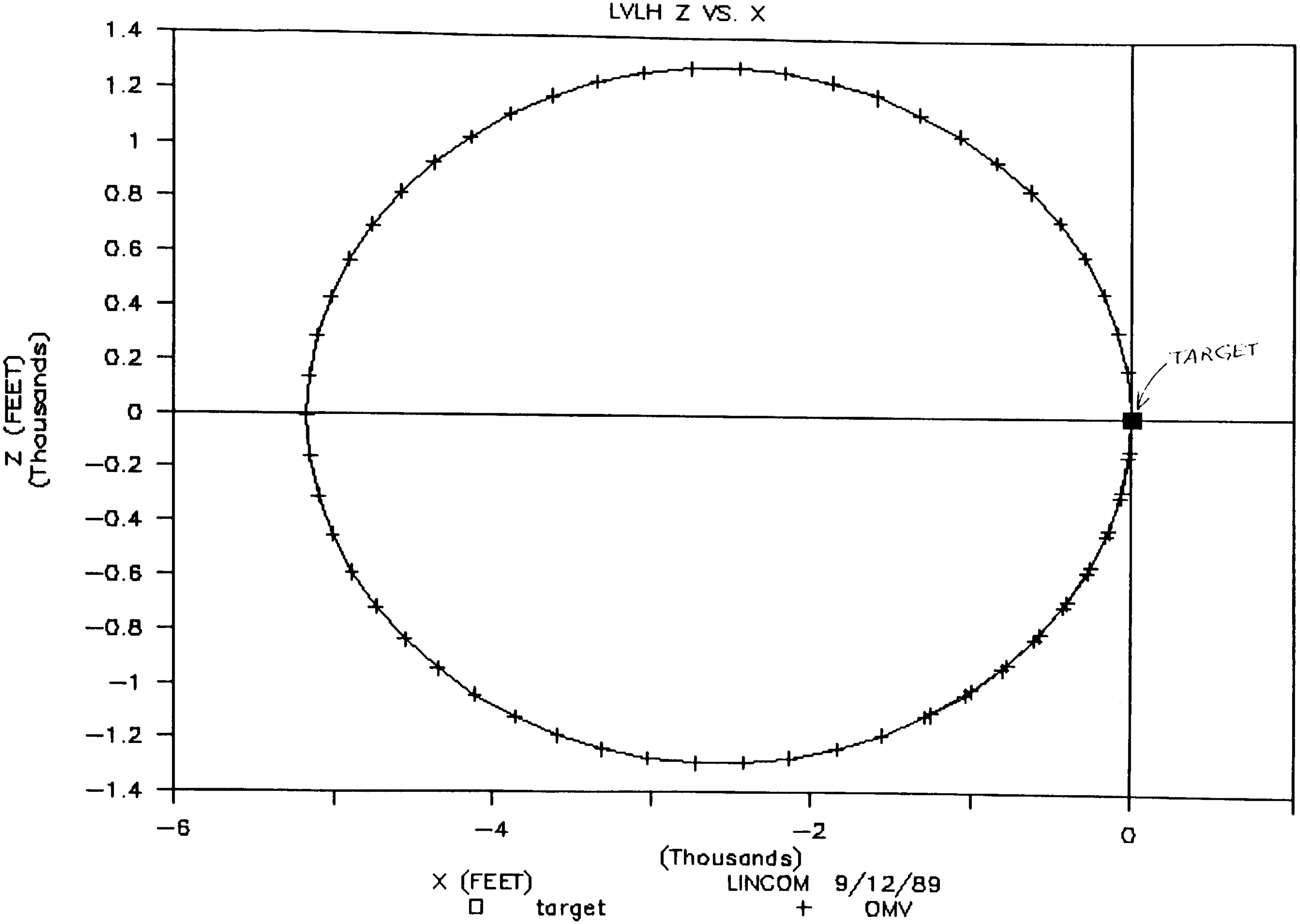


Figure 6-16

Separation velocity must be less than 0.01 ft /sec. If released at 45 deg pitch in LVLH, the target will circle the release point within a 20 ft radius in 90 min and will not drift away. The OMV can impart itself with 0.05 ft/sec velocity after the release and return to the same point within 90 min with less than 300' distance from the target. This will provide adequate time to evaluate the performance of a delivered payload, evaluate the docking sensor and short-range relative sensor, and perform docking if necessary with negligible delta-v penalty. The OMV can impart 1.5 ft/sec velocity to separate about one mile away from the target. Plots of these trajectories are shown in figures 6-9 and 6-16.

6.6.2.9 Mission Planner

For autonomous operations, the OMV will need an on-board mission planner that can plan the entire trajectory profile from 100+ nmi range. This profile generation requires computing a) pointing attitude for target acquisition, b) time to point the antennas, c) time estimates to perform orbit transfers, d) correction delta-v's, and e) stopping delta-v's and its time. It also requires numerous parameters from which the mission is planned and carried out. Furthermore, the mission planner must have the capability to plan the terminal rendezvous, final approach and docking events. The ability to differentiate between station-keeping and timeline replanning is also required. For this type of capability, an AI-based approach is anticipated, which includes processing of rules and knowledge bases. Current OMV planning will require such an upgrade for complete autonomous operations.

6.7 SUMMARY

In this report, our preliminary findings for autonomous trajectory control for MRSR and SSS missions are described and requirements levied on the AR&D System are discussed. These results need to be supported by detailed analysis and demonstration where feasible. Future work includes development of the exact sequence of steps for docking, real-time processing of sensor data and commands generation, mission timeline generation, and new sensor concepts that can easily achieve docking. The mission planner is an important component of the AR&D System that needs highly focused attention. At this time, the OMV is certainly a system that can be easily upgraded for AR&D missions.

REFERENCES

1. Satellite Servicer System Flight Demonstration, Statement of Work for Phase B Definition, June 1, 1989.

2. System Interface Control Document ICD2-19001, Vol. 14, Space Shuttle Payload Accommodation Section 8.1, JSC-07700, May 1985.

3. Mission Scenario Assessment- Autonomous Rendezvous and Docking, by R. Lee, TRW memorandum #89:W480.1-67, April 1989.

4. MRSR Reference Missions Summary, Version 2.3, FY88 Final Programmatic Review, JPL presentation and handout, September 1988.

SECTION 7
AUTONOMOUS RENDEZVOUS AND DOCKING
BASIC MECHANISM REQUIREMENTS
(Prepared by Lockheed Engineering & Sciences Co-Houston)

7.1 INTRODUCTION

7.1.1 Purpose/Objectives

A rendezvous and docking event may be considered completely or partially autonomous. Specific systems may operate autonomously, while the remaining systems depend on the autonomous systems or manual inputs. The emphasis of this section is placed on maximizing autonomy of AR&D operations, with specific emphasis on docking mechanism operations. The information reported here was released by LESC.

Autonomy of docking mechanism operations is feasible with the use of appropriate sensors, electronics, and software to control mechanism components. The docking mechanism for AR&D is essentially the same as that for docking operations which require human involvement. However, the interfaces between the hardware and humans are replaced by an interface between a computer and the docking hardware. The computer may provide simple control commands to mechanism components (e.g., mechanism extension command upon completion of terminal phase rendezvous), or it may provide diagnostic and decision making capabilities (e.g., optimization of attenuation based on feedback from load sensors). Overall, the basic mechanism design requirements for application of AR&D technology involve the same considerations as those for a standard docking mechanism, with additional electronic and hardware interfaces to allow for autonomous operations.

As emphasized previously, development of AR&D technology would enhance and expand the space flight capabilities of NASA. The general feasibility of docking has been proven repeatedly through the Gemini and Apollo programs. However, the entire docking procedure was conducted manually in each of those programs. In addition, since the Apollo-Soyuz Test Project, no docking events have been conducted by NASA, since the primary emphasis has been on development of the Space Shuttle. The development of Space Station Freedom (SSF), the initiation of on-orbit satellite servicing and retrieval, and the potential future programs of space exploration all indicate the need for AR&D capabilities.

The availability of AR&D capabilities could increase the options for unmanned interplanetary exploration missions involving multiple vehicles, especially for planetary sample return. A prime example of this kind of mission is the MRSR mission planned for the late 1990s.

The concept of AR&D is not limited to planetary exploration, but can also be applied to operations in Earth and lunar orbit. The OMV, designed for satellite servicing and reboost, will expand NASA's operational capabilities in LEO. The current plans for the OMV involve remote piloting for docking operations. The application of AR&D to the OMV would extend its capabilities, allowing use of the OMV at greater distances from a manned control station. The proposed development of the Orbital Transfer Vehicle (OTV), similar to the OMV but with greater propulsion capabilities, would extend the capabilities of the SSS to Geosynchronous Earth Orbit (GEO). In addition, the OTV would provide the United States space program with a convenient means to reach the Moon, and could support development of a permanent manned lunar base.

The primary objective of this segment of the document is to relate the requirements for design of a docking mechanism specifically geared towards autonomous applications. To familiarize users of this document with docking, a brief history of docking by the U.S. and U.S.S.R. is given in section 7.2. The requirements for the AR&D mechanism are discussed in section 7.3, beginning with a brief description of the mechanism functions required for three proposed mission scenarios. The basic functional requirements for docking are similar for each scenario. However, the projects proposed for utilization of AR&D technology have varied objectives, with each mission placing additional requirements on the docking mechanism. The vehicles proposed for the application of AR&D are currently in the preliminary design stage at best. Since data is not available for the development of specific design requirements, the presentation of this section will be general to provide necessary information for development of requirements for any AR&D mechanism. Where applicable, specific requirements for currently proposed missions are addressed following the discussion of general requirements.

7.1.2 Scope

It is intended that this section of the document be beneficial to any program involved in the design, development, integration, operation, test, and verification of any docking mechanism designated for use in AR&D. This section addresses, but is not limited to, three specific mission scenarios which are candidates for application of AR&D: 1) the MRSR mission, 2) OMV satellite servicing in LEO, and 3) LEO / lunar orbit transfer with the OTV. These three mission scenarios have been selected by the AR&D Project for initial emphasis due to availability of information for the spacecraft and mission objectives. The majority of future missions applying AR&D are expected to be similar in objectives and requirements to the three primary scenarios discussed herein. Since a docking mechanism to be used in a manned AR&D application will require design for manual backup, requirements for the design of mechanisms requiring human interfacing are included. Therefore, this document may be applied to docking mechanisms to be used during manual as well as autonomous operations

These requirements are designated as the starting point for AR&D mechanism design and will be applied in coordination with the "AR&D Systems Requirements Document." They may be used for the development and implementation of lower level requirements for specific vehicles as each program progresses. Each program applying AR&D will have specific requirements for docking mechanism design due to mission requirements and objectives. Therefore, vehicle and mission specific requirements beyond this document will be specified by the responsible program.

7.2. SUMMARY OF U.S./U.S.S.R. DOCKING HISTORY

The space programs of the United States and the Soviet Union include a lengthy history of operations involving rendezvous and docking. The first spacecraft docking was achieved by the U.S. in March of 1966, when Gemini VIII docked with an unmanned Gemini Agena Target Vehicle. Docking was successfully completed in three of the four remaining Gemini missions.

The Gemini missions used a solid cone docking system, in which a cone mounted on the docking adapter of the target vehicle was used to guide the active vehicle during the final moments of docking. The Gemini spacecraft's approach was completed with visual cues only, using no alignment or ranging sensors. After engagement of the capture latches, rigidization of the latches automatically secured the cone to the structure of the active vehicle.

The Soyuz era of the Soviet space program was inaugurated in November of 1966 with the launch of Kosmos 133, an unmanned version of Soyuz. The Soyuz spacecraft were the first Soviet vehicles with rendezvous and docking capabilities, using a probe-drogue mechanism similar to that used in the U. S. Apollo program. "Automated" rendezvous and docking was successfully completed twice by the unmanned Kosmos vehicles. The degree of automation is not clearly defined, but a manual capability was available during all operations.

The first docking of two manned spacecraft occurred in January of 1969, during the flight of Soyuz 4 and 5. The probe and drogue mechanism used for this docking did not allow for pressurized crew transfer between the vehicles at the mechanism interface. Therefore, an extra-vehicular transfer was completed by the crew of each vehicle.

Docking was an integral part of the mission profile for the U. S. Apollo program. Two independent docking events were required for the planned lunar missions. First, the Command/Service Module (CSM) was required to dock in Earth orbit with the stowed Lunar Module (LM) in order to release it from the Saturn IVB third stage. The LM would then undock in lunar orbit, descend to the lunar surface, and return to lunar orbit to dock again with the CSM after completion of the surface activities.

A probe-drogue system was selected for the Apollo program because it required minimum changes to the basic vehicle structure, had low system weight, and demonstrated superior dynamic characteristics in latching capability. This system was first used in March of 1969 during the Apollo 9 mission. The probe assembly was mounted on the CSM while the drogue was attached to the LM. Capture latches were housed in the probe head. After attenuation of contact loads by the probe, the vehicles were drawn together to effect structural latching. Twelve automatic latches mounted on the CSM docking ring provided for structural integrity and tunnel sealing between the vehicles. The release of a single lock allowed the probe assembly to be manually removed to allow crew transfers.

The second generation Soyuz docking mechanism, introduced in 1971 when Soyuz 10 docked with Salyut 1, featured a removable probe and drogue assembly to eliminate the need for extra-vehicular transfers. After structural latching, the probe assembly would swing open into the Soyuz while the drogue opened into Salyut.

Although the probe-drogue mechanisms performed adequately, some significant shortcomings became apparent. First, the probe and drogue could only be used as a matched set, reducing the number of possible vehicle combinations in rescue situations. Second, the need arose for a docking system in which the mechanism did not obstruct the transfer passageway between vehicles. Finally, due to the fact that the length of the probe-drogue mechanism is proportional to its diameter, potential problems existed for large diameter docking ports.

As early as 1966, U.S. designers had proposed an androgynous docking system surrounding the periphery of the docking port. The mechanism featured a ring and twelve guides which would match the ring and guides of an identical interface. Designers later concluded that four guides would be optimum for interface alignment.

In 1970, NASA and the Academy of Sciences of the U.S.S.R. opened discussions on means for ensuring compatible docking between future spacecraft. In 1972, an agreement was signed by the U.S. and the U.S.S.R. calling for a rendezvous and docking mission of a U.S. spacecraft with a Soviet spacecraft in 1975, leading to the ASTP.

The U.S. proposed an androgynous design which featured four guides with eight hydraulic attenuators. The Soviet design, which was configured to utilize the existing structural interface of the Soyuz, was also androgynous and featured three guides with six electromechanical attenuators. The Soviet concept, with the incorporation of the U.S. capture latch design, was selected as a baseline. It was agreed that no automatically engaging electrical connectors would be used at the docking interface, and that a seal-on-seal principle would be used for pressurization.

Separate docking mechanisms were manufactured by each country to the agreed specifications. The Soviets refitted the Soyuz with the androgynous mechanism, while the U.S. built a Docking Module (DM) to be used with the Apollo CSM.

The ASTP mission was launched in July of 1975. First, the Apollo docked with the DM to retrieve it from the Saturn third stage, then preformed the maneuvers necessary to rendezvous with the Soyuz spacecraft. Docking was performed successfully twice, first with the CSM as the chase vehicle and having the active docking mechanism, then with the Soyuz as the chase and active vehicle.

Since ASTP, the Soviets have continued to use a probe-drogue docking system with the Soyuz/Salyut flights. In 1978, Soyuz 27 docked with Salyut 6 at the aft docking port while Soyuz 26 was still docked at the starboard port, becoming the first simultaneous dual mission to a manned space station. Part of the Soviet station program includes Progress, an unmanned version of the Soyuz spacecraft modified for resupply and refueling. This vehicle requires automated docking, with Progress as the active vehicle. The degree of autonomy involved in Progress docking is undetermined. Manual control is available via ground control stations throughout rendezvous and docking.

The many successful docking events of the past, both manned and unmanned, manual and automated, indicate that autonomous rendezvous and docking is a reasonable and realistic goal. This confidence is demonstrated by the ongoing studies involving Orbiter to Space Station docking, during which the Orbiter is to be manually mated to the station. A probe-drogue system has been baselined for this task. In addition to Orbiter-to-Space Station docking studies, the concept of an International Docking Mechanism (IDM) is being researched. The IDM is intended to be a universal docking mechanism for use on the manned spacecraft of all nations. The common system would allow docking of any vehicle equipped with the mechanism to any other such vehicle.

7.3 AR&D MECHANISM REQUIREMENTS

The AR&D mechanism will provide the capability for autonomous rendezvous and docking between two spacecraft, unmanned or manned, in LEO and beyond to outer space and other planets, such as Mars. Standard procedures for rendezvous and docking stipulate rendezvous of the two vehicles at some distance to be specified by the program. From this point, proximity operations (PROX OPS) will be performed to maneuver the spacecraft to docking interface contact. The docking mechanism will then align the vehicles, attenuate loads, and structurally connect the spacecraft. The mechanism will provide the necessary mission-specific capabilities for transfer of crew, supplies, hardware, experiments, planetary matter, etc., whether pressurized or in a vacuum. The docking mechanism design methodology and requirements for autonomous applications do not differ greatly from the design of a manually operated docking mechanism, since the same parameters, such as contact conditions, are taken into account. The difference between autonomous and manual docking will lie in the derivation of design requirements for system capabilities versus pilot capabilities. For AR&D applications to manned vehicles, autonomous operations will be emphasized, but the design of the docking mechanism will account for human control. Specific requirements for AR&D mechanisms as applied to the missions mentioned in section 7.1 are presented in the following paragraphs.

7.3.1 <u>Mechanism Functional Descriptions</u>

Several different missions in which AR&D technology could be applied have been proposed. The basic hardware functions for the three primary mission scenarios are described in this section of the report: 1) MRSR mission, 2) OMV satellite servicing, and 3) OTV applications in support of the development of a manned lunar base via SSF and co-orbiting facilities in Earth and lunar orbit. For those readers unfamiliar with the mission scenarios, a list of references is included in section 7. 4.

7.3.1.1 MRSR Mechanism Description

The MRSR AR&D mechanism will enable autonomous docking of the MAV to the ERV, allowing an unpressurized transfer of a SRC from the MAV to the ERV. The interface configuration remains unspecified, with an androgynous and a probe-drogue interface as the initial candidates under investigation. Examples of the proposed interface configurations are illustrated in figures 7-1 and 7-2. The mechanism will consist of a passive docking mechanism on the MAV to reduce Mars lift-off mass, with an active or passive docking mechanism for the ERV. The ERV may require a removable Rendezvous and Docking Module (RDM) to minimize the ERV mass for the return flight.

The AR&D mechanism will be designed to accommodate the desired method for SRC transfer from the MAV to the ERV. The design of the docking mechanism will be such that proper connections between the transfer mechanism components on each vehicle are ensured. The transfer mechanism design will be considered in the design of the docking mechanism to ensure optimization of the docking and SRC transfer mechanism mass and volume allocations for the MAV.

The docking mechanism will be designed such that the mechanism may remain dormant for up to two years, with the capability of configuration to a docking readiness state within a period of time to be determined. The dormancy period may be interrupted to conduct periodic docking readiness tests (e.g., every six months).

In the AR&D mechanism design consideration will be given to a self-diagnostic capability for monitoring and verification of all critical mechanical, electrical, and thermal control systems. The monitoring function may be activated periodically throughout the mission. The self-diagnostics capability will remain available until separation of the ERV from the MAV.

7.3.1.2 OMV/SSS Mechanism Descriptions

Autonomous satellite servicing with the OMV requires an AR&D mechanism capable of mating the OMV to a spacecraft for servicing, reboost, or relocation. The mechanism will provide a rigid interface between the OMV and the payload and establish the necessary umbilical connections.

Current applications for the OMV as part of the SSS call for remotely piloted docking using one of two proposed mechanisms: the standard OMV docking mechanism, referred to as the RGDM or the TPDM. These mechanisms are illustrated in figures 7-3 and 7-4. The RGDM, which is retractable, is configured for payloads less than 20,000 pounds. The TPDM is an attachable mechanism designed for use with the Hubble Space Telescope (HST) and may be applied to any cantilevered payload similar to the HST weighing up to 75,000 pounds.

Use of the two mechanisms is covered in the "User's Guide for the Orbital Maneuvering Vehicle," which discusses the various proposed missions for the OMV. Further information on the OMV interfaces is available in SSP 30441 for the Space Station, and in the "OMV/HST Interface Requirements Document" for HST.

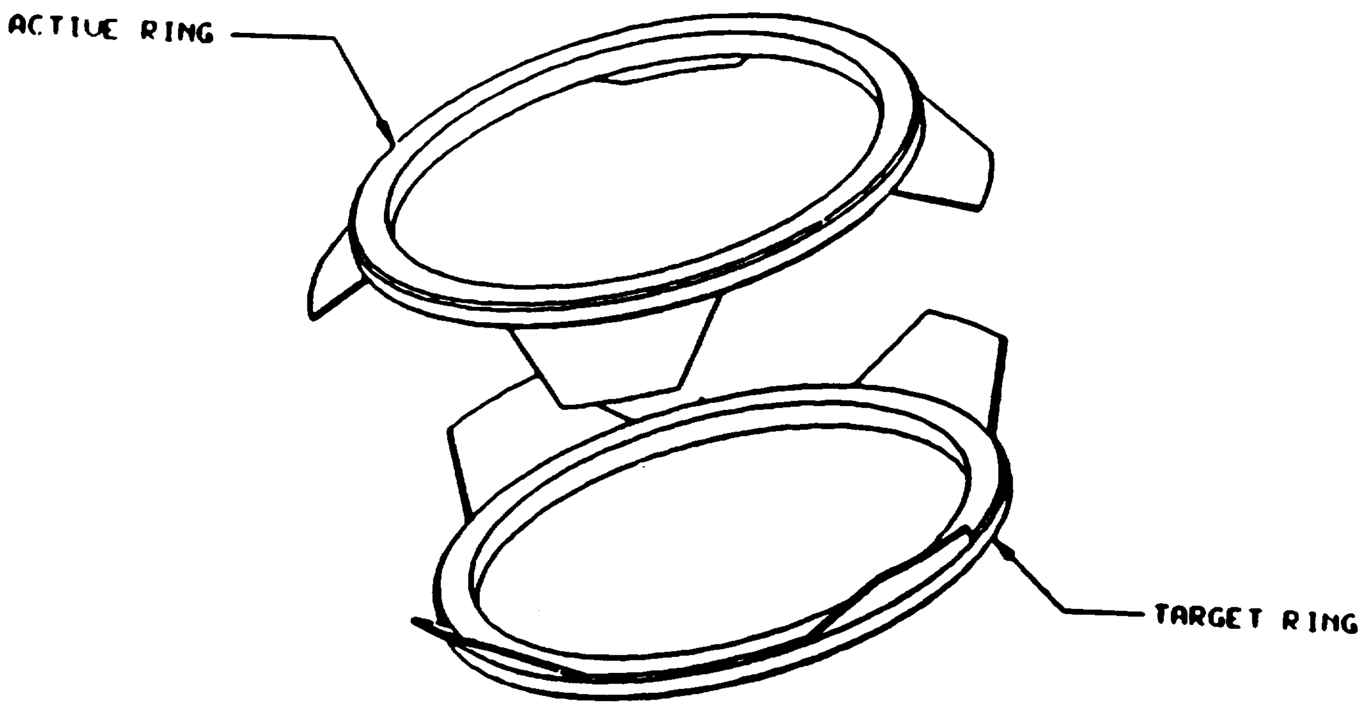


Figure 7-1. Androgynous Interface

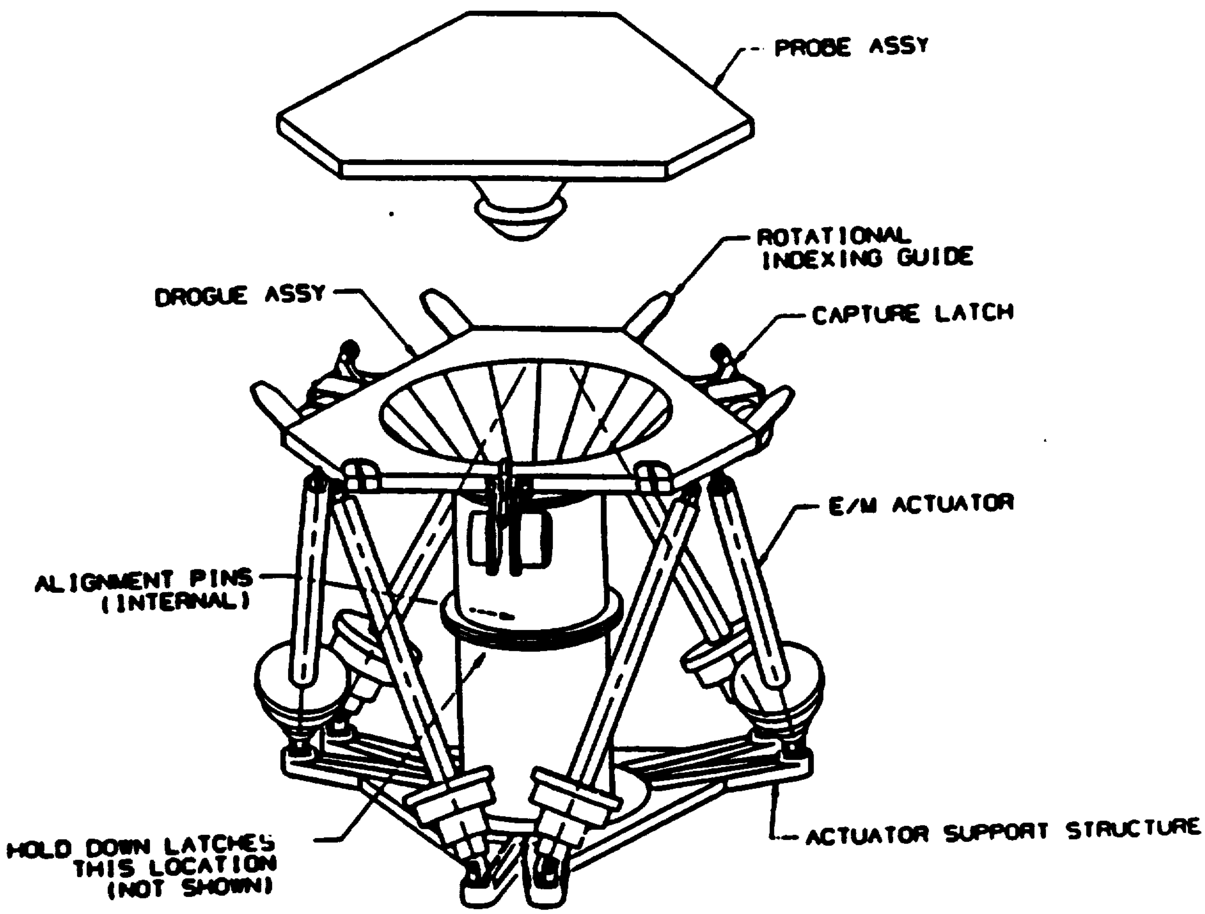


Figure 7-2. Probe-Drogue Interface

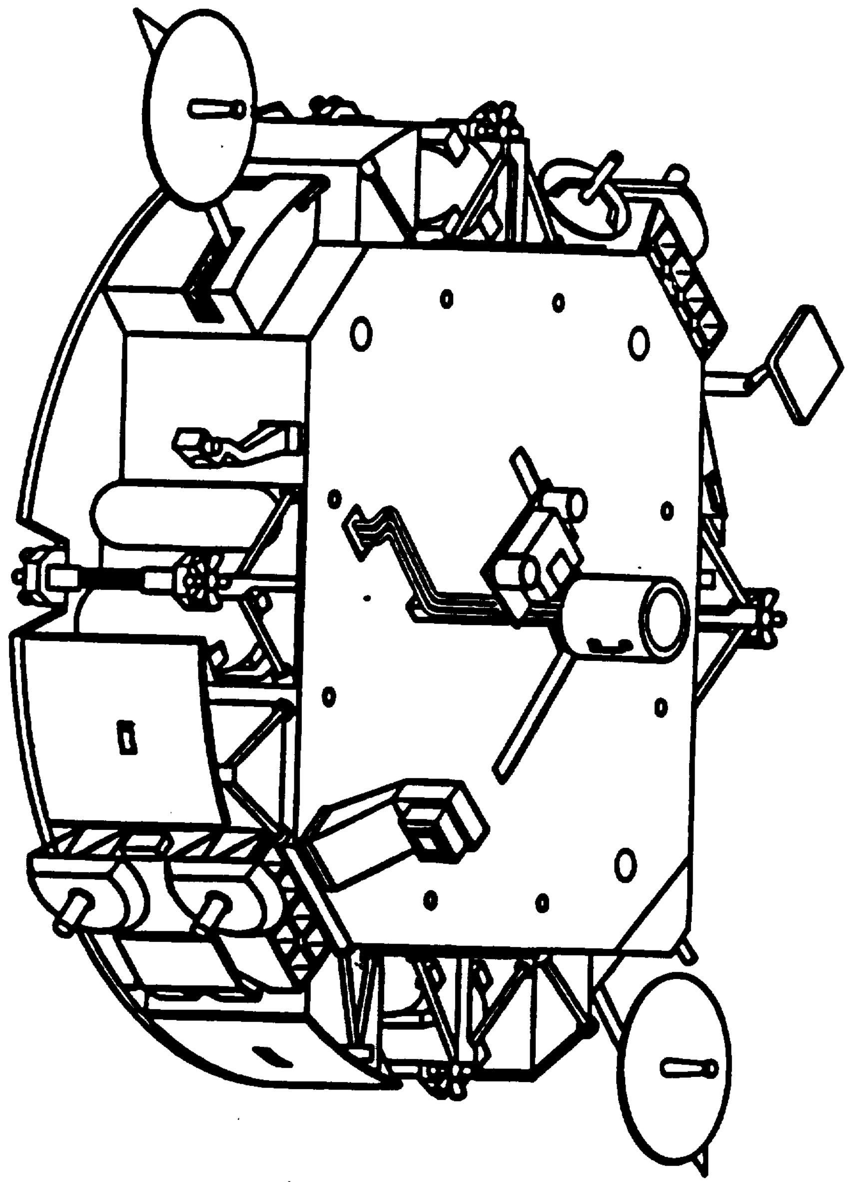


Figure 7-3. Grapple Docking Mechanism

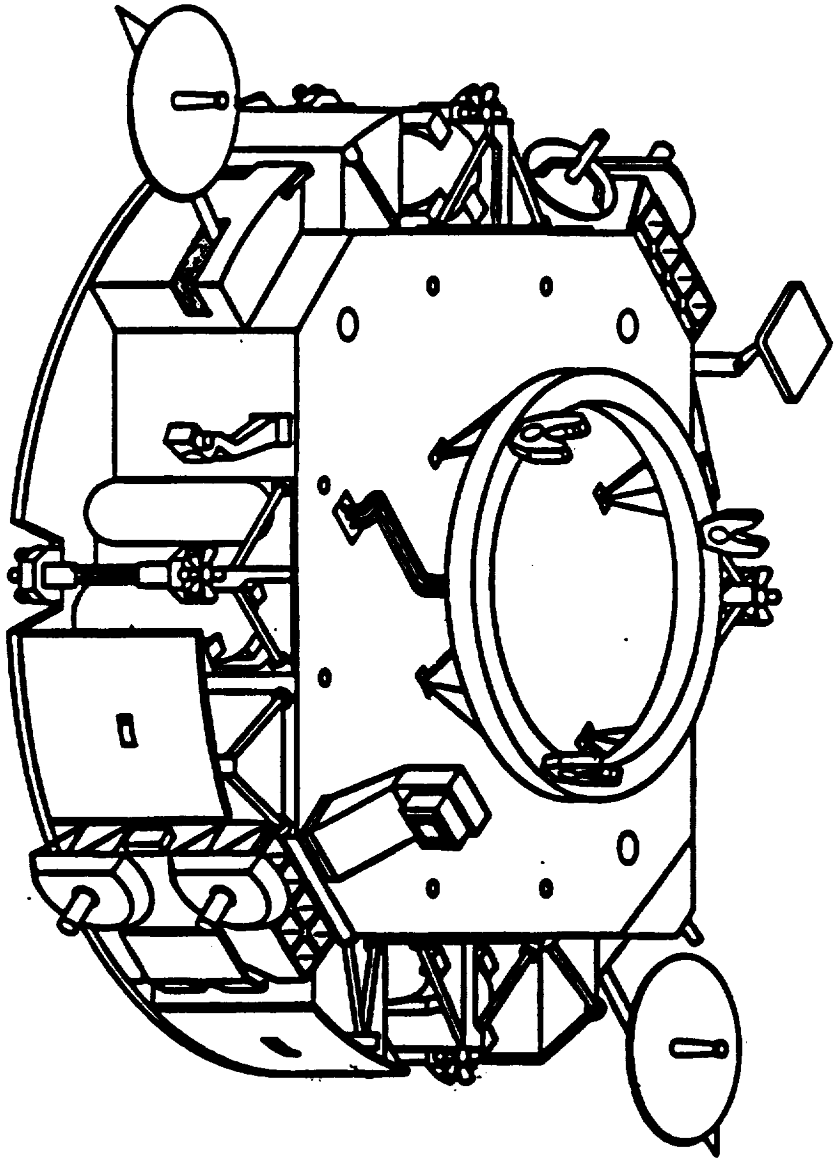


Figure 7-4. Three Point Docking Mechanism

C-2

The sensitivity of prospective payloads such as HST to the inertial loads requires such minimized contact conditions that the need for attenuation between the vehicles is eliminated. Therefore, the OMV docking mechanism has only one set of latches to effect capture and provide a rigid interface; the target interface consists of the required latch pins or grapple fixture. The latches provide a two-step capture sequence to ensure "fast" capture and "slow" latching, ensuring that minimized accelerations are imparted to the payload.

The TPDM was designed for manual docking, with emphasis on specific payloads, such as the HST. Modifications would be required to the TPDM, and possibly the OMV, for autonomous docking. For example, a capture latch proximity sensor would be required by the docking mechanism to trigger the capture latches at an appropriate range. In the future, additional mechanisms may be designed for AR&D with the OMV. These mechanisms may require attenuation at the interface to accommodate docking events in which more extreme contact conditions are imposed on the docking hardware.

7.3.1.3 OTV/Lunar Base Program Mechanism Descriptions

The missions proposed for development of a lunar base via SSF or a Transportation Node (TN) comprise four basic docking cases: 1) OMV plus crew module to SSF or a TN in LEO, 2) OMV to OTV or vice versa in LEO, with and without a payload, 3) OTV to the MLM following lunar ascent in lunar orbit with no crew transfer, and 4) OTV with crew module to the LOSF, MLM, or lunar-based reusable lander.

The missions may require several weeks or months in LEO or lunar orbit between docking events. This leads to a dormancy requirement similar to the MRSR mission. However, environmental conditions in LEO may be more destructive, depending on orbit and altitude.

7.3.1.3.1 Unmanned Applications

Case 2 focuses upon the need to mate two unmanned spacecraft for transport to and from SSF or a TN in LEO. Remote piloting may be readily available via SSF or a TN, depending on the rendezvous location. However, autonomous docking with manual backup may be safer. Case 3 concerns docking in lunar orbit between the unmanned OTV and an MLM for return to Earth. No transfer through the docking interface, crew or otherwise, is required. To minimize the propellant requirements for the MLM, the OTV is the active vehicle. Docking will be autonomous due to the possible loss of pilot proficiency during the duration of the mission. Additionally, a remote piloting console may be prohibitively massive, exceeding launch vehicle constraints.

7.3.1.3.2 Manned Applications

The first and last cases require crew transfer between the docking spacecraft using a crew module attached to the OMV or OTV. Therefore, the AR&D mechanism must allow both the establishment of a pressurized interface between the two vehicles and the necessary transfer operations. The requirements for case 4 will be the standard for any manned mission requiring docking beyond LEO, whether docking is autonomous or manually performed. In addition, the design of the AR&D mechanism for manned flights will take into account, where applicable, requirements for SSF and the IDM.

7.3.2 <u>Configuration Design Requirements</u>

7.3.2.1 Geometric Requirements

Geometric restrictions are a major factor in developing the configuration of a docking mechanism for a given spacecraft. The size, location, and type of mechanism used will depend on the spacecraft involved and the mission requirements.

7.3.2.1.1 Launch Vehicle Constraints

The docking mechanism will be designed to meet the payload requirements of the designated launch vehicle. This includes geometric restrictions due to the dimensions of the available payload volume. Possible launch configurations should be considered during the design of the docking mechanism.

7.3.2.1.2 Spacecraft Constraints

The docking mechanism will be positioned on the spacecraft such that a program-defined clearance exists around the interface to prevent structural damage to the vehicles during an abort or no-dock situation. The docking mechanism location and orientation will be considered in the spacecraft design such that the mechanism axes of symmetry are aligned with and as close as possible to a principle axis relative to the center of gravity (CG) of the vehicle. As the lateral distance between the CG and docking mechanism increases, the contact conditions become more extreme, necessitating increased mechanism strength and load-handling capabilities.

The docking mechanism may be designed such that a passageway between the docking interfaces allows the transfer of necessary crew or cargo, as applicable to the mission. The type of cargo to be transferred establishes the minimum size requirement for the passageway and, subsequently, the entire docking mechanism. For a manned mission, the mechanism must allow transfer of a crewmember in an extravehicular activity (EVA) suit. An unmanned mission may be more complicated since an automated transfer is required. The size of the cargo and the geometric envelope of the equipment necessary to clear the passageway and perform the transfer are factors which must be considered in the mechanism design.
The thermal control system design will not constrain spacecraft attitudes to maintain temperatures within operational and non-operational temperature limits.

7.3.2.2 Mass Properties

The mass and location of the docking mechanism will not cause a violation of the allowable range of locations for the CG of the spacecraft. A proper CG location is required for optimal attitude control and spacecraft stability, especially during aerobraking, re-entry, and docking. The mass of the docking system will be minimized to maximize the launch and payload carrying capabilities of propulsive elements.

7.3.2.3 Hardware Interfaces

The docking mechanism design will allow for integration of the necessary systems and subsystems to accomplish the AR&D task and support docked operations. This integration includes the interfaces between the docking mechanism and the host vehicle and the docked vehicle (umbilicals), as well as subsystems mounted directly on the docking mechanism.

During the docking task, required systems and subsystems may include video equipment, PROX OPS sensors, lighting, and any other necessary targeting and ranging hardware. The AR&D mechanism design will allow for structural mounting of such systems in their required locations. The mechanism design will provide the necessary connections to receive power from the spacecraft and deliver power to the above systems. Data links will be included between the docking mechanism and host vehicle to allow the transmittal of data from the mechanism mounted PROX OPS sensors to the spacecraft control systems, as well as for communication to Earth, SSF, etc. The data links will also allow the transmittal of data from systems monitors, such as the thermal control system and the attenuators, as well as from instrumentation for verifying interface status during all phases of the docking operation.

Communication and data links will be provided between the docked vehicles via automatic umbilical connections. Umbilicals will also be provided to enable pressurization of each vehicle by the other, and for fluids transfer, as required.

7.3.2.4 MRSR Configuration Requirements

Because of the long journey to and from Mars, minimization of the total mass of the mission is one of the major design objectives. The mass of the MAV docking mechanism will be minimized to maximize the launch capability from the Mars surface as well as the following on-orbit control capabilities.

A primary function of the docking mechanism for the MRSR mission is to effect the transfer of the SCA from the MAV to the ERV. Any obstructing equipment at the docking interface must be stowed or removed prior to this transfer. Therefore, the geometry of the SCA and the automated transfer mechanism, as well as that of the MAV and ERV, will be considered in the design of the docking mechanism.

7.3.2.5 OMV Configuration Requirements

As noted previously, two docking mechanisms have been designed for use with the remotely piloted OMV, the RGDM and TPDM. No additional mechanisms are planned at this time. The OMV interface requirement for the TPDM, and for any future mechanism for heavy payloads, is that the docking mechanism must attach to the OMV at the four bolt locations on the OMV face (figure 7-5). The OMV docking interface for AR&D, whether it is to be an upgraded TPDM or a new mechanism, must integrate the necessary cameras and sensors for docking, as well as automatic umbilical connections for power, data transfer, and fluids transfer between the OMV and payload. The RGDM is too small for cameras and relies on the OMV's camera system. Although the necessary umbilicals are integrated into the docking mechanism, the required sensors for AR&D must be attached to the OMV for RGDM use.

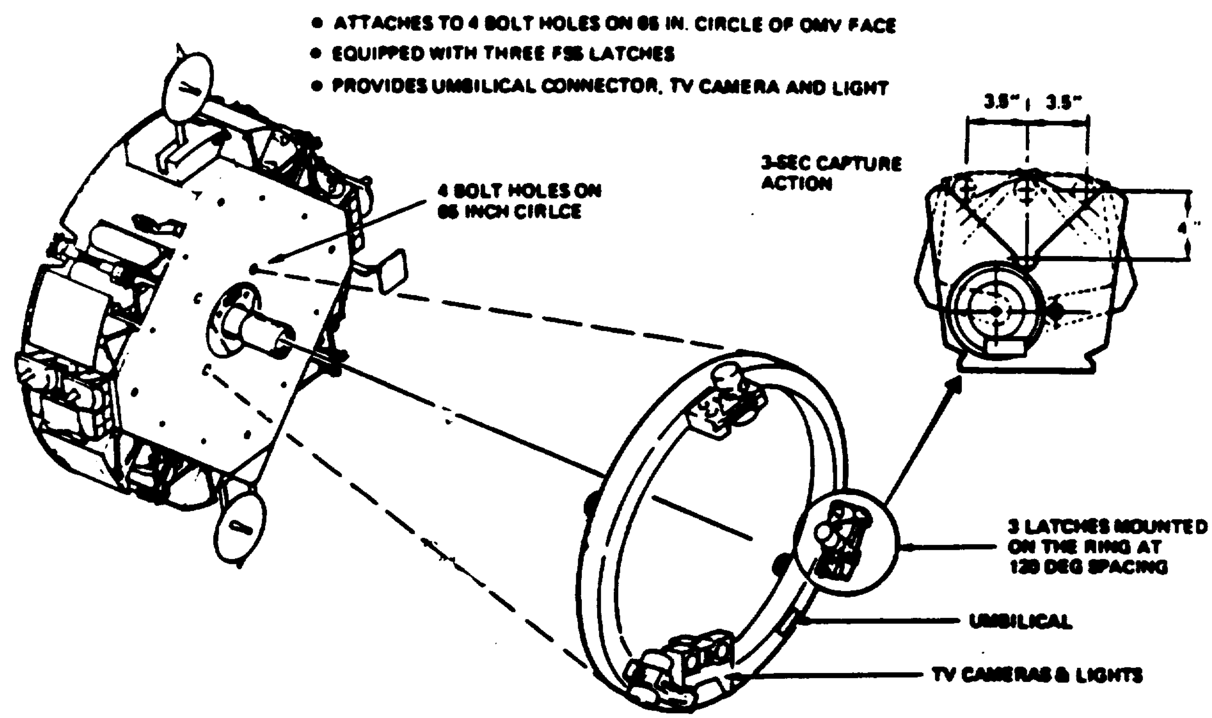


Figure 7-5. OMV Bolt Holes

ORIGINAL PAGE IS
OF POOR QUALITY

An OMV docking mechanism for AR&D applications will include the necessary autonomous systems to effect capture and structural latching or rigidization at the docking interface. The current mechanisms are manually operated, requiring pilot input to trigger the capture mechanism.

7.3.2.6 OTV Configuration Requirements

The configuration-specific requirements for the various OTV applications for development of a lunar base involve the use of multiple spacecraft or spacecraft combinations, commonly called stacks. Section 7.3.1.3 described the missions and the vehicles involved, including the OTV, OMV, SSF, TN, an MLM, a reusable lander, and the LOSF.

OMV to OTV docking, with and without a payload stack, (case 2) requires a mechanism similar to the TPDM, as described in section 7.3.2.5. OMV docking mechanism requirements for docking to SSF or the TN are indefinite since, at this time, it is not known whether the OMV and the OTV stack will be berthed or docked. Because the mating process for stack construction requires hangar operations, it is assumed that the prime mating method will be berthing. In the event that docking is required, it is assumed that the interface to be used will depend on the current mission (as described in section 7.3.1.3).

The mission involving delivery of an MLM to lunar orbit with subsequent MLM retrieval (case 3) requires a docking mechanism similar to that for the OMV. The mechanism must provide a rigid interface between the OTV and MLM, with no pressurized transfer required. The docking interface configuration is currently undetermined. However, the interface must allow for automatic umbilical connections for power, data transmittal, communications, and fluids and propellant transfer. Mechanism design will take into account geometric constraints imposed by the Earth launch vehicle and SSF or TN hangar dimensions for servicing and OTV stack mating. Due to the lack of an additional crew module, no crew transfer requirement is planned for MLM to OTV docking. In the event of an on-orbit failure of the MLM, docking with another manned vehicle for crew transfer may be required. Therefore, MLM docking mechanism design will include the requirement to allow for transfer of a crewmember in an EVA suit. A further requirement will be similarity to the IDM, where feasible, to allow for rescue operations by multiple spacecraft.

The docking mechanism for the final scenario (case 4) has configuration requirements similar to those for the MLM. The docking mechanisms may be identical in design, with the crew transfer capability for the MLM utilized only in an emergency situation. Propellant transfer is a firm requirement for this mission. Therefore, the crew module, LOSF, and MLM will carry the hardware necessary for the transfer.

7.3.3 Mechanical Design Requirements

During docking operations, the docking mechanisms must perform the following tasks: capture one another, negate the relative motion at contact, structurally connect the vehicles, establish a pressurized connection where applicable, and separate from one another when docked operations are complete.

7.3.3.1 Contact Conditions

When two spacecraft are docking, some relative state exists between them at initial contact. These docking port to docking port contact conditions are composed of the following parameters (figure 7-6):

1) lateral misalignment (Y_{DP}, Z_{DP})
2) closing velocity (XDOT)
3) lateral velocities (YDOT, ZDOT)
4) relative attitude (pitch, yaw, roll)
5) relative angular rates (p, q, r)

The contact conditions are a major driver in the design of the docking mechanism, defining the size, weight, and complexity of the hardware. The parameters which comprise the contact conditions, with the possible exception of closing velocity, would ideally be very small to optimize the docking mechanism design.

The attenuator size and type of latch depend greatly on the closing velocity at contact. Passive latches require some minimum amount of energy to ensure capture. Compensation for large lateral or rotational misalignments may use some of the kinetic energy required to effect latching, increasing the minimum closing velocity necessary. Active latches do not require a minimum closing velocity to effect latching.

The size, stroke, and type of attenuators depend on the maximum kinetic energy of impact which must be damped. Available volume or vehicle geometry may restrict the stroke of the attenuators, limiting the maximum allowable closing velocity for a given docking mechanism and vehicle. Relative lateral velocities and angular rates will also contribute to the attenuator performance requirements.

Vehicle mass and geometry must also be taken into consideration in the design of the attenuators. Due to kinetic energy limits of the hardware, more massive vehicles generally are limited to a lower closing velocity for a given docking mechanism. In addition, compensation for "jackknifing" may be required for some vehicle geometries and docking mechanism locations.

As an example, the allowable contact conditions for the ASTP are listed below:

1) maximum lateral misalignment = ±0.984 ft
2) closing velocity = 0.164 to 0.984 fps
3) maximum lateral velocity = ±0.328 fps
4) maximum pitch, yaw angular misalignment = ±7.0 deg
5) maximum roll misalignment = ±7.0 deg
6) maximum angular rates = ±1.0 dps

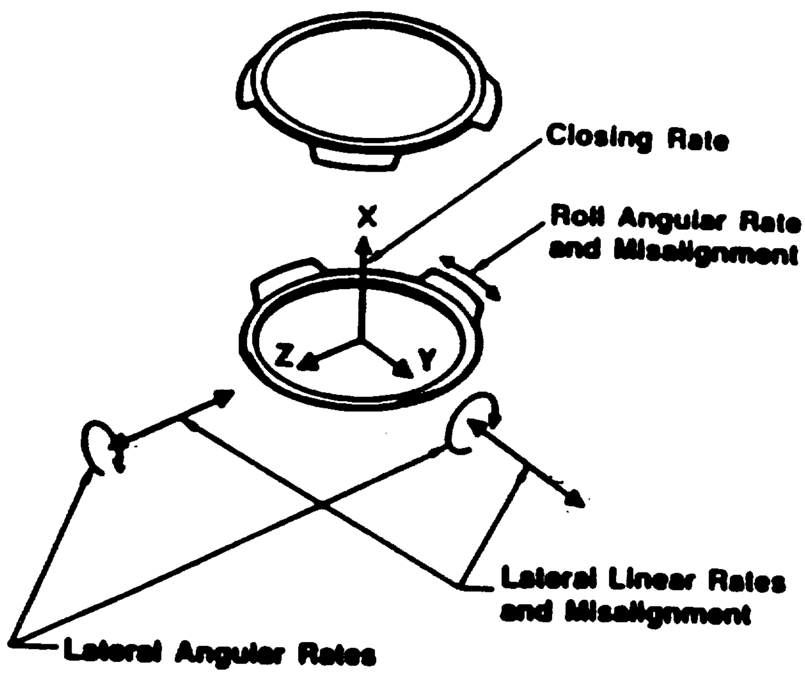


Figure 7-6. Docking Definitions

These numbers are listed for comparison purposes only. It is not reasonable to assume these contact conditions apply to all vehicles or missions since many factors influence the contact conditions. For a more detailed discussion of contact conditions, see Appendix A.

To ensure adequate mechanism performance, worst-case contact conditions must be clearly defined. It must then be guaranteed that these design contact conditions will not be exceeded. Otherwise, loss of the mission could result, depending on the docking mechanism design limits and the actual contact conditions. For AR&D, the relevant program will be responsible for providing the contact conditions, based upon vehicle geometries and mass properties, GN&C, and PROX OPS systems, for the design of a docking mechanism.

7.3.3.2 Interface Alignment

When the vehicles first "contact," some errors will exist in their relative lateral alignment and attitude. These errors must be accommodated by the docking interface to meet the limits of the capture envelope. The docking hardware will be designed in such a way that these lateral and angular misalignments are greatly reduced or eliminated before latching. For example, due to its shape, a probe and drogue mechanism will force alignment between the interfaces as they move together. An androgynous ring-and-guide system will also provide this advantage.

7.3.3.3 Capture

Capture of the target vehicle by the chase vehicle occurs after initial contact is made, when a sufficient (currently unspecified) number of capture latches are fastened to allow safe and efficient attenuation of the loads. The capture latches may be active (automated or manual) or passive. Active latches are triggered by proximity sensors or pilot input, whereas passive latches are triggered by the energy from the impact of the vehicles. If the conditions at contact are within the capture envelope, the allowable relative state from which capture latching may be effected, docking may continue as planned. If the contact conditions are outside the capture envelope, the chase vehicle will have to abort, reposition, and approach again. The capture latches will be structurally redundant such that expected dynamic loads can be successfully attenuated in the event of a latch failure. However, an automated latch release system will be considered in the event that an insufficient number of capture latches are triggered, causing the structural integrity of the mechanism or the vehicle to be threatened.

7.3.3.4 Attenuation and Structural Latching

7.3.3.4.1 Hard Dock

A "hard" dock is defined as mating when the relative motion of the vehicles at contact is such that attenuation is needed to prevent structural damage. Electrical and/or mechanical attenuators can be used to eliminate the relative motion between the vehicles, damping out the energy of impact to prevent damage to the docking hardware and the vehicles. The size and type of attenuators depend on the expected contact conditions and loads, geometry of the vehicles, mass properties, volume, time constraints, and reliability considerations. Some interface designs may also assist in removing lateral and angular errors, reducing the stroke length needed to attenuate the loads.

After attenuation, the docking interfaces will be brought into close proximity to permit structural latching and rigidization. This is necessary because the capture latches are designed only to allow attenuation, and are not certified to handle pressure loads, crew transfer, etc. To accomplish structural latching, one or both sides of the docking hardware must retract to bring the latches together. The structural latching will be automated, with manual backup capability where applicable. After structural latching is complete, all required umbilicals will be automatically engaged.

7.3.3.4.2 Soft Dock

In the case of a "soft dock," in which very little energy is imparted to the mechanism by impact, attenuators may not be necessary. This condition is more likely if the geometry of the vehicles is such that "jackknifing" is minimized. Only one set of latches may be necessary if attenuation is not required. Since docking mechanism extension and retraction would not be required, the capture latches may be rigidized. For example, the TPDM for the OMV has dual function latches, which form a rigid interface after capture has been effected.

7.3.3.5 Sealing

To allow pressurized transfer between vehicles, a seal must be formed between the two spacecraft following structural latching. Seal requirements will include the following:

a) the ability to resist adverse environmental effects while undocked, such as atomic oxygen in LEO
b) independence from the structural interface
c) the ability to maintain required atmospheric pressure with both vehicle hatches open
d) ease of leak detection; leaks will not be camouflaged in the interface
e) simplicity of design for ease in repair or replacement by EVA, where applicable
f) the ability to maintain seal integrity while subjected to loads such as jet firings, crew movement, internal pressure, etc.

7.3.3.6 Transfer Compatibility

For missions requiring transfer between vehicles, the hatch mechanisms will be automated. Manned missions will have manual backups for hatch operations. For an unmanned vehicle, the hatch mechanism and any transfer mechanisms will be designed in parallel with the docking mechanism to optimize mechanism capabilities and minimize mass and consumption of spacecraft volume. Transfer mechanism requirements for a spacecraft will be established by that vehicle's program, which may dictate any necessary AR&D mechanism design modifications.

7.3.3.7 Separation

Separation of two spacecraft with a pressurized connection normally consists of the following steps:

a) reinsert docking interfaces, if applicable
b) inspect hatches, then close and lock
c) depressurize space between hatches
d) inspect hatch seals
e) unseal docking interfaces
f) disconnect umbilicals
g) release structural latches
h) extend docking mechanism of one or both vehicles
i) release capture latches
j) separation maneuvers

All stages of separation will be accomplished autonomously, with manual backups where applicable. Steps a-e do not apply if no pressurized interface will be required. During all phases of operation, the hatch and seals must be capable of maintaining pressure within the vehicle. Indicators will be supplied to show the status of hatches, seals, pressure, umbilicals, and latches. Pyrotechnic backups for latch releases and emergency undocking will be considered in the design of the docking mechanism. A complete visual inspection, using cameras if necessary, will be mandatory after any emergency separation, prior to the next docking event. Servicing will be necessary after a pyrotechnic separation to replace the charges.

7.3.3.8 Multiple Events

A docking event is defined as initial contact through complete separation of the two spacecraft, including all operations in between. The use of the OMV and OTV in the proposed missions calls for multiple docking events within a short period of time. This may not allow regular maintenance between events (See section 7.3.7.4 Maintainability). An AR&D mechanism for multiple event applications will be designed such that a program-defined number of docking events may be performed without failure. An appropriate Factor of Safety (FS) will be applied to the expected service life. The capability will exist for a visual inspection of the docking interface and a self-diagnostic check-out between each docking even (See section 7.3.7.4 Maintainability).

7.3.4 Structural Design Requirements

The AR&D docking mechanism structure, when used in combination with the host spacecraft, will be of adequate strength and stiffness to resist without failure all imposed loads, both nominal and extreme, including ground testing, ground transport, launch, orbital operations, and normal and abort landings, as applicable.

7.3.4.1 Docking Mechanism Loads

A major factor in the design of a docking mechanism is the expected loading. The following loads and load combinations will be considered during design of an AR&D mechanism:

(1) fabrication/assembly
(2) testing
(3) transportation
(4) flight (lift-off, ascent, on-orbit, descent, re-entry, and landing, as applicable)
(5) emergency landing (if applicable)
(6) crew-applied loads
(7) pressure loads
(8) misalignment

The mechanism structure will be designed to the critical flight conditions for each mission, as identified by the responsible program.

7.3.4.1.1 Non-flight Loads

The non-flight loading conditions (items 1 through 3, noted above) will influence the structural design of the docking mechanism to the minimum possible extent. Where feasible, means will be devised such that loading conditions from hardware fabrication and assembly, handling, transportation, and storage do not cause an increase in the hardware weight over that required for flight.

7.3.4.1.2 Docking Event Loads

Loads are imparted to the docking mechanism during three phases of on-orbit flight: while the vehicles are docking, while the vehicles are docked and structurally connected at the interface, and while the vehicles are separating.

The loads on the hardware during docking operations are based upon the contact conditions, the vehicle mass properties and geometry, and the action of the attenuators. Most loads on the docking interface and immediate support structure should be limited by the attenuators. The design of the mechanism will be able to support the loads induced by the worst-case contact conditions.

After capture and attenuation, a rigid connection will be established if necessary. During this process, loads may be induced by the retraction of the docking mechanism and by the action of the structural latches engaging. The structural latches, seals, umbilicals, and transfer mechanisms will allow for all expected loads during docked vehicle maneuvers, including jet firings and crew activity.

During separation, the docking mechanism may be subjected to loads due to unlatching, extension of the mechanism, crew activity, and jet firings. It is suggested that both vehicles be in free drift during separation to prevent unnecessary loads on the capture latches. However, it is conceivable that auto-pilot mode will be necessary for certain operations. Loads induced during a pyrotechnic separation will not propagate from the mechanism and affect the primary vehicle.

The loads will be distributed throughout the structure. Effects of deformations, material nonlinearities, and temperatures on load distributions will be included in determining both the limit and ultimate load distributions in the structure, as required. Factors of Safety will be applied to account for uncertainties in load definitions, environmental effects, material properties, dimensional discrepancies, and any other factors that cannot be defined in detail sufficient for a loads analysis. As an example, the current SSF FS requirements are listed below:

Item	FS
General structure	1.5
Pressurized manned compartments	2.0
Pressure Vessels	1.5
Pressurized lines & fittings	
Less than 1.5 in diameter	4.0
1.5 in diameter or greater	2.0
Other pressure system components	2.5

The FS for AR&D mechanisms will be determined by the applicable vehicle program.

7.3.4.2 Allowable Mechanical Properties

Values for allowable mechanical properties of structural materials for the design environment will be taken from MIL-HDBK-5, MIL-HDBK-17, MIL-HDBK-23, and other sources approved by NASA. Where values are not available, they will be determined by analytical or test methods approved by NASA. Material "A" allowables from MIL-HDBK-5 will be used in all applications where failure of a single load path would result in loss of vehicle structural integrity. Material "B" allowables may be used in redundant structure where failure of a component would result in a safe redistribution of applied loads to the remaining load carrying structure.

7.3.4.3 Fracture Control

Regardless of the mission scenario, all docking hardware will meet all fracture control requirements for the governing launch vehicle. For payloads launched on the Space Shuttle, the applicable document is NASA-NHB-8071.1. Similar requirements are outlined for the Titan IV and other launch vehicles.

The AR&D mechanism structure will meet the appropriate fracture control requirements defined in NASA-NHB-8071.1 for on-orbit systems and hardware.

Safe-life design will be adopted for the AR&D mechanism structure to provide the capability of performing, without failure, four times the number of mission cycles expected in service (shown by analysis or by test through a rationally derived cyclic loading and temperature spectrum). A mission cycle is considered to be a complete docking event.

7.3.4.4 Docking Mechanism Strength

As a goal, the AR&D mechanism structure will be designed such that the failure of a single structural member will not degrade the strength or stiffness of the mechanism to the extent that the crew or mission is placed in jeopardy, or that a catastrophic failure results. Basic strength requirements are defined in MSFC-HDBK-505A, Structural Strength Program Requirements, including general requirements for a structural strength program of analysis and testing. The design will be verified by analysis, or analysis supplemented by tests, to show that the hardware meets the proper strength design requirements with sufficient margin of safety to ensure adequate strength, service life, rigidity, and safety of equipment and personnel at all times.

7.3.4.4.1 Limit Loads

The structure will be designed to have sufficient strength to withstand simultaneously the limit loads and other accompanying environmental phenomena for each design condition without experiencing elastic or plastic deformation beyond the structural design limits.

7.3.4.4.2 Ultimate Loads

The structure will be designed to withstand simultaneously the ultimate loads and other accompanying environmental phenomena without failure. No FS is applied to any environmental phenomena except loads. Structural deformations will not precipitate structural failure during any design conditions and environment at loads less than ultimate loads.

7.3.4.4.3 Margin of Safety

Margins of safety must be positive for all load conditions, including combined loads and stresses. However, for minimum weight, the structure will be designed for the smallest practical and permissible margin of safety greater than zero.

7.3.4.4.4 Damage Tolerance

The mechanism structure will be tolerant of damage from sources such as crew activities and impact from space debris, such that the mission will not be impeded or reduced in its objectives.

7.3.4.4.5 Creep

The design of the mechanism structure will preclude cumulative creep strain which may lead to rupture, detrimental deformation or creep buckling of compression members during the expected service life. Analysis will be supplemented by test to verify creep characteristics for the critical combination of loads and temperatures.

7.3.4.5 Commonality

The AR&D mechanism structure will employ standard fasteners and tools for structural and mechanical attachments. Commonality of structural interfaces for attached hardware will be maximized. ORUs will be used to the maximum practical extent.

7.3.4.6 Atmospheric Leakage/Sealing

For missions requiring a pressurized compartment, the design of the AR&D mechanism interfaces will include consideration of the sealing and atmospheric leakage requirements. A maximum atmospheric leakage level will be established for each spacecraft, which the AR&D mechanism must not degrade, for both the docked and undocked phases of the mission.

7.3.5 Environmental Compatibility Requirements

7.3.5.1 Natural Environment Design Criteria

The docking mechanism will be designed such that it meets all performance requirements for operations in the on-orbit or interplanetary natural environment. The natural environment includes, but is not limited to, atmospheric density and composition, plasma, charged particle and electromagnetic radiation, meteoroids and space debris, gravitational and magnetic fields, thermal effects and pressure. Relevant environmental data may be found in NASA TM X-64627, "Space and Planetary Environment Criteria Guidelines for use in Space Vehicle Development (1971 Revision)." Additional up-to-date data defining specific environments, such as the Mars surface or orbit, may be acquired directly in support of the mission.

The docking mechanism will be designed to meet all ground performance requirements with a design goal of no operational constraints relative to the natural environment conditions during assembly, check-out and launch.

7.3.5.2 Induced Environment Design Criteria

The docking mechanism will be designed to meet all performance requirements while operating in the induced environments of the assembly, check-out, launch, and orbital locations. The induced environment includes, but is not limited to, vibration, linear acceleration, temperature, radiation, plasma, reduced atmosphere, and contamination. The induced environment of a spacecraft is controlled by design to accommodate the mission requirements for that vehicle, not necessarily including docking mechanism operation. The design of any vehicle planned for on-orbit operations involving PROX OPS with other vehicles must take into account the induced environment of those vehicles. Some vehicles designed early in the U. S. space program and those designed by other international space programs may have more extreme induced environments than recent spacecraft. This is due to the previous lack of knowledge concerning some induced environments and to the applications of early spacecraft beyond the original design criteria.

7.3.5.3 Materials and Processes (M&P) Selection and Control Requirements

Materials used in the design and fabrication of hardware, including instruments, should be selected based on operational requirements as well as engineering properties. Materials selection guidelines include, but are not limited to, environmental compatibility, functional acceptability and suitability, extended life, technological maturity, manufacturability, inspectability, contamination characteristics, specific strength, availability, cost, safety, etc. SSP 30233 provides material selection requirements for SSF. However, other programs will meet the intent of SSP 30233. MSFC-HDBK-527/JSC 09604 includes materials rated with respect to safety and performance in certain areas, e.g., flammability, toxicity, and thermal vacuum stability. SSP 30233 calls out the basic test requirements for flammability, odor, offgassing, and fluid compatibility for materials. Specific program M&P requirements, such as for MRSR, would take precedence when in conflict with SSP 30233.

7.3.5.4 Thermal Control Design Requirements

The AR&D mechanism will be designed to accomodate the worst case thermal environments expected during its planned mission, whether in LEO, lunar or Mars orbit, or between planets. As a design goal, the thermal control design will not rely on specific attitudes or constrain attitude durations to maintain temperatures within operational or non-operational temperature limits. Surface optical coating selections will take into consideration temperature requirements of the mechanism.

7.3.5.5 MRSR Environments

NASA TM X-64627 provides environmental data for all phases of the MRSR mission from Earth orbit to Mars orbit. However, recent analysis has provided information with greater detail for the interplanetary and Mars orbit environments in JPL 642-520 Revision A, "Mars Observer Environmental Estimates, 16 Jan 1989." A portion of the docking mechanism is to be carried to the surface on the MAV, resulting in additional design requirements. The Mars atmospheric and surface environments have been further defined in NASA TM-100470, "Environment of Mars, 1988." The data available in these two documents should be used to provide the necessary design information to meet environmental compatibility requirements for a Mars surface mission, manned or unmanned.

The selection of materials for the MRSR mission will take into account the environments to be encountered and their effects on materials. Of primary importance is the ability to maintain operational capability after an extended dormant period. Materials such as lubricants and seals must be able to withstand the environments and maintain their physical and dynamic properties. Specific M&P requirements will be stated in the MRSR program requirements.

7.3.5.6 OMV Satellite Servicing Environment

The OMV is designed for LEO operations, individually and in conjunction with the Space Shuttle and SSF. The OMV, therefore, will need to manage the natural environment of LEO, as well as the induced environments of the Orbiter, SSF, international spacecraft (Hermes, etc.), any future U. S. spacecraft (free-fliers, co-orbiting construction facilities, etc.) and any satellites to be serviced or relocated. The natural environment for LEO is best defined in SSP 30425. This document will take precedence over any previous data. The induced environments of the NSTS and SSF have also been or are being documented. SSP 30000 defines the induced environment criteria for the SSFP. SSF is to be designed to control the induced environment, as will any future U. S. spacecraft. The Space Shuttle induced environment is described in NSTS 07700, Volume XIV, Appendix 1.

Materials selection for the OMV docking mechanism is of primary importance due to the possible extended durations of the OMV in LEO. The mechanism must withstand the atomic oxygen, ultraviolet radiation, and space debris environments experienced during flight in LEO. Any of the above may affect mechanism materials, degrading both the structural and mechanical operational lifetimes of the hardware.

7.3.5.7 OTV/Lunar Environments

The missions planned for development of a lunar base are similar to the MRSR and future manned Mars missions. The major differences are the duration of dormancy and the more benign lunar environment. The OTV is the prime vehicle in the lunar base program, with the requirement to function in LEO as well as interplanetary space and lunar orbit. In addition, portions of the hardware will be carried to the lunar surface on manned missions. The previously discussed environmental data for the SSFP (section 7.3.5.6) may be applied to the LEO phase of the mission. Data for the interplanetary and lunar phases of a mission is available from NASA TM X-64627, with additional data to be acquired prior to the initiation of base construction according to the program proposals described in JSC 23613.

The materials and processes requirements for the lunar base program using the OTV may be looked upon as a combination of the MRSR and OMV requirements. Materials must be able to withstand extended periods in LEO, interplanetary space, and the lunar environment, for multiple missions.

7.3.6 Control and Instrumentation Requirements

Sufficient instrumentation will be integrated into the AR&D mechanism to enable control of the mechanical systems and monitoring of the mechanism structure and mechanical systems. Specific requirements are stated below as minimums for the mechanism design.

7.3.6.1 Capture Latch Position/Status

The mechanism will have the necessary instrumentation to provide capture latch position and status to the proper control station.

7.3.6.2 Capture Latch Control

The capability will exist to send the commands to unlatch the docking capture latches via either vehicle. Priority will be set as to which vehicle will initially send the command. Since all satellites will not have the capability to command the unlatching, redundancy will be necessary for the OMV latches. Pyrotechnic backups will be available and controlled by the crew or controlling orbital ground station.

7.3.6.3 Mechanism Structural Position and Status

The instrumentation will exist to provide the position and status of the docking mechanism. This information will be available as required for the capture latch position and status (section 7.3.6.1).

7.3.6.4 Structural Latch Control

Each interface will have the independent capability to trigger the structural latches to latch or unlatch the vehicles. Priority will be set as to which vehicle controls the latches.

7.3.6.5 Structural Latch Status

The necessary instrumentation will be provided to indicate to the proper control stations that structural latching or unlatching has been completed successfully.

7.3.6.6 Umbilical Status

The necessary instrumentation will be provided to indicate to the proper control stations that the necessary umbilical connections have been completed successfully.

7.3.6.7 Pressurization Capability

For manned missions, the docking interfaces will provide the instrumentation and hardware to pressurize/depressurize or equalize pressure between the two vehicles. Each manned vehicle will have instrumentation provided to indicate the relative pressure across the docking hatch.

7.3.7 Quality Assurance

The quality assurance organization for any program utilizing AR&D will include the docking mechanism as part of the quality assurance activity, to satisfy the requirements established herein as well as those established specifically by the program. The quality organization will prepare a quality assurance plan to describe and verify compliance with the requirements herein and establish the capability of the hardware to satisfy the respective mission requirements in conjunction with the primary vehicle.

7.3.7.1 Safety

7.3.7.1.1 Manned Missions

All manned missions will require AR&D mechanism hardware design to comply with the safety requirements specified in MIL-STD-1512, NHB 1700.1, and NHB 1700.7. The docking mechanism will comply with the crew safety requirements as specified in NASA-STD-3000 for manned missions.

All pressurized structures and interfaces will be designed to leak-before-rupture criteria. This will be obtained via a proper fracture control program (section 7.3.4.3).

The application of pyrotechnics for separation will meet the safety requirements of AFR-127-100 and JSC 08060.

As a minimum, single-fault tolerance will be designed into the docking mechanism and its subsystems (i.e., fail-operational, fail-safe), but systems will also be restorable without termination of a mission. Noncritical system functions will be designed to fail in a safe mode with restorable capability where possible, within mission limitations (such as storage of replacement parts, etc.). The docking mechanism will preclude propagation of additional failures after failure occurs.

The design will specify as a goal that all bulkhead interfaces, hatches, and seals of which integrity is required to maintain pressurization will be accessible for visual inspection, maintenance, or repair by shirtsleeve crewmembers.

The AR&D mechanism will include safety interlocks, hardware and/or software implemented to preclude unsafe operations.

The AR&D mechanism and any Ground Support Equipment (GSE) will comply with the safety requirements of KHB 1700.7, NHB 1700.7, and NSTS-13830 if launched on NSTS.

7.3.7.1.2 Unmanned Missions

The docking mechanism and any GSE, including ground test and verification hardware, will comply with the safety requirements of KHB 1700.7, NHB 1700.7, and NSTS-13830 if launched on NSTS.

7.3.7.2 Reliability

A reliability program will be implemented for the AR&D mechanism and any proposed missions to meet the applicable requirements of NHB 5300.4 (ID-2).

Critical functions, related to crew safety or loss of mission for manned or unmanned vehicles, will have backup or redundant modes. For manned missions the need for EVA transfer of the crew, equipment or supplies will be reserved for emergencies and will not be used to downgrade criticality.

The design of an AR&D mechanism will include a Failure Modes and Effect Analysis (FMEA) activity. Proper identification will be made of any single-point failures which could create hazards to the crew or result in loss of a mission. Any credible failure modes which will degrade mission operations will be identified, and the capability for corrective measures will be implemented into the mechanism design. Any corrective measures for an unmanned mission will be highly automated, with minimum control inputs required from a ground or orbiting control station. For a manned mission, corrective measures may be automated, with the capability for manual interrupt by the crew.

The AR&D mechanism design will provide for the ability to handle one or more lengthy dormant periods between active operations. Minimization of mechanism sensitivity to the natural and induced environments will be applied and proven, especially for the lengthy dormant periods of flight, but also for active operation of the docking mechanism. Materials selection and application will be documented with respect to sensitivity to the environment and the dormancy requirement.

The AR&D mechanism will be designed and verified to handle the expected loads efficiently and for a program defined number of events.

7.3.7.3 Test and Verification

The AR&D mechanism design will include a verification program with the following general requirements applied.

7.3.7.3.a Each performance and design requirement specified herein will be verified by test, assessment, analysis, or similarity in support of certification of the design for operational use in a given scenario.

7.3.7.3.b Certification will be structured to verify the full range of design requirements under the specified environments. Specific attention must be paid to the dormancy requirements under these conditions. Where practical and technically sound, accelerated life test techniques will be utilized.

7.3.7.3.c Where redundancy exists in the design, each redundant path will be verified for all applicable design and performance requirements.

7.3.7.3.d All physical and functional interfaces between the docking mechanism hardware and host vehicles will be demonstrated as compatible and functional prior to flight.

7.3.7.3.e Off-limit testing will generally not be conducted. However, in specific instances it should be considered. These instances include, but are not limited to, when the design margins are relatively small with respect to off-nominal abort conditions, or when uncertainty exists in the definition of the design criteria. This may also apply in the event that a single-point failure mode exists.

7.3.7.3.f Testing of a unique design will be conducted at the point of hardware assembly which is considered by the program to be the most cost effective.

7.3.7.3.g All qualification test specimens will be processed through specified acceptance testing prior to any qualification testing.

7.3.7.3.h Where applicable, verification of maintainability, accessibility, and ease of operation will be demonstrated.

7.3.7.3.i The need for unique GSE and other support equipment during ground testing will be minimized.

7.3.7.4 Maintainability

7.3.7.4.1 Ground Maintenance

The AR&D mechanism will be designed to facilitate ground maintenance, inspection, and repair prior to launch. This will apply for all missions, including unmanned deep space missions such as MRSR.

7.3.7.4.2 On-Orbit Maintenance

The AR&D mechanism will be designed, where applicable, with the capability for on-orbit maintenance, inspection, and repair to the ORU level. The mechanism will be designed with distinct and definable interfaces for all systems and subsystems to facilitate removal, repair, and/or replacement on-orbit by automated means. The use of a RMS or Flight Telerobotic Servicer (FTS) system will be considered. Current SSS design tests include application of the FTS and other similar systems in conjunction with the OMV for LEO remote and automated servicing.

The use of EVA for docking mechanism maintenance of manned vehicles will be considered only in the event that automated maintenance or IVA is not practical or cannot correct the fault. Mechanism hardware will be compatible with current EVA tools and procedures.

The AR&D mechanism design will consider implementation of self-diagnostic capabilities which will include the ability to enable on-orbit functional verification of critical mechanical and electrical connections and functional paths. In the event of a path failure, the necessary work-around corrective measures will be fully automated with manual backup capability. Following on-orbit maintenance, the crew or controlling station will be capable of resetting the self-diagnostics to the primary functional path.

Non-critical mechanism subsystems will be designed to fail-safe such that the necessary repairs can be accomplished in a timely and effective manner within the constraints of the mission.

7.3.7.5 Redundancy

Redundancy will be applied to the AR&D mechanism to satisfy the stated reliability requirements. The actual redundancy requirements for all systems/components (except primary structure) will be established on an individual basis. The redundancy requirement for any single component will be no less than fail-safe.

The AR&D mechanism redundant functional paths will be located or protected such that an event which initiates loss of one functional path will not result in the loss of another redundant functional path. All safety and mission critical subsystems will be designed such that the failure of a single subsystem will not cause the loss of a redundant functional path.

In the event of a malfunction and possible loss of a functional path, information will be provided by the self-diagnostic systems to the crew, ground control station and/or orbital control station regarding the status of the affected system and available redundant paths/systems. Any corrective measures will be automated, with minimum control inputs required from the crew or controlling station.

7.4. APPLICABLE AND REFERENCE DOCUMENTS

The following documents were used in the preparation of this document, either as references or as sources of requirements. The latest issue of all applicable documents will be used.

7.4.1 Applicable Documents

7.4.1.1 Military

MIL-HDBK-5	"Metallic Materials and Elements for Aerospace Vehicle Structures"
MIL-HDBK-17	"Plastics for Flight Vehicles"
MIL-HDBK-23	"Structural Sandwich Composites"
MIL-STD-1522	"General Requirements for Safe Design of Pressurized Space Systems"

7.4.1.2 NASA

JSC 08060	"Space Station System Pyrotechnic Specifications"
JSC 30230	"Space Station Induced Environments Data Book"
KHB 1700.7 / S-100	"Space Transportation System Payload Ground SAMTO HB Safety Handbook"
MSFC-HDBK-527 / JSC 09604	"Materials Selection List for Space Hardware Systems"
NASA-STD-3000	"Man-Systems Integration Standards"
NASA TM-100470	"Environment of Mars, 1988"

NASA TM X-64627	"Space and Planetary Environment Criteria Guidelines for Use inSpace Vehicle Development (1971 Revision)"
NHB 1700.1	"NASA Safety Manual - Portions Applicable to Space Volume I Station Flight Hardware and Processing"
NHB 1700.7	"Safety Policy and Requirements for Payloads Using the Space Transportation System"
NHB 5300.4 (ID-2)	"Safety, Reliability, Maintainability and Quality Provisions for the Space Shuttle Program"
NHB 8071.1	"Fracture Control Requirements for Payloads Using the National Space Transportation System"
NSTS 07700 Volume XIV, App. 1	"System Description and Design Data - Contamination Environment"
NSTS 13830	"Implementation Procedures for STS Payloads System Safety Requirements"
SSP 30000	"Space Station Program Definition and Requirements"
SSP 30233	"Space Station Requirements for Material and Processes"
SSP 30425	"Space Station Program Natural Environment Definition for Design"
SSP 30441	"Interface Requirements Document Orbital Maneuvering Vehicle / Space Station"
no number	"AR&D Project Systems Requirements Document" (Draft), August 1989
no number	"International Spacecraft Docking Study" (Preliminary Draft / December 1987) JSC
no number	"Orbital Maneuvering Vehicle / Hubble Space Telescope Interface Requirements Document" (Draft)
no number	"User's Guide for the Orbital Maneuvering Vehicle" (Draft) MSFC

7.4.1.3 Other

AFR-127-100	"Explosives Safety Standards"
JPL 642-520	"Mars Observer Environment Estimates, 16 January 1989"

7.4.2 Reference Documents

7.4.2.1 NASA

NASA SP-4002	"Project Gemini Technology and Operations: a Chronology"
NASA SP-4209	"The Partnership: a History of the Apollo-Soyuz Test Project"
NASA TM-100470	"Environment of Mars, 1988"
SSP 30000	"Space Station Program Definition and Requirements"
SSP 30441	"Interface Requirements Document Orbital Maneuvering Vehicle / Space Station"
no number	"Orbital Maneuvering Vehicle / Hubble Space Telescope Interface Requirements Document" (Draft), MSFC
no number	Schliesing, John A. ,"Dynamic Analysis of Apollo-Salyut/Soyuz Docking," 1972, JSC
no number	"User's Guide for the Orbital Maneuvering Vehicle" (Draft), MSFC,July 1989

7.4.2.2 Other

JPL 642-520	"Mars Observer Environment Estimates, 16 January 1989"
JPL 650-1-100	"MRSR Program Requirements" (Draft), May 1989
no number	Johnson, N.L.,"Handbook of Soviet Manned Space Flight", Science and Technology Series, Volume 48

7.4.3 List of Additional AR&D Documentation:

TRW 89:W480.1-56, "Review of Satellite Servicer System Missions," dated 23 March 1989.

TRW 89:W480.1-67, "Mission Scenario Assessment - Autonomous Rendezvous and Docking," dated 6 April 1989.

TRW 89:W480.1-75, "Mars Rover and Sample Return Kick-Off Meeting," dated 12 April 1989.

TRW 89:W480.1-113, "Review of Satellite Servicer System Flight Demonstration System Requirements Document," dated 22 May 1989.

TRW 89:W480.1-115, "Assessment of AR&D FY 90 Project Plan," dated 1 June 1989.

TRW 89:W480.1-129, "Results of AR&D Intercenter Working Group Meeting," dated 13 June 1989.

TRW 89:W480.1-177, "Review of Satellite Servicer System Flight Demonstration Program Pre-Phase B Study - Final Report," dated 7 August 1989.

TRW 89:W480.1-186, "First Draft of Autonomous Rendezvous and Docking (AR&D) System Requirements Document," dated 15 August 1989.

TRW 89:W480.1-196, "Trip Report - AR&D Intercenter Working Group Meeting," dated 28 August 1989.

7.5 Definitions

Abort Termination of operations, especially due to equipment failures or malfunctions. Does not necessarily imply loss of mission.

Androgynous Interface A non-polar interface; one which can physically join with another of the same design.

Attenuation Damping of loads and dissipation of kinetic energy of the docking mechanism caused by vehicle contact.

Autonomous Operating independently of external control; self-contained.

Capture Fastening of the capture latches after initial vehicle contact is made; initial connection between the docking interfaces to allow attenuation and establishment of a rigid connection.

Captures Envelope Allowable relative state between docking interfaces from which capture latching may be effected.

Chaser Spacecraft The spacecraft that performs the maneuvers in rendezvous and proximity operations.

Closing velocity Relative rate at which the docking interfaces are moving together. Includes only the component of velocity parallel to the docking mechanism centerline of the target vehicle.

Commonality Common and non-common hardware and software which are categorized as follows:

Common Physically and functionally identical with the same part number

Interchangeable Same function; mechanically compatible interface with different internal parts

Compatible Capable of functioning and operating effectively when integrated with other elements

Contact Conditions Relative state between docking interfaces at initial vehicle contact. Components include: lateral misalignment, closing velocity, lateral velocity, relative attitude, and relative angular rates.

Critical Function A single failure point where a hardware item's function cannot be checked out in orbit, loss of the function is not readily detectable by the flight or ground crew, or loss of the function is not restorable on-orbit.

Deadband Guidance, Navigation & Control (GN&C) software feature limiting the maximum allowable angular errors of a vehicle relative to a given attitude. Sets "hard" limits on attitude errors by causing RCS jets to fire when the maximum allowable attitude error is exceeded.

Docking The physical joining of two spacecraft wherein their relative velocity brings the vehicles' docking interfaces into contact.

Docking Interface The area of contact between two docking mechanisms.

Docking Medhanism A device that performs functions to connect one spacecraft to another in a docking operation.

Dormancy Period of quiescence between docking events.

Factors of Safety (FS) FS are multiplicative constants applied to maximum expected or limit loads that occur during any phase of the hardware use, from manufacture throughout its operational life, to account for uncertainties in load definitions, materials, properties, dimensional discrepancies, etc.

Fail-Operational Having the ability to sustain failure and retain full operational capability.

Fail-Safe Having the ability to sustain a failure and retain the capability to successfully and safely complete the mission.

Failure The inability of a system, subsystem, component, or part to perform its required function within specified limits, under specified conditions for a specified duration.

Fracture Control Program wherein mechanism design, materials selection, manufacture, and testing deal with the understanding and prevention of flaw propagation leading to catastrophic failures.

Function A separate and distinct action required to achieve a given objective, to be accomplished by the use of hardware, computer programs, personnel, facilities, procedural data, or a combination thereof; an operation a system must perform to fulfill its mission or reach its objectives.

Gravity Gradient Tendency for an object in orbit to move to a stable position due to the force of gravity on the mass properties of the structure.

Ground Support Equipment All Earth-based personnel and equipment required for support of operations during manufacture, testing, and operations.

Guided V-Bar Proximity operations technique by which the chase vehicle approaches the target vehicle using software to calculate the effects of orbital mechanics on relative position. Intended to minimize jet firings, and consequently, plume impingement and fuel consumption.

Hard Dock Contact when the relative motion between the vehicles is such that attenuators are required to safely dissipate the kinetic energy between the vehicles.

Jackknifing Situation caused by momentum, in which the vehicles continue to move after interface contact, pivoting about the point of contact. Most likely when the docking ports are offset from the vehicle centers of gravity.

Lateral Misalignment Relative lateral offset between the centerlines of the docking interfaces.

Lateral Velocity Component of total velocity normal to the centerline of the passive vehicle's docking interface. The rate of change of lateral alignment.

Lift-Off Mass The mass which must be boosted from a planet's surface. Includes mass of the vehicle(s), crew, payload, and fuel.

Limit Load The maximum load expected from all design conditions and operations.

Maintainability A comprehensive set of design characteristics which enables an item/system to be replaced, repaired, or restored to a specified operational condition within a specified time period and within specified resources.

Margin of Safety (MS) A ratio of the excess strength to the required strength

$$MS = \frac{\text{Allowable Stress (or Load)}}{\text{FS x Applied Limit Stress (or Load)}} - 1$$

Operating Cycles The cumulative number of times an item completes a sequence of activation and return to its initial state; e.g., a switched-on/switched-off sequence, a valve- opened/valve-closed sequence, a tank pressurized/depressurized, or Dewar cryogenic exposure/drain.

Orbital Mechanics The effect of perturbations on orbiting bodies. Includes effects of gravity and of other orbiting bodies on orbital altitude, inclination, and shape.

Orbital Replacement Unit (ORU) The lowest level of component or subsystem hardware that can be removed and replaced on location under orbital conditions.

Proximity Operations (PROX OPS) Maneuvers of two spacecraft in close proximity following rendezvous and/or immediately after vehicle separation.

Redundancy The existence of more than one means for performing a given function.

Relative Angular Rates The rate of angular rotation of the active vehicle's docking port relative to that of the passive vehicle. Rate of change of relative attitude.

Relative Attitude The angular misalignment between the docking interfaces of two vehicles.

Reliability A characteristic of a system, or an element, thereof, expressed as a probability that it will perform its required functions under defined conditions at designated times for specified operating periods.

Rendezvous The process of bringing two spacecraft into approximately the same orbit. Includes coarse maneuvers which cannot be approximated linearly.

Safety Freedom from chance of personnel injury or fatality, and damage to or loss of equipment or property.

Safe-Life Predicted service life of a structural component based on conventional fatigue analysis.

Separation The act of undocking two spacecraft. Includes proximity operations to move the vehicles out of close proximity.

Single Point Failure A single item of hardware, lacking redundancy, the failure of which could lead directly to loss of life, mission, or critical mission function.

Single Fault Tolerance Having the ability to function after the failure of any single component. Also, the subsequent failure of any single component will not cause damage to life or spacecraft.

Soft Dock Contact when the relative motion is such that attenuation of the loads is not necessary to prevent hardware damage

Station-Keeping The act of maintaining the position of one vehicle relative to another.

Target Spacecraft The non-maneuvering spacecraft in rendezvous and proximity operations.

Umbilicals Lines for transfer of power, data, fluids, or any other substance from one vehicle to another. To be an integral part of the docking mechanism.

Unrecoverable A vehicle is considered unrecoverable if it can not be retrieved through docking, berthing, or extra-vehicular manuevers - usually due to high angular rates.

Section 7
Appendix
Docking Contact Conditions:
Factors and Concerns

Appendix A
Docking Contact Conditions:
Factors and Concerns

A. Docking Contact Conditions Factors

A.1 Lateral Misalignment

Lateral misalignment between the docking interfaces at contact is driven by the following vehicle/task characteristics:

A.1.1 The targeting system, whether laser, rendezvous radar with transponder, image processor, or some other device, will be a major factor. The more accurate the targeting device, the smaller the lateral misalignment will tend to be. Therefore, the selected device should provide accurate relative position data throughout PROX OPS. Since failure of the targeting device could result in unacceptable lateral misalignment between the docking interfaces and probable loss of the mission redundancy may be necessary for AR&D, especially for unmanned missions. For a manned mission, a pilot can control lateral alignment given the proper optical alignment hardware with a visible target or targets on the target vehicle.

A.1.2 The configuration of the RCS for each vehicle may affect the lateral alignment by limiting translational and rotational controllability. The total number, location, and orientation of the jets, the pulse size, and the allowable fuel use may restrict maneuverability during the docking approach.

A.1.3 In conjunction with the RCS limitations, the GN&C characteristics of both vehicles may be a driving factor. A "cooperative" target, one with attitude hold capabilities, will reduce lateral misalignment at the docking interface. A target maintaining relative attitude will not have large angular rates or lateral velocities, minimizing the effects of imperfect controllability due to RCS limitations or of time lag in operations from a control center. A "cooperative" target will also help to minimize lateral velocities in the chase vehicle, since fewer compensating jet firings will be necessary to maintain relative position.

A.1.4 The geometry of both vehicles must also be considered, especially if the docking interfaces are not aligned with their respective vehicle CGs. Relative attitude errors will cause the lateral misalignment to increase as the offsets between the docking interfaces and vehicle CGs are increased. This effect is compounded if the lateral alignment target is not centered on the docking port of the target vehicle.

A.2 Closing Velocity

The nominal closing velocity is the velocity which is targeted by the chase vehicle. The range or limits of the nominal closing velocity will be defined by docking mechanism requirements (specifically the capture mechanisms) and the mission scenario. Errors in achieving the targeted velocity may be introduced by vehicle geometry, the GN&C software, and RCS configuration.

A.2.1 The docking hardware, including latching devices and attenuators, will define the range of the nominal closing velocity. Passive latches may require some minimum closing velocity to supply the energy needed to ensure capture. Compensation for large lateral or rotational misalignments may use some of the kinetic energy needed for passive latching, increasing the necessary minimum closing velocity. The design of the attenuators (electrical or mechanical) defines the maximum kinetic energy (maximum closing velocity) that can be dissaipated for given vehicle configurations. Available volume or vehicle geometry may restrict the stroke of the attenuators, limiting the maximum allowable closing velocity for a given docking mechanism.

A.2.2 The relative mass and geometry of both vehicles must also be taken into consideration with the attenuators. Due to kinetic energy limits of the hardware, more massive vehicles generally are limited to a lower closing velocity for a given docking mechanism. In addition, if the docking ports are offset from the vehicle CGs, the momentum of either (or both) vehicle(s) may cause "jackknifing", a situation where the vehicles continue to move after interface contact and capture, pivoting about the docking interface. This effect grows more pronounced as the CG offset and/or the closing velocity, and the resulting momentum, are increased.

A.2.3 Minimizing sensor errors will increase the accuracy in achieving the nominal targeted closing velocity at docking interface contact. The sensors used for lateral alignment may also be used for ranging. Once again, failure of the device may result in an unacceptable closing velocity (too fast or too slow) and possibly the inability to dock. Therefore, redundancy of this device is necessary for AR&D unmanned missions. A pilot should be able to control the closing velocity within allowable limits, given proper visual information of the target vehicle.

A.2.4 As with lateral alignment, GN&C software and RCS hardware configurations may restrict pulse size and maneuverability, limiting the accuracy with which the nominal closing velocity may be attained.

A.2.5 A "cooperative" target will aid in minimizing errors, increasing the probability of achieving a closing velocity within the acceptable range.

A.3 Lateral Velocity

Lateral velocity between the docking interfaces depends on the following factors:

A.3.1 The GN&C software and the location and orientation of the RCS jets determine the maneuvering characteristics of each vehicle. The firing of each jet or combination of jets imparts some lateral velocity to the vehicle. This velocity is determined by the jet configuration and size, the number of jets fired, and the duration of each firing. The maximum lateral velocity induced by RCS activity is defined by the expected maneuvers during the final stages of an approach.

A.3.2 The rendezvous and PROX OPS technique used can also have an effect on lateral velocities. For example, if lateral alignment is maintained during the final stages of an approach, lateral velocities should remain fairly low. If, however, a guided V-bar approach is used, a significant lateral velocity is to be expected in one axis (toward the center of the planet). An approach along the R-bar could also cause lateral velocities due to orbital mechanics. The main factors in determining approach technique are vehicle geometry and time and fuel requirements.

A.3.3 The geometries of the vehicles may affect the lateral velocity if the docking ports are not centerlined. If this is the case, the CGs of the vehicles will be in slightly different orbits when the docking interfaces are aligned. As the offset increases, the effects of orbital mechanics grow more pronounced. Orbits of different inclinations will induce relative velocities in the Y direction; the vehicles will tend to move away from each other during half of the orbit and toward each other during the other half. Left uncorrected, the paths of the vehicles would cross twice in each orbit. Orbits of different altitudes will induce relative velocities in the Z direction (docking port frame). If they are maintaining the same velocity, the vehicles will tend to separate; this tendency increases as the difference in altitude increases.

A.4 Relative Attitude

Maximum relative attitude errors will be determined almost exclusively by the GN&C software (or lack thereof). For AR&D mechanism design, it is assumed that the chase vehicle will have some form of attitude control to attempt docking. Redundancy shall be provided to prevent loss of attitude control by the chase vehicle. For minimization of contact conditions, the target vehicle should have some form of attitude control, resulting a "cooperative target". However, in the event of no active attitude control, a "non-cooperative" target will allow docking under certain flight conditions. The factors concerning relative attitude error for docking to a "non-cooperative" and "cooperative" target will be discussed in the following sections.

A.4.1 Non-Cooperative Target

If the target vehicle is "non-cooperative", i.e. it has no active attitude hold capabilities, two possible situations exist:

A.4.1.1 If the target vehicle is in a stable orbit, it will eventually reach a "stable" gravity gradient position. With an appropriately situated docking interface, mating may be possible with allowable contact conditions.

A.4.1.2 Due to plume impingement from the chase vehicle, an unstable orbit, or inappropriate target vehicle geometry, the target vehicle may begin to tumble slowly and deviate from the "stable" gravity gradient attitude. Potential for docking in this situation would be marginal at best, especially if the rotational rates are not constant. An approach must be carefully timed, and the docking mechanism must be able to withstand large variances in all parameters of the contact conditions, creating the need for a larger and heavier docking mechanism.

A.4.2 Cooperative Target

If the target vehicle is "cooperative", attitude errors will be based on the following:

A.4.2.1 The magnitude of the attitude deadbands for each vehicle, combined with the errors from the Inertial Measurement Units (IMU), define "hard" limits on the maximum attitude errors. IMU errors are generally small compared to the vehicle deadbands and can be quantified for a given system. Small deadbands are desirable from a docking mechanism standpoint, but fuel use increases as the deadbands are decreased. The deadband settings can be optimized in relation to fuel use and docking mechanism size limitations.

A.4.2.2 In addition, the RCS system on the target vehicle should have redundancy to assure attitude control. The degree of redundancy may vary depending on mass requirements, such as for the MAV mass minimization to maximize lift-off capability.

A.4.3 ACS and Vehicle Mass Properties

The mass properties of a vehicle may cause it to "hang up" on a particular deadband or set of deadbands. Due to orbital mechanics, the vehicle tends to rotate toward its gravity gradient position. Although RCS jets fire to keep from "overshooting" the deadband, orbital mechanics return the vehicle to the deadband, causing the RCS to fire again. If the docking mechanism is designed to the ACS deadbands, this behavior should not prohibit the ability to successfully dock.

A.4.4 Relative Attitude

Active attitude control may significantly affect the other docking contact conditions. When a vehicle reaches a deadband, jets will fire to reverse the rotational rate about that axis. This can drastically affect the lateral alignment and relative velocities of vehicles having susceptible geometries (i.e. those where the docking port is offset from the vehicle CG in the docking axis). The effect still exists when the docking port and vehicle CG are aligned, but is not so pronounced. For a given geometry, the effect of deadbands on the other contact conditions can be quantified.

A.5 Angular Rates

Maximum relative angular rates depend on the same parameters as attitude errors (GN&C software, orbital mechanics, and plume impingement), as well as RCS locations and orientations. The factors affecting relative angular rates for "non-cooperative" and "cooperative" targets will be discussed in the following sections with reference to relative attitude.

A.5.1 Non-Cooperative Target

If the target vehicle is "non-cooperative", the following will affect the angular rates:

A.5.1.2 If the target vehicle is in a gravity gradient position, an approach can be attempted. The chase vehicle may have angular rate limits, which are similar to deadbands; when the vehicle reaches a rate limit, jets are fired to reverse the direction of the rate. This puts a "hard" limit on the chase vehicle's rates, but does not benefit the target vehicle. The jet activity of the chase vehicle may induce rates in the target vehicle due to plume impingement, causing it to tumble. After this point, relative rotational rates cannot be guaranteed, possibly resulting in the inability to successfully dock.

A.5.1.2 A rapidly tumbling "non-cooperative" target should be considered unrecoverable. The maximum allowable rotational rate for the target vehicle will be defined primarily by the capabilities of the autonomous docking software and the corresponding sensors on the chase vehicle.

A.5.2 Cooperative Target

If both vehicles have attitude control capability, the rate limits may be set, thereby insuring that maximum relative rates will not be exceeded. As with deadbands, rate limits may also affect the other contact conditions. In order to ensure adequate mechanism performance, worst case contact conditions must be clearly defined. It must then be guaranteed that these design contact conditions will not be exceeded. Otherwise, loss of the mission could result, depending on the docking mechanism design limits and the actual contact conditions.

SECTION 8
DEVELOPMENT OF A VIDEO-BASED AUTOMATIC RENDEZVOUS AND DOCKING SYSTEM
(Prepared by Marshall Space Flight Center)

8.1 INTRODUCTION

This section focuses on the hardware and software design of a video-based AR&D system. An automated system will generally have to deal with one of two situations: cooperative targets that have been designed for easy automated docking, and uncooperative targets that were designed before automation in this field was a serious consideration. This section addresses docking with cooperative targets only. The information presented here was provided by MSFC.

8.2 PAST WORK IN AUTOMATIC DOCKING

In the past there have been both software and hardware studies of automatic docking using this and alternative sensing techniques. In the 1982-1984 period, Martin Marietta of Denver conducted a three-part study of automatic docking under contract to NASA/MSFC. In phase 1 (1) a broad comprehensive survey of several techniques was conducted, from which the three most promising were chosen for further work. These three were evaluated using software simulations, and the most successful was found to be an approach which used an active three-point target synchronized with the video camera by a radio command link. An in-depth evaluation of its capabilities was performed by Dabney using extensive software simulations, which established its dynamic performance envelope and verified the control scheme (2). A hardware demonstration of the system was performed at Denver under phase 2 (3), which validated the concept of synchronizing the flashing target with the video. Further improvements were made under phase 3 (4), which included, the use of Kalman filtering for the entire state vector.

8.2.1 Mobility Base/Air-Bearing Facility

The large scale simulations took place at MSFC's air-bearing facility in the flight robotics laboratory. The facility has a "flat floor" area 44 x 86 on which the floor height varies less than 1/1000th of an inch between any adjacent square foot. The mobility base used for the tests is an air-bearing vehicle. It weighs 4500 pounds but floats on three air-bearings to give an almost frictionless base on which to move. The vehicle is propelled by four banks of six thrusters (fig. 8-1), each of which delivers three pounds of thrust. This results is in the vehicle having roughly the same amount of control authority as the current OMV configuration. The thrusters and air-bearings are fed by tanks of compressed air on board the vehicle. A PDP 11/34 computer is on board to implement the thruster firing logic and timing. The inputs to the PDP come through an RF modem link on board the vehicle, fed by an RF modem in an engineering station off the flat floor. That modem can be connected to a terminal or to the serial port of a computer. All of these on-board items allow the mobility base to be completely autonomous.

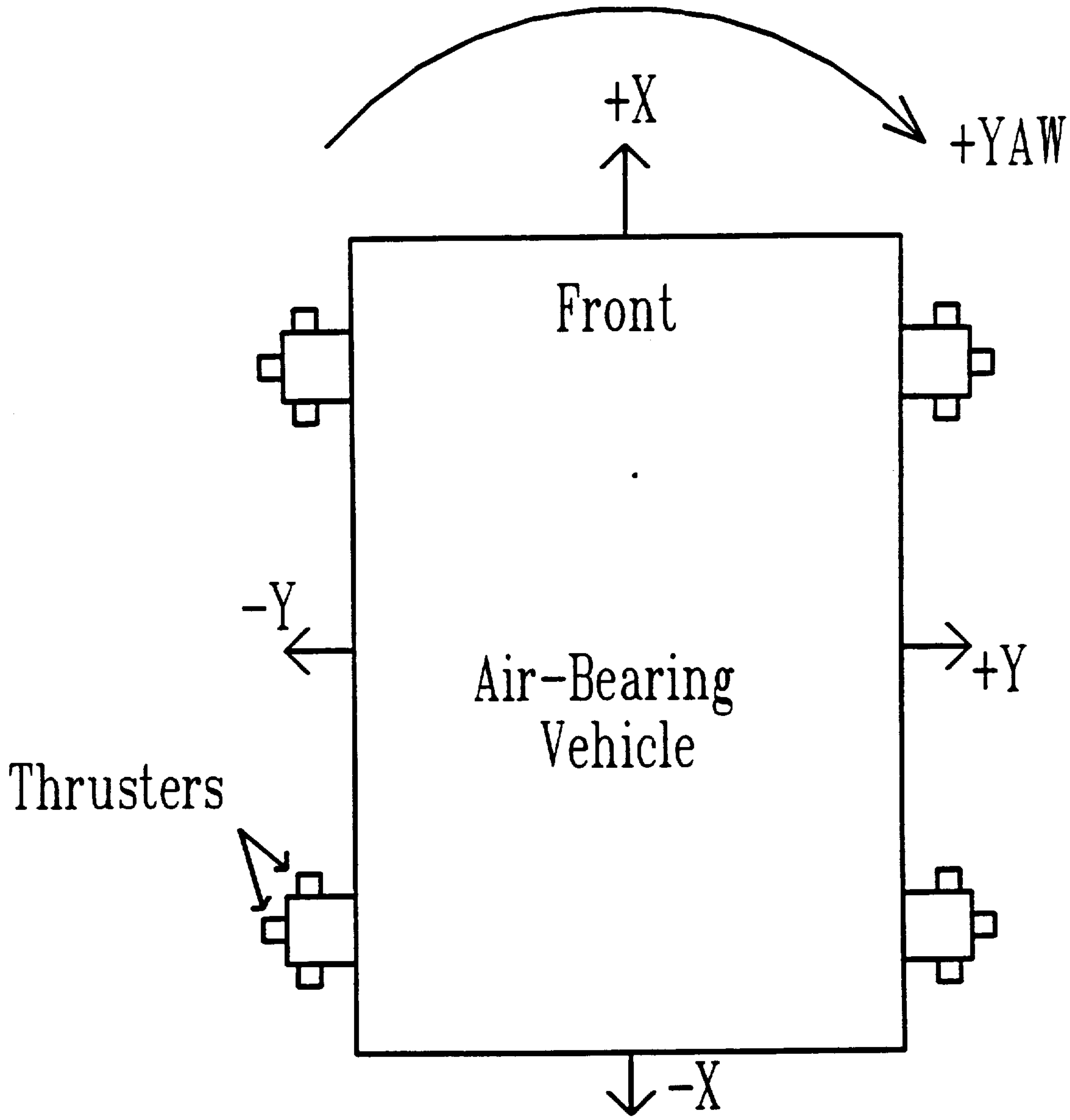


Figure 8-1. Mobility Base

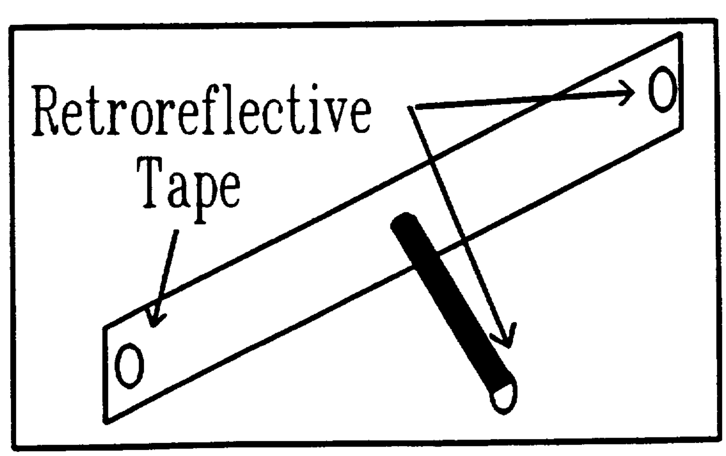


Figure 8-2. Docking Target

The flight robotics laboratory also has an eight DOF overhead crane located above the flat floor. It can travel the entire length of the floor and can be run both manually and by computer. A VAX controls the crane and contains dynamic models of the OMV, including the orbital mechanics. This crane will be used in the future for actively docking the mobility base to a target that can move in all six DOF's.

8.2.2 Docking Target Geometry

The docking target employed in this system is a three-point design, adapted from the standard RMS target found on many present-day spacecraft. Three points are the minimum needed for six-DOF relative state determination, with a fourth spot added to give complete knowledge of right-side-up versus upside-down. Any additional target points would be redundant. The target points consist of one-half inch squares of retro-reflective tape, one at each endpoint of the centerline, and one on the tip of the central post (figure 2). Attitude and position of the target relative to the sensor are computed from the two-dimensional coordinates of the three spots as tracked by the charge injection device (CID) image processor. The basic equations needed for these calculations can be found in reference 1. These equations were derived under the assumption that the sensor would be mounted on the chase vehicle, and the target mounted on the stationary vehicle. Due to physical limitations of the mobility base used in this demonstration, it was deemed necessary to reverse this arrangement, and the equations have been modified accordingly.

8.2.3 Video Sensor

The sensor portion of this automated guidance and docking system is built around a CID. It is a 256x256-pixel integrated circuit manufactured by General Electric. The chip is mounted in a case that has a thermo-electric cooler which keeps the chip at zero degrees Celsius to keep the noise at a minimum. The software controlled electronics around the chip allow the reading of any group of 4x4 pixels. Off chip transistors and op-amps amplify each pixel's signal 100-fold before being sent through an analog-to-digital converter and read by the micro-processor. The pixel values are stored in memory until all of the desired areas of the CID have been read out, and then the image is processed.

The sensor used in this system is the Retroreflector Field Tracker (RFT) flown on STS26 as a part of the solar array flight experiment. There are three Z-80 micro-processors that control target acquisition, image processing, and data output. The RFT can track up to 23 targets simultaneously and outputs the target data in angular terms θx and θy. The data is passed to the GSE serially at 4000 bits per second and passed out of the GSE serially at 19200 bits per second. The update rate for one entire field of data (up to 23 targets) is two per second.

Each picture consists of two full read-outs of the sensor: one with the target illuminated by the laser diodes and one of the background without any illumination. The background pixel values are subtracted from the illuminated picture to generate a picture with a high signal-to-noise ratio. The bright areas in this picture are the spots, which consist of the desired reflections from the retroreflectors and any other reflections that may occur, that reflect the laser diodes. Once the initial desired spots have been found, they are tracked by finding the least squares difference between the predicted positions of the tracked spots and the centroids of the spots currently detected.

8.2.4 Control Algorithms

In the design of the control algorithms, it was necessary to minimize the processing time needed, because the majority of each half-second control cycle on the COMPAQ DESKPRO 386/20 computer was required for serial I/O. Fortunately, the geometry of the hardware is such that the docking target is directly in front of and aligned with the sensor whenever the docking mechanism is employed, eliminating the need for calculating any offsets. Thus, all that was needed was an algorithm that would basically drive each error to zero at a reasonable rate and then keep it there. As a result, a single control strategy which could be executed efficiently and independently for each axis was developed. The following equation,

$$e_i = \theta_i \; a_i \; - w_i \; b_i$$

where: e_i = error signal for axis i
q_i = position measurement
a_i = position gain
w_i = rate measurement
b_i = rate gain

is used to compute the basic error signal for each axis. It is then quantized to five levels, because the thrusters on the mobility base can only be operated in 100 millisecond increments of a 500 millisecond computation cycle. Also, a hard-limiting rate control will command a short (200 millisecond) pulse in the opposing direction if the rate exceeds a specified maximum. The gains and rate limits used are shown on page one in the appendix, and a flow chart illustrating the program flow is on page two in the appendix.

8.3 PRELIMINARY RESULTS

The work accomplished here has verified that automatic docking with a stabilized vehicle using a passive target is completely feasible. The performance envelope of the system discussed herein has been tested at a range of up to 11 meters, an initial yaw position of up to 35 degrees, an initial yaw rate of up to 3 degrees per second, and an initial lateral position offset of up to 8 degrees (limited by the sensor's field of view). The tracker was built to track retroreflectors at 100 feet, so it seems very likely that docking can be successfully accomplished out to the maximum range allowed by the flat floor.

8.4 FUTURE EFFORTS

In the future, efforts will be directed toward evaluating the systems ability to work with rolling and tumbling targets. These scenarios will be simulated using the overhead crane to provide dynamic target motion. Also, the tolerance of the system to ambient solar illumination will be evaluated and improved. New sensors will be integrated into the system to provide higher data rates and the ability to track more reliably. The docking algorithms will be upgraded to include Kalman filtering, adaptable goal setting logic, and compensation for orbital mechanics.

8.5 REFERENCES

1. Tietz, J. C. and Kelley, J. H.: Development of an Autonomous Video Rendezvous and Docking System. Martin Marietta Corporation, Contract No. NAS8-34679, Phase One, June 1982.

2. Dabney, Richard W.: Automatic Rendezvous and Docking: A Parametric Study. NASA Technical Paper No. 2314, May 1984.

3. Tietz, J. C. and Richardson, T. E.: Development of an Autonomous Video Rendezvous and Docking System. Martin Marietta Corporation, Contract No. NAS8-34679, Phase Two, June 1983.

4. Teitz, J. C.: Development of an Autonomous Video Rendezvous and Docking System. Martin Marietta Corporation, Contract No. NAS8-34679, January 1984.

Section 8
Appendix

Gains and Rate Limits

Input Signal	Position Gain	Rate Gain	Rate Limit	Output Signal
θ_0 Range	.25	10	*	e_0 (X)
θ_1 Sensor Azimuth	.5	5	1.4	e_1 (Y)
θ_2 Target Azimuth	.1	.5	1.0	e_2 (Yaw)
θ_3 Sensor Elevation	.18	1	2.8	e_3 (Z)
θ_4 Target Elevation	.1	.1	1.0	e_4 (Pitch)
θ_5 Roll	.15	.1	2.8	e_5 (Roll)

* Rate Limit = range/120 + .01

Program Flow

Target Azimuth, Elevation,
Sensor Azimuth, Elevation,
Relative Range, Roll

Filter Sensor Data to
Reduce Noise

$\theta_i\ (t) = .5^* \ (\theta_i\ (t) + \theta_i\ (t\text{-}dt)$

Derive Rates

$w_i\ (t) = 1/2dt) * (\theta_i\ (t) - \theta_i\ (t\text{-}dt))$

Compute Error Signals

$e_i = (\theta_i\ a_i + w_i\ b_i) * 6.0$

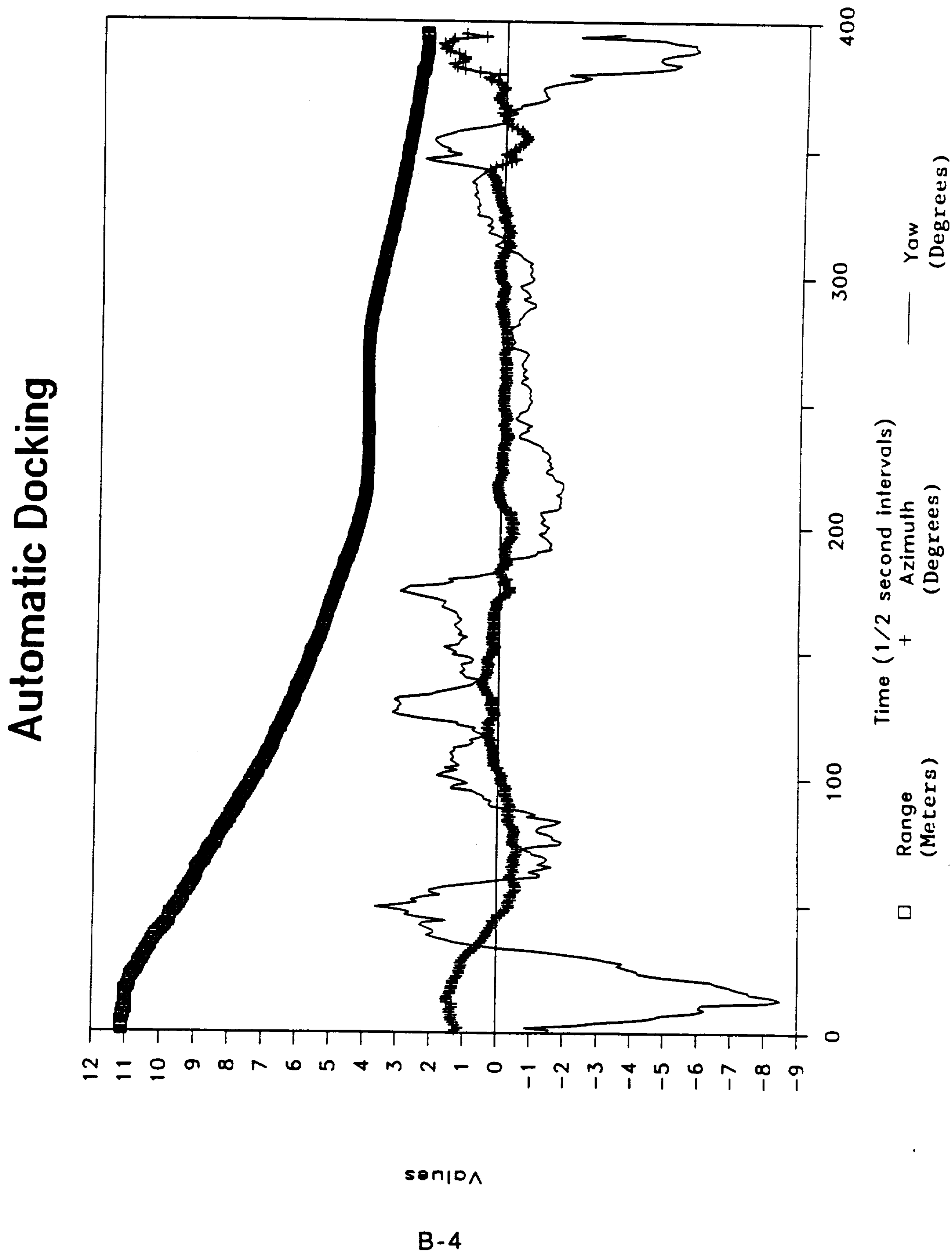
Upper and Lower Error
Command Limits

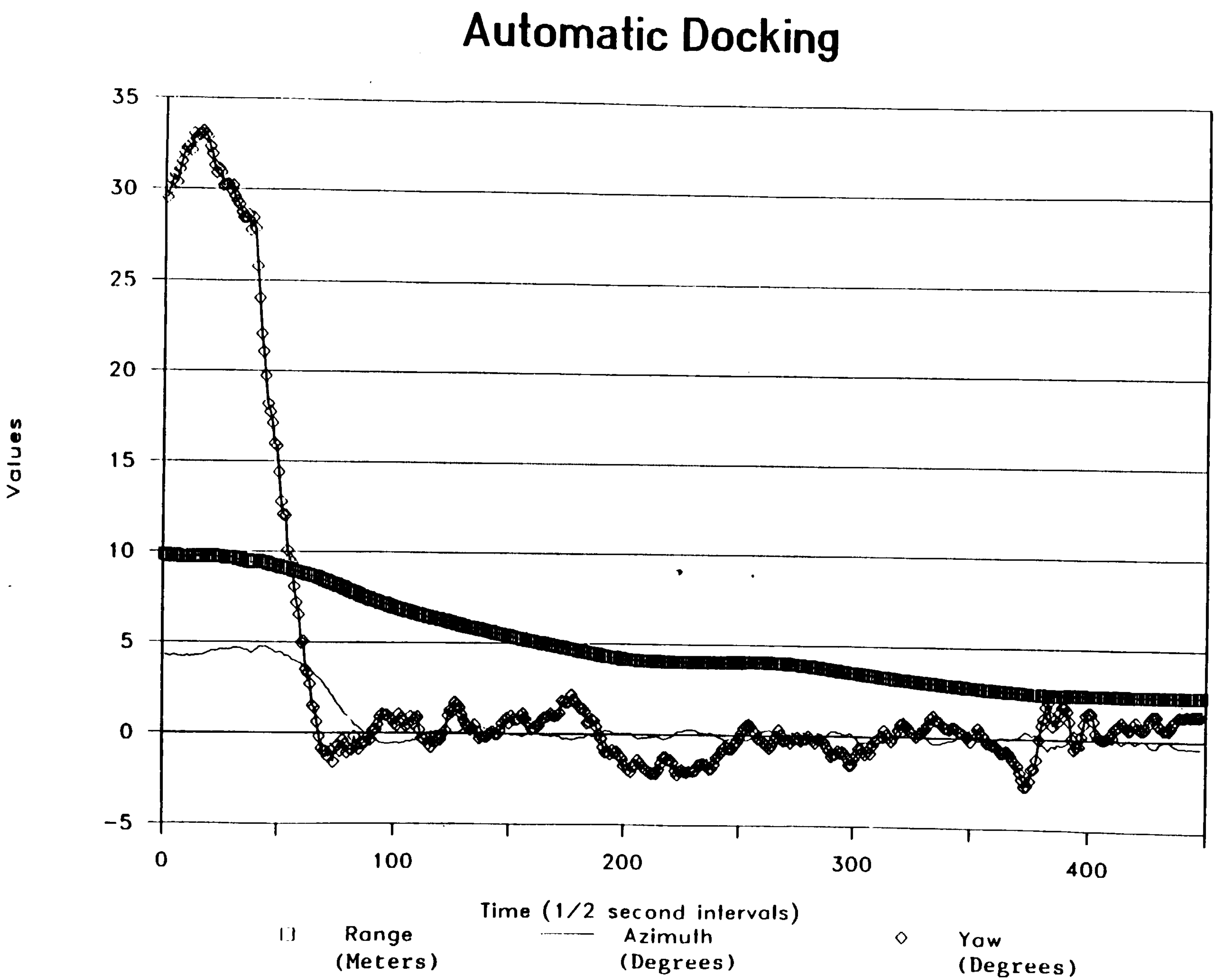
if $(e_i > 6)$ then $e_i = 6$

if $(e_i < 2)$ then $e_i = 2$

Upper and Lower Rate Limits
if $w_i >$ rate limit then
$e_i = .3$

Vehicle Commands to Thruster
Selection Logic
(Through serial port)


Automatic Docking
Values
Time (1/2 second intervals)
Range (Meters)
Azimuth (Degrees)
Yaw (Degrees)
12
11
10
9
8
7
6
5
4
3
2
1
0
-1
-2
-3
-4
-5
-6
-7
-8
-9
0
100
200
300
400


Automatic Docking
Values
35
30
25
20
15
10
5
0
-5
0
100
200
300
400
Time (1/2 second intervals)
Range
(Meters)
Azimulh
(Degrees)
Yaw
(Degrees)

FY89 Accomplishments

- Successful Full-Scale Hardware-In-The-Loop Demonstration
  -Firmly Establishes Feasibility of Concept

    - Video Sensor
    - Simplified Phase-Plane Control Algorithms
    - Passive (Reflective) Target
    - Laser Illumination

  -Provides Test Bed For Further Work

    - System Requirements Definition
    - Control System Optimization

- Generated Experimental Data For Requirements Derivation
  -Sensor Capabilities

    - Resolution
    - Field of View

  -Docking Target Design (Patent Applied For)
  -Vehicle Contro System Characteristics

    - Acceleration Levels
    - Time Delays

- Developed Versatile High-Fidelity Auto Docking S/W Simulation
  -Can Pre-Program Parameterized Run Sequence In External Set Up File
  -Can Accommodate Multiple Sensor Models
  -Contact Dynamics Modeling Capability
  -Orbital Mechanics/Gravity Gradient
  -Accurate Solar Position Model

FY89 Experimental Results

- Acceptable Initial Conditions

  -Initial Attitude Misalignment: 30 Degrees

    - Limited By Characteristics Of Reflective Tape
    - Max Angle Can Be Increased If Bandpass Filters Used

  -Initial Chase Vehicle Line-Of-Sight Azimuth Angle: 15 Degrees

    - Limited By Field-Of-View Of Sensor
    - Can Be Increased by Changing Sensor Lens

-Initial Yaw Rate: 15 Degrees Per Second

- Limited By Vehicle Performance, Tracking Algorithm
- Can Be Increased By Increasing Thrust And Use Of More Sophisticated Tracking Algorithms

-Success Rate: Approximately 90%

- Failures Usually Due To Target Mistracking
- More Image Pre-processing Needed
- Better Tracking Algorithms Needed

NASA

National Aeronautics and Space Administration

REPORT DOCUMENTATION PAGE

1. Report No. TM 102163	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle Pathfinder Autonomous Rendezvous and Docking Project FY 89 Annual Report		5. Report Date August 1990
		6. Performing Organization Code EG
7. Author(s) Stephen Lamkin, NASA-JSC and Wayne McCandless, Lockheed--Editors		8. Performing Organization Report No. S 607
9. Performing Organization Name and Address Navigation, Control and Aeronautics Division Engineering Directorate NASA Johnson Space Center Houston, TX 77062		10. Work Unit No. 591-21
		11. Contract or Grant No.
12. Sponsoring Agency Name and Address		13. Type of Report and Period Covered Technical Memorandum FY 89
		14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

The AR&D Project will develop and demonstrate capabilities to support manned and unmanned vehicle operations in lunar and planetary orbits. In this initial phase of the project, primary emphasis has been placed on definition of the system requirements for candidate Pathfinder mission applications and correlation of these system-level requirements with specific requirements. The FY 89 activities detailed in this document are best characterized as foundation building. The majority of the efforts were dedicated to assessing the current state of the art, identifying desired elaborations and expansions to this level of development and charting a course that will realize the desired objectives in the future. This document details efforts across all work packages in developing those requirements and tools needed to test, refine, and validate basic AR&D elements.

17. Key Words (Suggested by Author(s)) Controls, Guidance, Navigation, Automation, Sensors, Tracking, Requirements	18. Distribution Statement Unclassified Subject Category 01

19. Security Classification (of this report) Unclassified	20. Security Classification (of this page) Unclassified	21. No. of pages	22. Price

For sale by the National Technical Information Service, Springfield, VA 22161-2171

JSC Form 1424 (Rev Aug 89) (Ethernet Jan 88)

PATHFINDER AUTONOMOUS RENDEZVOUS & DOCKING PROJECT FY89 ANNUAL REPORT DISTRIBUTION LIST

ES43/L. Othon
DM4/D. J. Pearson
EE6/J. L. Prather
J. C. Lamoreux
EE6/H. Nitschke
ER2/K. Baker
K. K. Hiers
EG/K. Goodwin (CSDL)
D. Zimpfer (CSDL)
EG2/G. McSwain
E. T. Kubiak
EGA/W. L. Jackson
EG12/D. C. Brown
ES/A. D. Gist
ES6/T. O. Ross
ER4/L. Stokes
ER4/C. J. Gott
ET4/R. S. Merriam
PT4/R. T. Savely
IA/J. S. Moore
IC/M. Nolan
IE/3/R. C. Kahl
IE5/N. Lance
JM2/Technical Library (3)
EG4/S. Lamkin
G. C. Hite
CB/E. Ochoa

NASA Headquarters
R/A. D. Aldrich
RS/L. M. Couch
RC/J. D. DiBattista (3)
RC/L B. Holcomb
RS/J. C. Mankins

Ames Research Center
210-9/C. Jorgensen
202-3/Library (2)

JPL
301-125L/A. R. Klumpp
301-125/L. J. Wood
301-170K/R. E. Lock
111-113/Library (2)

Langley Research Center
MS 161/D. Price
MS 185/Technical Library (2)

MSFC
EB24/R. T. Howard
ED13/R. W. Dabney
ER21/J. B . Haussler
CN22/Central Library (2)

C. S. Draper Laboratory
555 Technology Square
Cambridge, MA 02139-3529
2B/P. Kachmar
4A/E. Bergmann
4A/R. Larson
12B/T. Brand
14C/S. Croupnick

Environmental Research Institute of Michigan
K. Gilbert
P. O. Box 8618
Ann Arbor, MI 48107

GE Aerospace
J. R. Tower
ATL Bldg
Moorestown Corporate Center
Moorestown, NJ 08057

General Electric
D. Darooka
P. O. Box 8555
Philadelphia, PA 19101

LESC
B16/W. McCandless
C32/O. Olszewski
C070A/J. Moore

Martin Marietta Space Systems
S8082/B. Clark
S8082/W. Woodis
P. O. Box 179
Denver, CO 80201

McDonnell Douglas Space Systems Co.
21-1/L. S. Braun
5301 Bolsa Ave
Huntington Beach, CA 92647

Rocketdyne
TA21/S. Varkhoudarian
6633 Canoga Ave
Canoga Park, CA 91303

Saturn Tech Corp.
D. Isenhower
71 Rogers St.
Cambridge, MA 02142

Southwest Research Institute
H. Spence
6220 Culebra Rd
P. O. Drawer 28510
San Antonio, TX 78284

SPAR Aerospace
M9L-2W7/c. Roy
1700 Ormont Dr.
Weston, Ontario

TRW--Houston
J. Baxter
D. Eick
R. Lee

TRW Federal Systems Division
P. Campbell-Mail Code R11/2720
One Space Park
Redondo Beach, CA 90278

USAF/Space Systems Division
ALII/D. P. Ball
P. O. Box 92960
Los Angeles, CA 90009-2960