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0007F

OPERATIONS CONCEPT

14 FEBRUARY 1983

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1.0 INTRODUCTION

1.1 Purpose

The purpose of this report is to define the operations concept developed to retrieve the spacecraft from orbit with the space transportation system. The focus of the report is to define the sequence of activities that are required and the vehicle modifications required to support the retrieval operation.

1.2 Scope

This document will confine itself to those aspects of the mission that directly support orbital operations.

2.0 MISSION OBJECTIVES AND GROUNDRULES

2.1 Mission Objectives

The primary STS mission objective is to rendezvous with the spacecraft, berth the spacecraft in the payload bay, and return the spacecraft to Earth.

The primary spacecraft mission objective is unchanged from previous missions, where retrieval by the STS was not an option. This dictates the requirement that the capability must be retained throughout the retrieval mission to deboost the spacecraft into a broad ocean area should the retrieval be unsuccessful.

2.2 Groundrules

The list of groundrules pertinent to the operation that this study was performed under is listed below:

-- Manually removable standard grapple mounted on the mid section.

-- Some appendages may have to be jettisoned, either via remote control or by EVA, prior to berthing.

-- STS launch from Vandenberg AFB

-- Any remaining spacecraft may be the retrieval target.

-- The spacecraft +Z will be aligned with the Orbiter's -Z when berthed.

-- The spacecraft will be in an acceptable, stable orbit (LVLH attitude) prior to STS launch, and no maneuvers are planned for the spacecraft during the nominal rendezvous.

3.0 READINESS

3.1 Remote POCC

The remote POCC will be located at the Air Force Satellite Control Facility in Sunnyvale, California. Direction for performance of spacecraft activities will originate from the AFSCF throughout the entire retrieval mission.

The spacecraft data and commands will originate from the VOA MCC-2, TA Area A complex. Coordination of the Shuttle activities will be accomplished from the VOS MCC B complex. A Mission Interface Document (MID) defining each office's roles and responsibilities will be written, and sufficient secure communications between the two operations areas will be established.

3.2 Remote POCC/TSC Interface

The POCC/JSC interface will terminate in the VOS MCC-B complex. Other than monitor only lines on existing drops in the VOA complex, no communications to JSC will be added to the VOA complex.

No additional interface requirements above and beyond that already established between the VOS operations complex and JSC are envisioned.

3.3 Rehearsals and Joint Simulations

Due to the need to concentrate on the primary spacecraft mission, it is currently envisioned that VOA would not rehearse the retrieval mission during the spacecraft prelaunch readiness phase.

After the primary mission is completed, which could be up to one year after spacecraft launch, the spacecraft would be put into its stowable orbit to await the STS launch. Approximately two or three months should remain before the STS would be ready for launch, which should be ample time to hold a number of development rehearsals to prepare for the retrieval mission. Since the spacecraft is still in orbit, the VOA participation in these development rehearsals will be affected by the need to continue to support the vehicle on orbit. However, since VOS will be conducting the STS activities, this is not considered to be an unworkable handicap.

VOS development rehearsals could begin at any time prior to the mission, with a simulated VOA interface if VOA were not available due to operational activities.

4.0 PRE-STS LAUNCH ACTIVITIES

4.1 Spacecraft Parking Orbit Definition

Once the primary mission has been completed, a major readiness review should be held to evaluate the spacecraft and STS readiness to conduct the retrieval mission. The primary focus on the spacecraft would be to determine if enough consumables reman to enable the spacecraft to wait for the Orbiter's launch and a safety review of the spacecraft in its post primary mission configuration. The STS readiness activities would also be reviewed to verify that the STS will be ready to conduct the retrieval within the lifetime of the spacecraft.

If the results of this review are positive, the spacecraft should then be moved into a parking orbit to await the STS launch. The definition of this parking orbit would be the result of a joint Air Force/ NASA study to minimize STS maneuvering during rendezvous.

The characteristics of the parking orbit shall be determined from expected time delay between the end of the primary mission and the STS launch date. Consideration must be given to the most efficient use of the fuel remining while still protecting the SV deboost capability. In Appendix B it is suggested that if a six-month delay is anticipated, the SV be placed into a storage orbit of 181x186 NM and be allowed to delay to a retrieval orbit of a 153.8 NM circular (assuming a B-factor of .716 which results in a Q of 16.0). If the delay is only 90 days, the storage orbit should be at 171.4 NM circular using up 477 lbs. of propellant. This is more efficient than staying in the retrieval orbit for that time which may use up 1000 lbs. in 90 days for orbit maintenance and ACS control.

Besides fuel usage another consideration may be power generation capability to protect deboost. Once the SV is boosted out of the sun synchronous mission orbit the beta angle will no longer remain constant. The walk in the angle changes the solar array charging capability as indicated in Reference A. Since prior to solar array separation we wish to have the batteries fully charged, the final beta angle will be important. In the case studies for the 180 day storage, the angle changed from 16.8° to 29.4° thus reducing the charging capability by 10%. This should also be considered for the correct time of year in the planning of the final storage orbit.

In summary, the storage orbit characteristics should consider:

-- Fuel remaining/protect deboost capability
-- B-Factor
-- Retrieval orbit
-- Time of year
-- Battery charging requirements
-- Beta angle variation
-- RTS station contacts

NASA and the SPO need to coordinate on the model of the Earth which is to be used for mission planning. The model used by NASA was operated and resulted in a different circular orbit altitude (for a Q = 16) than Appendix B. This needs to be designated in the PIP or ICD to assess rendezvous targeting strategy.

4.2 Spacecraft S/C Safing Criteria

The spacecraft may coninue to perform secondary mission objectives while in the parking orbit at the direction of the Program Office. In this case all of the safing functions that can be accomplished prior to proximity operations will be postponed until just prior to STS launch.

The nominal scenario for safing the S/C is as follows:

-- Immediately following the primary spacecraft mission the main sensor doors are closed and remain closed with power off.

-- At the termination of the parking orbit, after a rendezvous orbit has been attained, the orbit adjust system will be safed by conventional ECS command and message sent through the RTS and verified by telemetry on the ground.

-- If there are appendages on the spacecraft (other than solar arrays) that fall outside the cargo bay envelope (for berthing and payload bay door closure) these appendages will be released or stowed prior to STS launch. Conventional ECS message and ground telemetry verification will again be used.

-- If desired, an intermediate low power configuration can also be commanded at this time.

-- The reaction control system (RCS) safing macro will be loaded via the ECS to be executed prior to grapple. The Safety Board requires triple-redundancy for "RCS safing." Triple redundancy for safing the primary thrusters can be achieved as follows: assuming the redundant FCEA is OFF and it has been verified that there is no leak on the redundant thruster:

a. (PRI ACS) RCS Thrust Valve Power Off

b. RCS Isolation Valves 1A and 1B Closed

c. Primary Modulator to Redundant Valve Drivers XSTRP

RCS thruster safing can be accomplished by issuing the following prestored macro commands in the following order:

(PRI ACS) RCS THRUST VALVE PWR-

RCS ISO VALVE 1A (& 4A) CLOSE

RCS ISO VALVE 1B (& 4B) CLOSE

PRI MOD 1/2 TO RED DRIVER XSTRAP

PRI MOD 3/4 TO RED DRIVER XSTRAP

PRI MOD 5/6 TO RED DRIVER XSTRAP

PRI MOD 7/8 TO RED DRIVER XSTRAP

Additional safing commands may be needed to ensure that the SV does not radiate in the cargo bay. All safing commands need to be verified at the STC prior to authorization to grapple. This execution can be initiated by real time command through the RTS, with verification through downlink TLM [redacted] onboard the STS.

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4.3 Spacecraft Configuration Prior to STS Launch

Once it was determined that the spacecraft was in an acceptable orbit for rendezvous, the spaceraft would be configured in the following manner:

-- OAS would be safed
-- All payload systems off
-- All pallets off
-- ECS and MCS would be on and loaded with station contacts and the RCS safing macro
-- RCS on
-- TT&C system on as required

4.4 Remote POCC/JSC Interface Activiation

Approximately three days prior to launch, the VOS and VOA coordination links will be manned on a 24-hour per day basis. The VOS flight control team would also man their MCC to monitor the STS pad cycle and to manage vector coordination between VOA and JSC. Full manning for both VOA and VOA would begin approximately twelve hours prior to launch.

5.0 RETRIEVAL MISSION ACTIVITIES

5.1 Mission Time Line

The mission time line is laid out to take advantage of BRM4 requirements for a seven-day mission. In this way post MECO rendezvous maneuvers need not be rushed. In this scenario the primary grapple attempt can be planned to occur on flight ~~day~~ DAY 3 or 4. Appendix A shows a preliminary overview of the mission time line to provide for contingency retrieval operations and STS deorbit on flight day six. In general, all maneuvers to make the STS coorbital with the SV are done on the first and second day (third day also, if required). These rendezvous maneuvers will place the Orbiter in a catch up orbit with the SV but displaced in time per Figure 5-1. That puts the SV at a safe distance ahead of the Orbiter and 10 NM below prior to the sleep cycle.

After waking, the crew would assess the relative navigation situation. They then perform height adjust NH3 and the terminal phase maneuver of the rendezvous. They arrive at a point 1000 ft. from the SV, and coorbital along the velocity vector, per Figure 5.2.

Proximity operations then begin to maneuver the Orbiter to within 200 ft. of the SV along the $\bar{V}$. In order to reach the grapple of the SV and avoid plume impingement from the Orbiter, the Orbiter must rotate around the SV. An $\bar{R}$ aproach seems satisfactory with the ACS being shut off on the SV when the center of gravity of the two vehicles are about 90 ft. apart. The Orbiter would be proceeding at .1 ft./sec. until the RMS can be extended at 35 ft. reach distance. This would allow six minutes for the RMS operator to grapple the SV before the position errors exceed the requirements of Vol. IVX of the Payload Accommodation Specification Section 8.3.1. The time for the retrieval operation is shown in Figure 5.3.

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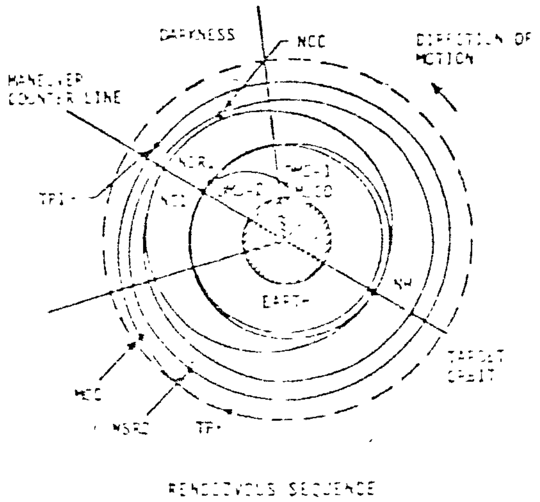
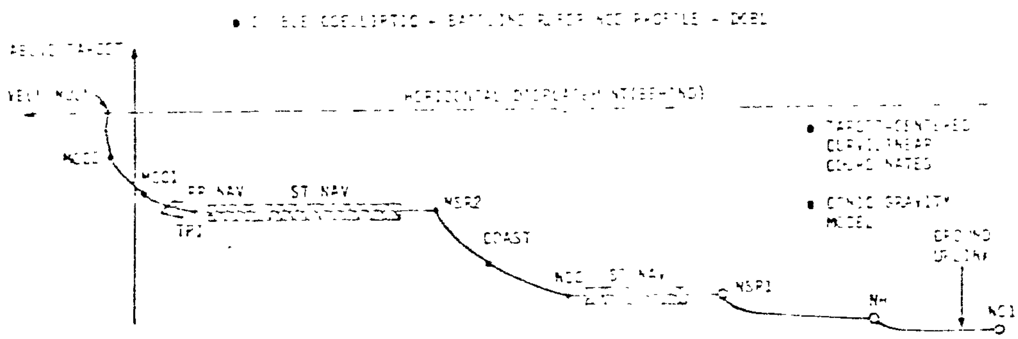


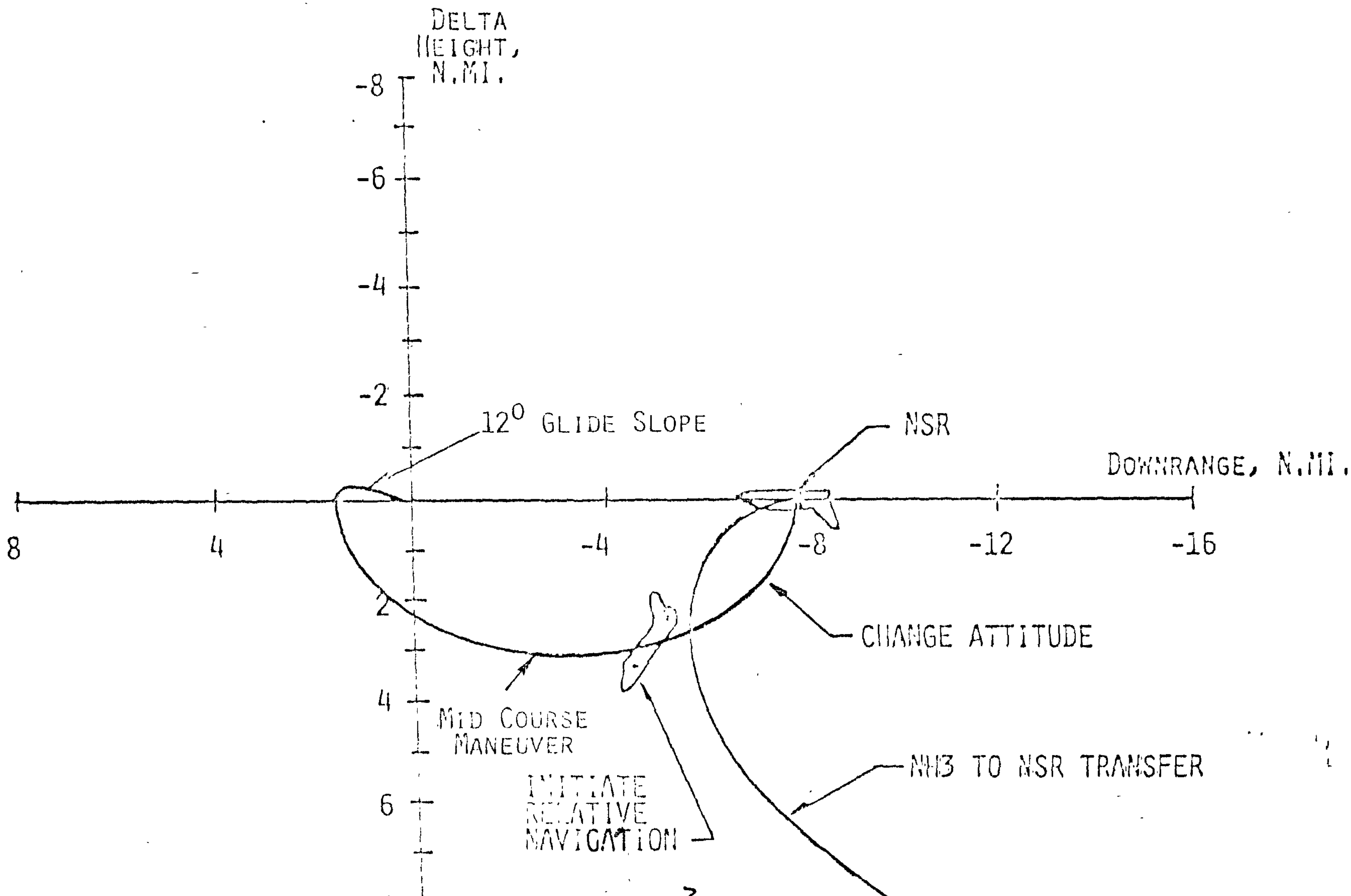
	MANEUVERS
MECO	MAIN ENGINE CUTOFF
OMS-1	INSERTION
OMS-2	CIRCULARIZATION
NC1	PHASING
NH	HEIGHT ADJUST
NSR1	FIRST COELLIPTIC
	NAV TRACKING (S TRK)
NCC	CORRECTIVE COMBINATION
NSR2	SECOND COELLIPTIC
	NAV TRACKING (S TRK)
TPI	TERMINAL PHASE INITIATION
	NAV TRACKING (RADAR)
MCC	[illegible]
	NAV TRACKING (RADAR)
[illegible]	TERMINAL PHASE [illegible]
	BRAKING
	TRANSITION TO OFFSET
	STATIONKEEPING
	PROXIMITY OPERATIONS



TIME, MIN	EVENT	RANGE
0	GROUND UPLINK	239.34
4.0	NH MNVR - GROUND TGTD	[illegible]
[illegible]	NSR1 MNVR - GROUND TGTD	[illegible]
[illegible]	STAR TRACKER NAV INIT	—
[illegible]	STAR TRACKER NAV END	--
[illegible]	NCC MNVR	68.49
[illegible]	NSR2 MNVR	36.85
[illegible]	STAR TRACKER NAV INIT	--
[illegible]	STAR TRACKER NAV END	--
[illegible]	TPI MNVR	11.05
[illegible]	RENDEZVOUS RADAR NAV INIT	--
[illegible]	MCC1	2.44
[illegible]	MCC2	1.03
275.8	VEL NULL	1000 FT
276.8	AUTO STATIONKEEPING	1000 FT
300.0	[illegible]	--

Figure ~~5.2~~. Relative Motion Plot of Double Coelliptic Rendezvous Maneuvers

	STABLE ORBIT RENDEZVOUS TERMINAL PHASE	



[redacted]re 5.2

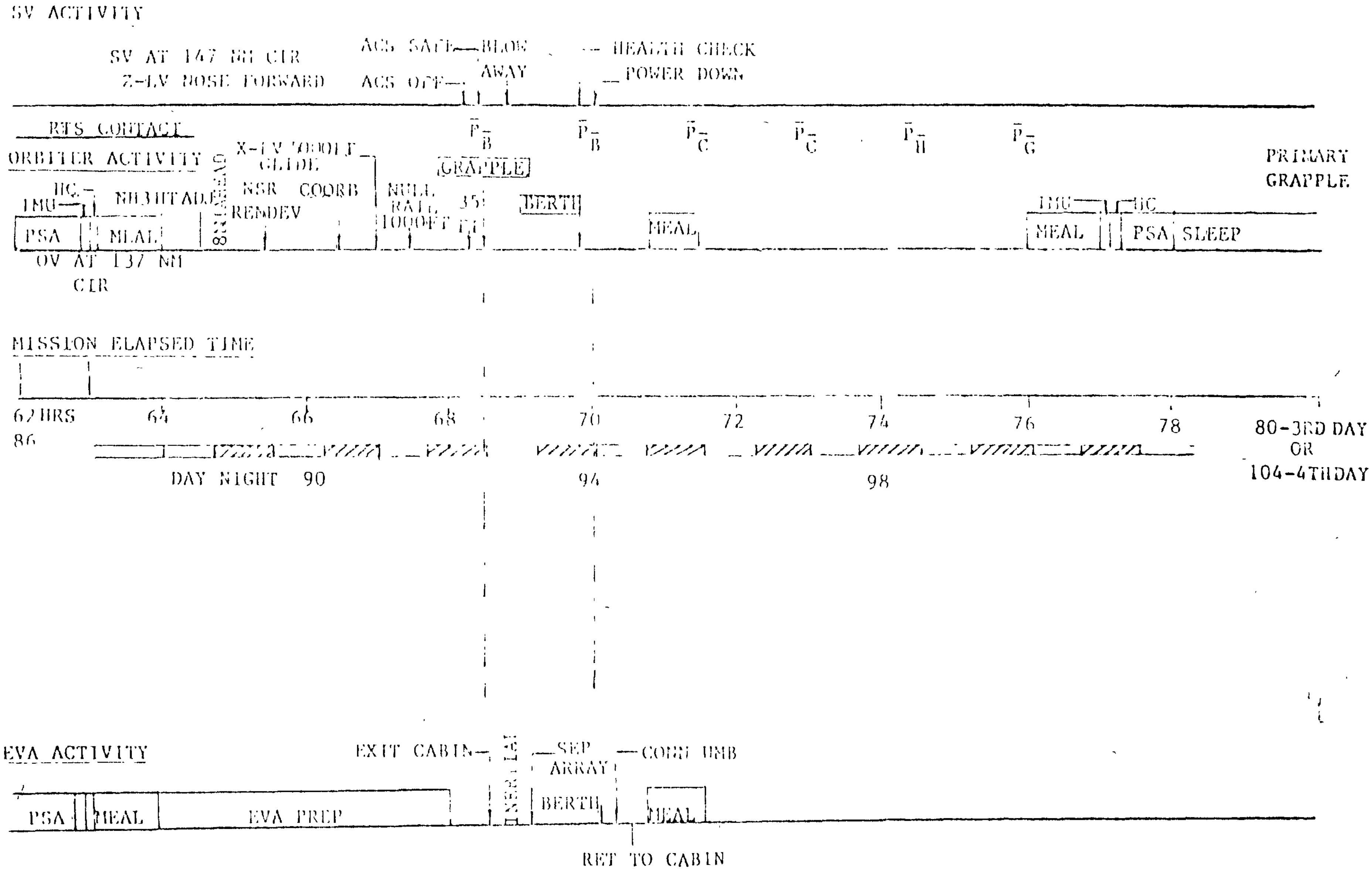


Figure 5.~~4~~3. Retreival Timeline Operations

An alternate approach to the grapple pin from 200 ft. would be along the $\overline{V}$ as shown in Figure 5.4. In this scenario the Oribter approaches the SV to within 50 ft. The Orbiter than rotates 30° in inertial space under the SV, while the SV rotates in Z-LV orientation by 30° in 7.5 minutes. The crew, using vernier control maintains relative altitude with orbiter rate in pitch to station-keep with the SV. This position will also place the SV X axis in line with the Orbiter C.G. to minimize relative R motion.

After grappling the rigidization, the Orbiter goes into an inertial mode. Then the RMS operator extends the SV on the ARM such that the solar array is away from the Oribter and pointing towards the Earth. As shown in Figure 5.4, the solar arrays can then be separated and will safely follow a trajectory which will not impact the Orbiter. The payload is then configured to be berthed by the RMS operator into latches prior to a health check and powering down the SV.

5.2 STS Rendezvous Activities

The currently planned standard STS rendezvous will be the double coelliptic method. This method is depicted in Figure 5-1. The spacecraft will be in a known, stable orbit and will not maneuver during the rendezvous sequence. The spacecraft's desired position at rendezvous depends primarily on lighting conditions, and the duration of the rendezvous could range from hours to days.

In order to accomplish a rendezvous misson, several items must be taken into consideration:

a. A typical target vehicle orbital altitude is greater than can be reached in powered flight using the Orbiter's main engines and solid rocket boosters. Thus, main engine cutoff (MECO) will occur considerably below the target's orbit, and subsequent burns will be required to force the Orbiter's orbit to intercept the targets's orbit.

b. Achieving an appropriate phase angle between the target and the Orbiter at orbit insertion will either restrict the launch window or force the inclusion of a phasing maneuver which would allow the Orbiter to "catch up" with the target.

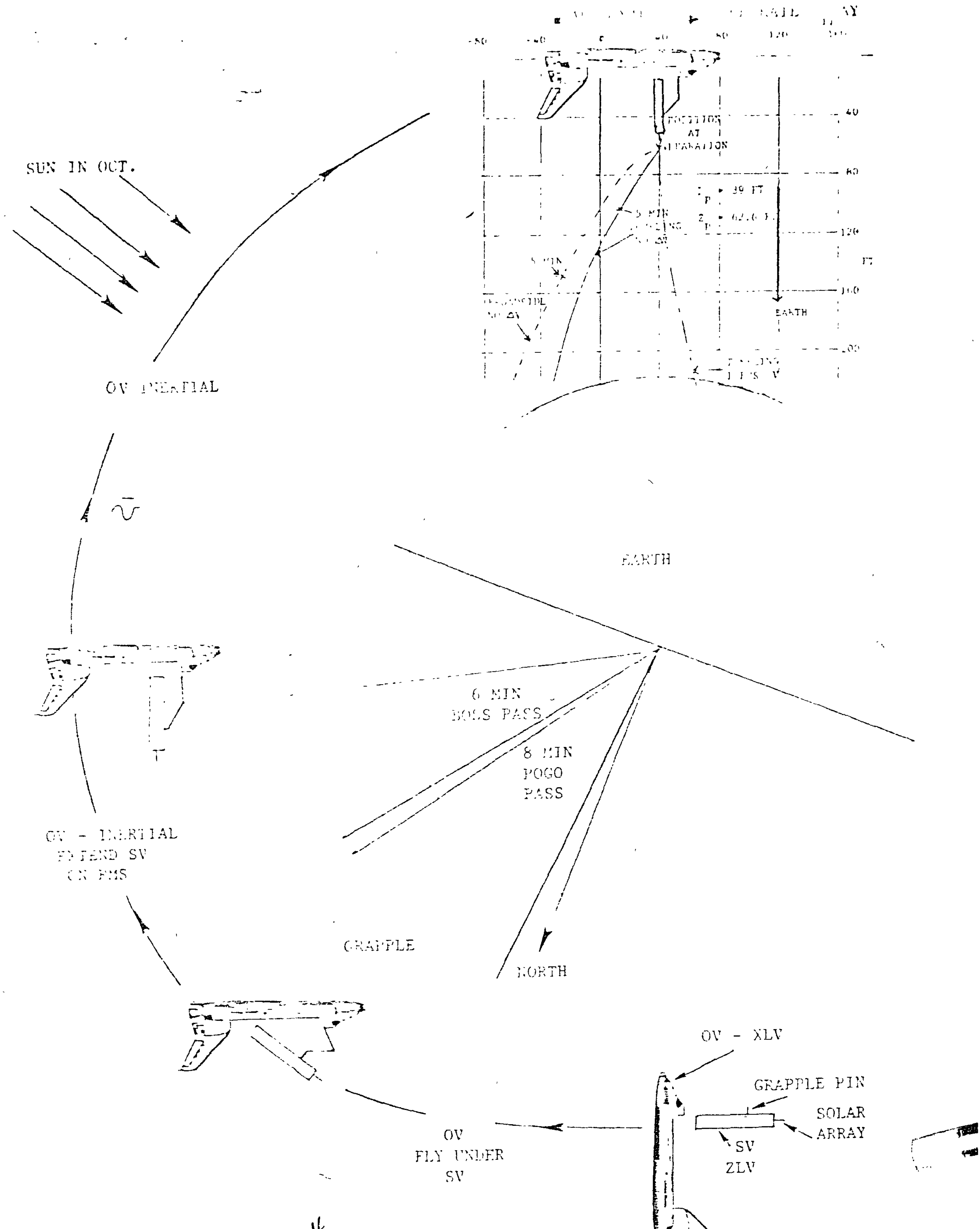


Figure 5.4. Grapple and Solar Array Separation

c. The launch window and the direction of the launch is designed so that the Orbiter will be approximately within the orbit plane defined by the target vehicle. The time of launch within the launch window, as well as errors which devewlop during the powered phase of ascent, may result in the Orbiter and target being slightly different orbit planes. The rendezvous sequence, then, must include the potential for an out-of-plane corrective burn.

d. Of particular importance are the lighting conditions during the final portions of the rendezvous phase. As the final maneuvers are to be performed visually by the crew, both vehicles must be in the light. Furthermore, they must remain in the light long enough to allow completion of the rendezvous and PROX OPS phases, as well as satisfy the mission objectives (deploy/retrieve payload, etc.). In a similar way, many of the onboard targeted maneuvers are scheduled to occur with the target in the light, so that Star Tracker marks may be taken and incorporated into the navigation software before burns are executed.

e. The rendezvous sequence must have the flexibility to compensate for the errors inherent in the navigation, guidance, and flight control systems.

f. The rendezvous sequence also must not violate any other mission rules. Thus, the launch window must allow for the correct lighting conditions at launch, and, in case of aborts, enough Reacion Control System (RCS) fuel must remain onboard to accomplish the mission, as well as to return to the landing site in case of Orbital Maneuvering System (OMS) failure.

g. Lastly, with all these items considered, the resultant launch window must be of a realistic duration.

It is important to note on Figure 5-1 that after the NSR1 (first coelliptic burn) all remaining burns are targeted by onboard software, based on onboard sensor data, not by the ground. The only sensors available are the Ku-band radar, the Star Tracker, or the crew optical alignment sight (COARS).

If the spacecraft is a passive radar target (no radar transponder) the Ku band radar is not expected to be able to acquire at ranges greater than 12.5 NM. This would not be until after terminal phase initiation, so something other than the Ku-band radar must be used for the NCC and NSR2 burns.

If the lighting conditions can be planned properly, the Shuttle Star Tracker can be used in the place of the rendezvous radar. Spacecraft look angles to the Shuttle during these periods, spacecraft reflectance, and sun angle requirements all combine to make using the Star Tracker as a rendezvous sensor very difficult.

If any of the criteria mentioned above cannot be fulfilled, the COAS could be used if a strobing beacons were mouned on the spacecraft to allow astronaut visual acquisition.

Lighting must still be considered, as some of these navigation marks must be taken at ranges approaching 300 NM. The accuracy of this instrument leaves much to be desired, however, and this inaccuracy will require larger fuel reserves to be maintained for use later in the rendezvous sequence to eliminate errors.

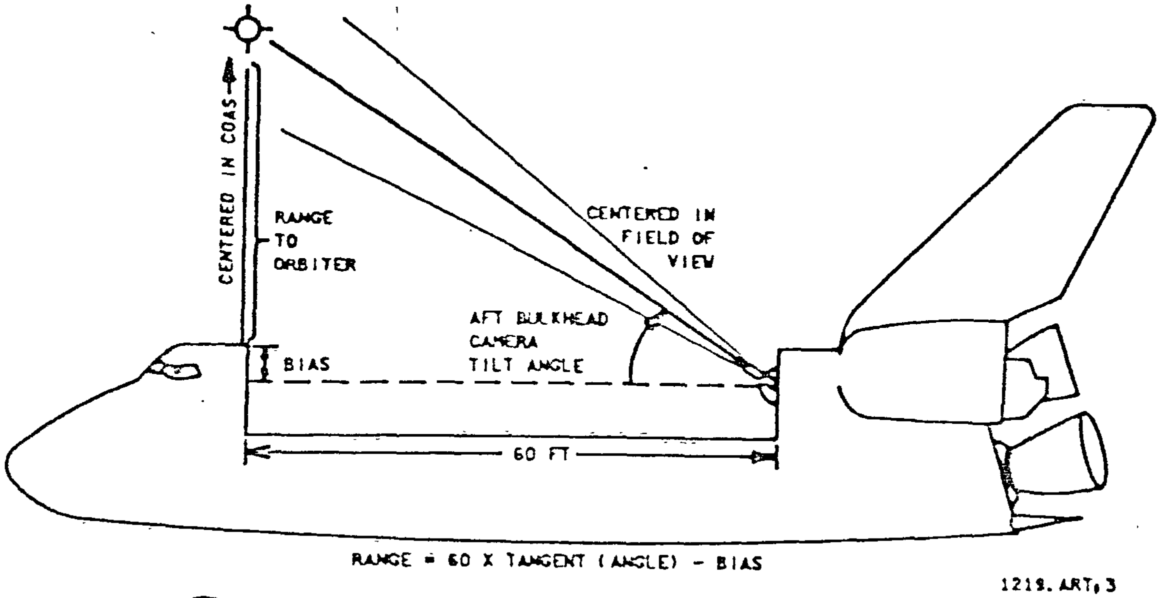
The only spacecraft activity that would occur during this period is the powering up of any rendezvous aids such as transpoders or light beacons.

5.3 Terminal Phase Activities

On the day of retrieval, the crew would use whatever sensor was available to provide data for an onboard solution for the TPI burn. Based on data obtained after Ku-band radar lock (approximately 12.5 NM apart) mid course corrections will be performed as required. This phase will be completed when the Orbiter arrives at a point 5000 ft. in front of the spacecraft and on a line of sight 12° above the spacecraft velocity vector (Figure 5-~~3~~ 2).

5.4 Proximity Operations

The Orbiter then proceeds down the 12° "glide slope" towards the spacecraft. During this approach, as in all other phases of the mission, a contingency plan to allow retrieval mission completion in the event of Ku-band radar failure must be thought through. This could involve optical targes on the spacecraft for use ~~wiht~~ WITH the Orbiter's CCTV system (Figure 5-~~4~~ 5). Th constraint is stated as follows:



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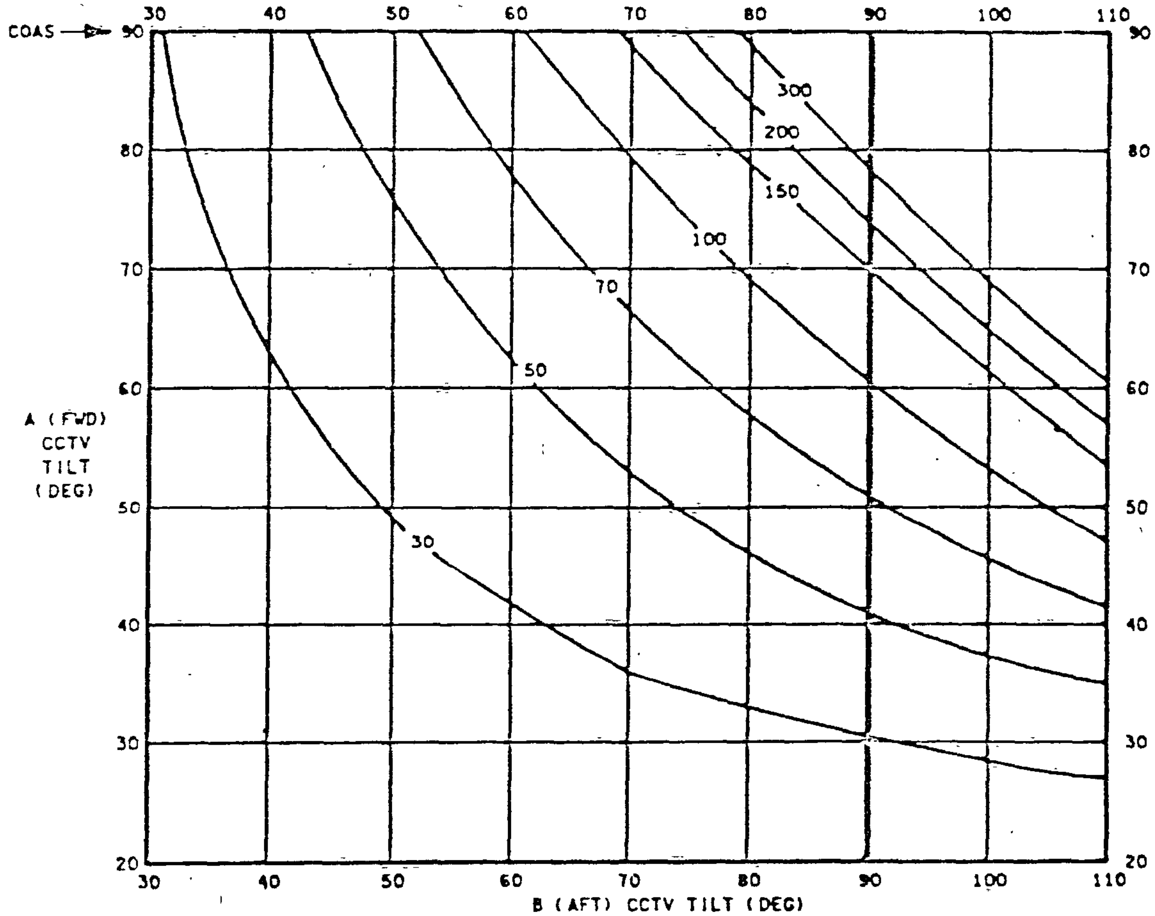
~~Figure 3-3.~~ Use of aft bulkhead camera tilt angle to determine range.



5,5
Figure ~~3-4.~~ CCTV ranging chart, range to bulkhead (ft).

For station-keeping, transition, final approach, and rendezvous intercept targeting outside a 200-ft. range, the crew must be provided with a direct onboard measurement of line-of-sight angle (LOSA), range (R), inertial line-of-sight angle rate (LOSA), and range rate (R) data. The accuracy and frequency of this data must be sufficient to provide an unambiguous picture of the trajectory progress, and to allow timely selection of an effective evasive maneuver should one be required. (STS Operations Guidelines and Constraints, Preliminary, Dec. 1982).

Other constraints that every attempt must be made to satisfy are:

1. Orbiter/payload attitude requirements
   a. It is desirable for the payload to be in local vertical/local horizontal (LVLH) attitude hold with its grapple fixture out of plane along the positive orbital momentum vector (H-bar) and the "short end" of the payload toward the Orbiter for all Prox Ops.
   b. The Orbiter will be in attitude hold from approximately 200 ft. to grapple. The reference for attitude hold is with respect to the stabilized payload (LVLH or interial).
   c. The Orbiter aft crew station overhead window will remain pointed at the payload during proximity Operations, except as noted in paragraph 3d below. The Orbiter X-axis is preferred to be in the payload orbit plane except when procedures require otherwise.
2. Payload communication
   a. Rendezvous radar shall be active from maximum acquisition range to grapple range except when the antenna is used for Ku-band communications.
   b. Payload communications or tracking requirements, including blockage analyses, communications for Instrumentation Cotrol System (ICS) and Attitude Control System (ACS) deactivation, and radio frequence interference (RFI) analyses, are to be considered.

3. Visibility requirements
   a. The sun may not be within $\pm 20^{\circ}$ of the line-of-sight to the payload as long as payload viewing requirements exist (direct visual, CCTV, etc.).
   b. The payload must be within the overhead docking light beam as long as viewing requirements exist for nighttime approaches.
   c. Grapple fixture lighting must be considered for final approaches to grapple.
   d. Visual or electronic acquisition is required from approximately 1000 to 200 ft. If acquisition within this range must be interrupted, the interrupt duration may not exceed the time for the separating range to decrease to 200 ft.
   e. Visual aquisition from 20 ft. through grapple is required.

Two grapple attitudes were studied and still need to be evaluated. One attitude is a rotation to a point parallel to and below the SV. In this position the solar arrays are over the crew station, and an $\bar{R}$ approach is used (see Figure 5.6) to close the remaining distance for grapple. This approach will minimize the effect of Orbiter plume impingement, and has the advantage of using orbital mechanics for breaking.

The other approach studied, is one where the Orbiter rotates to a relative 30° attitude to the SV with the SV-X-axis in line with the Orbiter C.G. This approach keeps the solar array further from the crew cabin and reduces the inherent $\bar{R}$ motion. Fuel consumption and safety need to be traded off to come to a recommended grapple attitude.

The Orbiter approahes the SV at the rate of .1 ft./sec. along the $\bar{R}$ in the LOW-Z mode. At 90 ft. separation of the two centers-of-gravity (CG) the Orbiter will stop and command the SV attitude control system to be inhibited. This can occur [redacted] or the RTS. Figure 5.3 shows the inhibit being commanded over POGO. SV data would then be transmitted to the STC either via RTS [redacted] to verify safing.

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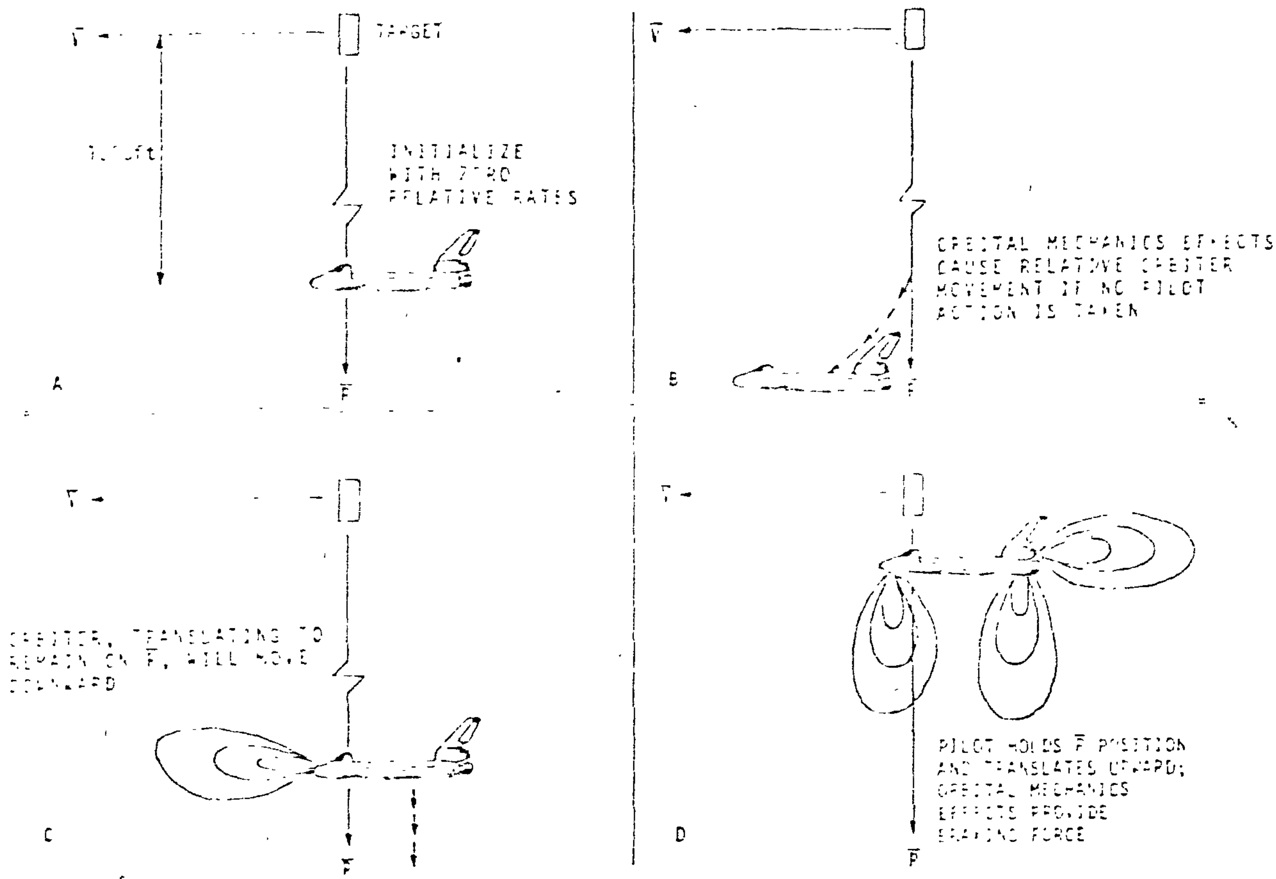

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Figure 5.~~6~~ R-Bar Approach
6

After verification of SV safing, the Orbiter can proceed to a C.G. separation distance of 35 feet at .1 ft. per sec. rate. At this distance the RMS should be extended toward the grapple pin for SV capture. Figure 5.4 shows the sun angle to be in a position where the Orbiter is shading the SV with the crew looking away from the sun. Attitude control on the SV has now been turned off for about 600 seconds at the point where the Orbiter is about to grapple the SV.

At this point it is recommended that the Orbiter attitude control be turned off so that ACS firings do not effect the RMS operators motion. It should stay off during loose grapple and rigidizing to prevent interaction. These internal torques shall cause rates to build up about the cominbed system C.G. The Orbiter attitude control system can now be turned on to stabilize the system in an inertial attitude.

With the Orbiter inertial, the RMS extends the SV such that the solar array is fully extended. As in Figure 5.4, the system stays inertial for approximately 1/4 rev so that the solar array is closest to the Earth for safe separation. In this orientation the solar array trajectories are shown in Figure 5.4 for various conditions. Relative drag effects on the solar arrays and the Orbiter have been analyzed to show that the solar arrays will not impact the Orbiter as long as maneuvers are avoided for the next rev.

Further consideration may have to be given to crew determination of the SV relative attitude. On the dark side, at large distance running, lights may have to be put on the SV to aid the proximity operatons.

To determine the actual distance between the Orbiter C.G. and the SV C.G. a keel camera may have to be placed in the cargo bay. A target on the SV can be placed so that camera pan and tilt angles can be used for ranging at close distances. The crew would then fly the Orbiter for station-keeping to stabilize to camera and COAS angles prior to an attempted grapple, PER FIG 5.5.

5.5.1 Attitude Control

For station-keeeping after rendezvous and attitude control has been safed, the Orbiter should maintain attitude using the verniers and avoid translations in the normal mode. This would have a minimal effect on the SV attitude movement.

To evaluate SV safing and attitude errrors prior to grapple, tests were run on the SV after the attitude control system ACS was shut off. The results of the 4 tests were tabulated in Table I. These results agreed fairly well with the analysis. Initially, the rates at turnoff were low during the tests, resulting in a slow builup of errors. In the analysis, the initial condition (I.C.) for 0^{o}/sec. and 1σ were evaluated. As a result, it is not recommended that any longer than 15 minutes be considered as a time between ACS turn off and grapple. The time lines shown herein reflect that limitation.

Resynchronization of the SV was accomplished within 400 seconds in all cases when the ACS was off for less than 20 minutes. However, it is noted that in no case tested did the SV drift beyond the field of view of the horizon sensors in 20 minutes. The analysis does indicate that the pitch position errors, with initial rates of 1σ, would exceed this sensor FOV. As a result, capture would take longer than demonstrated in the tests.

Therefore, it is recommended that in order to meet the RMS grapple attitude requirements (less than 20 degrees of error with a rate error of less than $.1^{o}$/sec.), the SV be grappled in less than 15 minutes from its turn-off time. If grapple cannot be accomplished by then, that the Orbiter back away from the SV, and that the SV ACS be turned on in 20 minutes. Caputure should be initiated in a roll search mode to protect against Orbiter plume impingement disturbances. Five thousand one hundred (5,100) seconds should be allocated for capture time and SV stabilization, if roll search is to be initiated.

5.5.2 Contingency Grapple Considerations

Due to the nature of the mission at this time, contingency operations need to be planned in case of failures or delays in timing of events.

TABLE I

ATTITUDE CONTROL TESTS ON 9030 AND ANALYSIS

TEST CONDITIONS		ATTITUDE ERROR				RATE ERROR									CAPTURE	
ID #	OFF DURATION	DEG IN 900 SEC		DEG IN 1200 SEC		Tc (O/S)			900 SEC (O/S)			1200 (O/S)			METHOD	TIME $\theta<1°, \dot{\theta}<.05°$
		R	P	R	P	R	P	Y	R	P	Y	R	P	Y		
3312	15 MIN	-4.8	+7.2	--	--	.002	-.0015	.005	-.017	.023	-.003	--	--	--	SYNC	30+ 100 <400 SEC
3313	15 MIN	-6.7	+7.0	--	--	.001	-.0015	-.001	-.017	-.021	-.004	--	--	--	SYNC	60+ 60 <400 SEC
3324	20 MIN	-4.5	6.8	-9.0	+12	+.002	-.001	+.0003	-.014	-.019	-.0012	-.024	-.034	-.001	SEARCH	130+ 120 <400 SEC
3326	20 MIN	-6.7	+6.7	-12.5	+14	.0005	-.0015	-.0012	-.018	+.019	-.0028	-.026	-.035	-.002	SEARCH	140+ 120 <400 SEC
MAX VALUES		6.7	7.2	12.5	14	.002	.0015	.005	.018	.023	.004	.026	.035	.002		140+ 120 <400
I.C. = [illegible]°/SEC ANALYSIS		4	7	9	22	0	0	0	.008	.02	.038	.024	.06	.09		
I.C. = 1[illegible]°/SEC		14	22	12	58	.01	.005	.005	.028	.043	.085	.052	.1	.21		

[redacted]

5.5.4 Rendezvous Time

If it takes longer to approach the SV using the rendezvous techniques as proposed, there is enough extra time for a seocnd POGO pass before a meal. This time is shown on Figure 5.~~1~~3 to be PET at HR-71. In case the operation is delayed further, the approach can be made after lunch during POGO-COCK or POGO-HULA passes.

(b)(1)
(b)(3)
10 USC ⊥ 424

5.5.5 One Day Delay

For a case where crew illness or activity delays the grapple operation by a day, the same operations can be done the next day. Since the SV is in a daily repeat cycle, the RTS contacts will be the same only a day later (see Figure~~s~~ ~~5.1 and~~ 5.4).

5.5.6 Primary Attempt Failure

If the primary attempt at grappling the SV fails before lunch, the Orbiter must backoff and the SV-ACS turned on (see Section ~~5.6~~ 5.5.1). This contingency can occur on either the 3rd or 4th day. Figure 5.7 shows the time line if the Oribter needs to back away from the SV. The Orbiter will back away to a safe distance (greater than 1000 ft.) and the ACS turned on [redacted] the next POGO contact. After acknowledgement of SV control, the crew would have a meal.

After the meal, the crew would proceed to rendezvous again with the SV and grapple in a manner described in the original time line. Only in this case the grapple attempted would be made on a POGO-COCK pass.

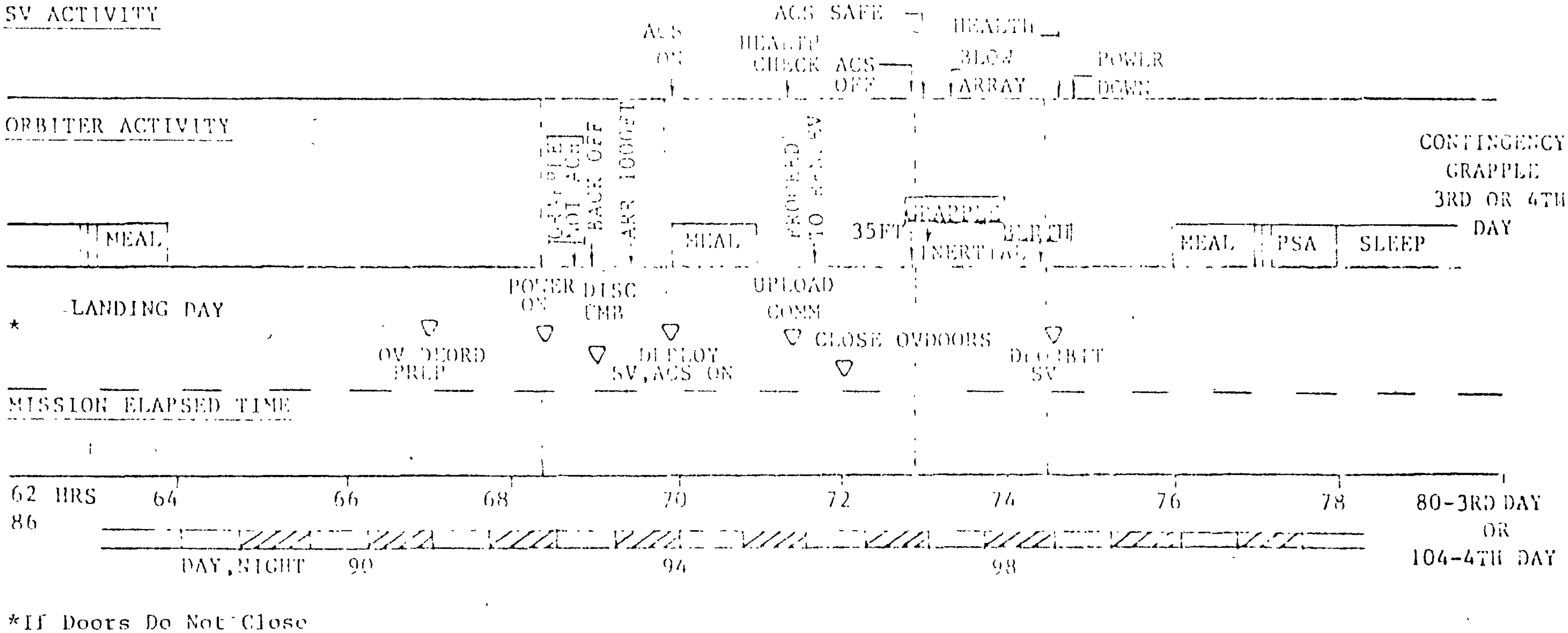

Figure 5.7 Contingency Grapple Timeline

If the ground crew, NASA and SPO need to evaluate the cause of failure, the crew would back off to a safe distance for sleeping. SV changes and crew activity plans can be evaluted during the crew sleep cycle in preparation for the next day's attempt.

5.5.7 SV Deorbit

If for some reason the SV must be deorbited after a berthing attempt, the landing day contingency is also shown in Figure 5.7. The time line shows the SV reaction over RTS with deorbit occuring near HULA as shown.

5.6 TBS

5.7 Post Berthing Activities

In the event an EVA is required, the following constraints must be taken into account:

Extravehiccular Activities (EVA's)

1. All EVA preparation activities will be conducted at a cabin pressure of 10.2 pounds per square inch absolute (psia).

2. EVA's will not be performed on orbits that pass through the high radiation regions of the South Atlantic Anomaly.

3. EVA exposure to the high latitude ($>50^{o}$) will be minimized.

4. Direct sunlight on the Manned Maneuvering Unit (MMU), while in the payload bay, will be limited to TBD minutes duration.

5. All electrical equipment (including the Ku-band, payload bay lights, and the RMS) in the payload bay, except the television cameras, will be inhibited during extravehicular activities.

After berthing the SV is to be powered down. This is done to minimize the battery power drain while still retaining a command capability to deboost the SV, if the doors do not close. Verification of latching and power removal can be done via downlink, AFDP indicators or the extinguishing of indcator lights on the SV. Indicators on the SV must be visually observable from the AFD.

6.0 RMS EVALUATION

Successful operation of the RMS is essential to the retrieval mission. As a pathfinder for other programs, its use must be proven on our mission. DoD has expressed its concern over the many areas where the RMS does not conform to DoD standards. In several areas NASA and DoD are working on a joint approach to the problems. This evaluation will indicate major problems and whether a solution is being worked or is proposed.

6.1 RMS Failure Mechanisms

Several hundred single point failures have been identified in the system mechanization for the arm. If th arm fails and cannot be stowed, the doors cannot close; thus, creating a safety hazard. NASA's position is that they can avoid this by separting the RMS shoulder joint. However, if the RMS fails with a payload attached, the SV must then be deorbited with the RMS attached. This solution does not seem acceptable to DoD.

To avoid most of the single point failure possibility, NASA has developed a sparing philosophy which will enable on-orbit replacement. Although there are weight penalties involved, this approach is recommended as a contingency for our mission.

6.2 End Effector Failures

Several failures have occured during qualification and on-orbit testing of the end effector. Many of the problems relate to materials, quality and process control DoD believes that by having an end effector built to DoD manufacturng and material specificaions, inspection, testing, accountability on a DoD procured end effector may improve the sitation. However, no attempt has been made to redesign the end effector to make its mechanical systems fail-operational.

6.3 Removable Grapple

Since the end effector has not been designed to be removable, a payload would have to accommodate the separated RMS during deboost.

To protect against this problem, an EVA compatible removal grapple has been designed. NASA has a design which the contractor is evaluating. This approach would avoid a deboost problem if the end effector fails to release prior to berthing. If the end effector fails after berthing, the mission is still accomplised and NASA does not have to separate (and lose) the RMS. There is no consideration of an automatic grapple separation due to the introduction of added failure mechanisms.

A consideration may be given to provide a second grapple attachment or a replaceable/removable grapple. This was considered for a condition when the SV is to be deployed after an RMS failure. Since this is two-failure protection, a requirements for a dual grapple is not imposed on the SV.

6.4 Lack of Control Authority

The software mechanization is such that in some regions there is a lack of control. In some cases the amount of control is reduced by a factor of 100. This has severe implications in that it represents areas to stay out of operationally. A contractor is presently mapping out these regions. We will have to verify that with our grappling and berthing scenario the SV avoids these areas. NASA has suggested that these areas be transversed rapidly for small payloads. However, for large payloads, the situation is different. It was suggested that with the offset grapple we now have (15 deg. from center), the problem may be avoided.

However, there is no effort to design DoD software to eliminate the singularities in the present mechanization.

6.5 Dynamic Motion

The RMS was specified originally to be capable of stopping a 32,000 pound payload in 2 ft. However, to date, a total simulation of the interdynamics has not been run. This is needed to evalute Orbiter control system stability when a payload is being grappled; when the Orbiter is being maneuvered; during station-keetping, etc. If the Orbiter is stable, how much distance would be needed for payload dynamic motion clearance.

Presently, there are no simulation results available to validate the operational use of the RMS. Some case studies are planned, but the results are only as good as the fidelity of the computer model. The JSC computer model to be used for these case studies has not been validated.

6.6 Constrained Motion

Tests run at SPAR end simulation at Grumman under conditions representing berthing have shown oscillatory conditions. Both during engaging and release from the latches, RMS joints have oscillated at 2 Hz frequency for up to 40 seconds. During this period, the payload needs to transfer high torques from the point of constrained motion (trunnion during berthing) to the RMS grapple attachment. These torques are up to 300 ft.-lbs. and forces of 100 lbs. at the attachments. These oscillations cause excessive gear wear resulting in premature joint failures mechanically.

This problem has not been fully addressed, nor is there a recommended solution. For our mission, the SV must be modified to accept the load transfer torques without deformation. In the event of a joint failure during the mission, the SV can be separated using the removable grapple during an EVA.

6.7 Validation Plan

The validation of the RMS model with on-orbit performance has not been planned to DoD satisfaction. In order to evaluate future operations, a valid simulation must be created and verified. Enough fidelity must be present so that operations can be planned with a high degree of confidence. If our mission is part of this validation procedure, a simulation must be created, and test cases run to exercise the full range of operational scenarios.

Considering that the cargo bay was sized to accept our vehicle, no large payload simulations have been run for retrieval to date. A control system model exits of the Orbiter, as well as a hi-fidelity model of the RMS joints. It would not be difficult to factor into a combined simulaion the dynamics of our payload. Case studies on a Monte Carlo basis can be run to evalute overall system interactons. These same cases can be run on orbit and instrumented to validate the model.

A good model can then be used to evluate hardware and software changes to improve on problems identified in Sections 6.4 and 6.5.

7.0 TBS

8.0 TBS

9.0 TBS

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APPENDIX A

[redacted] Memo

30 Dec 82

A82-5422.0-04

Handle Via Byeman
Control System Only

Approved for Release: 2017/02/08 C05094769

APPENDIX B

RETRIEVAL ORBIT STUDY

1.0 Orbit Transfer (18 July 1985):

1.1 The transfer from the mission orbit to a six month storage orbit requires 222 ft. per second delta velocity and consumes 688 pounds of OAS propellant.

a. Mission Orbit:

Period	= 88'48.1"
Period altitude	= 94.3 NM
Apogee altitude	= 151.3 NM
B-factor	= .575

b. Storage Orbit

Period	= 91'8.1"
Perigee altitude	= 181.2 NM
Apogee altitude	= 186.0 NM
B-factor	= .716

1.2 It is estimated that 154 fps delta velocity (477 lbm of propellant) would be required to transition to a 90 day, 171.4 NM circular storage orbit.

2.0 Storage Orbit (18 July 1985 - 15 Jan 1986):

Day	Circ Alt (NM)	Period	Beta Angle
-180	183.7	91'8.1	16.8°
-168	182.4	91'5.3"	18.1°
-137	181.3	90'57.2"	20.0°
-107	173.8	90'48.4"	20.5°
-92	171.8	90'44.1"	20.5°
-76	168.8	90'38.2"	21.0°
-61	167.8	90'32.8"	21.8°
-46	167.4	90'26.0"	23.4°
-31	163.7	90'19.4"	25.1°
-15	157.6	90'10.2"	27.5°
0	153.8	90'1.6"	29.4°

4.2 Day 5 and Day 6

Sub Cycle	RTS	Rev	Acq GMT	Dur Sec	Max Elev	Rev	Acq GMT	Dur Sec	Max Elev
1	INDI	65	1922	322	4.6	81	1921	329	4.9
	POGO		1946	462	17.4		1945	462	17.4
	HULA		2003	343	5.5		2002	335	5.2
2	POGO	66	2114	458	16.4	82	2113	457	16.4
	HULA		2132	412	10.7		2131	414	11.1
3	POGO	67	2243	478	22.6	83	2242	477	22.5
4	POGO	68	0012	501	50.6	84	0011	500	50.0
	GUAM		0034	485	72.9		0033	484	68.2
5	BOSS	69	0134	389	8.1	85	0133	384	7.8
	POGO		0141	501	43.2		0140	500	43.7
6	BOSS	70	0303	473	21.6	86	0302	475	22.4
	POGO		0311	454	14.2		0310	454	14.3
7	COOK	71	0435	99	0.3	87	0434	67	0.2
	POGO		0442	338	4.8		0441	339	4.8
8	COOK	72	0600	494	61.2	88	0559	495	64.7
	POGO		0616	36	.04		0615	42	.06
	INDI		0638	477	36.9		0637	477	39.1
9	HULA	73	0730	221	1.9	89	0729	208	1.6
10	HULA	74	0857	465	20.7	90	0856	467	21.6
11	--	75	--	--		91			
12	GUAM	76	1156	446	15.7	92	1155	443	15.0
	POGO		1219	275	2.9		1218	271	2.7
13	GUAM	77	1326	233	2.1	93	1325	246	2.3
	POGO		NA				1347	420	9.9
	BOSS						1357	172	1.0
14	POGO	78	NA			94	1516	488	28.0
	BOSS						1524	491	71.1
15	POGO	79	NA			95	1645	502	77.5
	BOSS						1656	102	0.4
16	INDI	80	NA			96	1750	392	8.5
	POGO						1815	484	27.7
	COOK						1826	489	75.5

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4.2 Day 3 and Day 4

Sub Cycle	RTS	Rev	Acq GMT	Dur Sec	Max Elev	Rev	Acq GMT	Dur Sec	Max Elev
1	INDI	33	1923	313	4.3	49	1923	317	4.4
	POGO		1947	464	17.5		1946	463	17.5
	HULA		2004	354	6.0		2003	349	5.8
2	POGO	34	2116	460	16.5	50	2115	459	16.5
	HULA		2133	410	10.3		2133	410	10.5
3	POGO	35	2244	479	22.9	51	2244	478	22.8
4	POGO	36	0013	502	51.5	52	0012	502	51.1
	GUAM		0034	488	79.6		0034	487	76.8
5	BOSS	37	0136	394	8.5	53	0135	392	8.3
	POGO		0142	502	42.6		0142	501	42.9
6	BOSS	38	0304	472	20.7	54	0303	472	21.1
	POGO		0313	454	14.1		0312	454	14.1
7	COOK	39	0436	130	0.6	55	0436	118	0.5
	POGO		0444	338	4.7		0443	338	4.8
8	COOK	40	0602	494	56.6	56	0601	494	58.5
	POGO		0617	32	.03		0617	338	.03
	INDI		0637	477	34.2		0639	477	35.3
9	HULA	41	0731	238	2.2	57	0731	231	2.1
10	HULA	42	0859	462	19.4	58	0858	463	19.9
11	--	43	--	--		59			
12	GUAM	44	1157	450	16.7	60	1156	448	16.3
	POGO		1220	281	3.0		1219	279	2.9
13	GUAM	45	1328	213	1.7	61	1327	222	1.8
	POGO		1349	426	10.4		1349	424	10.2
	BOSS		1400	212	1.6		1359	202	1.5
14	POGO	46	1519	492	29.1	62	1518	491	28.8
	BOSS		1527	494	62.5		1526	493	64.5
15	POGO	47	1648	505	75.0	63	1647	504	75.6
	BOSS		1659	44	.07		1658	630	0.1
16	INDI	48	1753	405	9.6	64	1752	401	9.3
	POGO		1817	486	27.4		1817	486	27.5
	COOK		1829	493	87.5		1828	492	84.4

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4.2 Day 1 and Day 2

Sub Cycle	RTS	Rev	Acq GMT	Dur Sec	Max Elev	Rev	Acq GMT	Dur Sec	Max Elev
1	INDI	1	1924	309	4.2	17	1923	310	4.2
	POGO		1947	465	17.6		1947	464	17.5
	HULA		2005	359	6.2		2005	357	6.1
2	POGO	2	2116	461	16.6	18	2116	460	16.6
	HULA		2134	411	10.3		2133	410	10.3
3	POGO	3	2245	480	23.0	19	2245	480	23.0
4	POGO	4	0014	503	51.9	20	0013	503	51.8
	GUAM		0035	490	82.1		0035	489	81.4
5	BOSS	5	0136	396	8.6	21	0136	395	8.6
	POGO		0143	503	42.4		0143	502	42.4
6	BOSS	6	0304	471	20.4	22	0304	471	20.5
	POGO		0313	455	14.1		313	455	14.1
7	COOK	7	0437	140	0.7	23	0436	137	0.7
	POGO		0455	339	4.8		0444	338	4.7
8	COOK	8	0603	494	54.9	24	NA		
	POGO		0618	38	.05				
	INDI		0640	479	33.3				
9	HULA	9	0732	244	2.3	25	NA		
10	HULA	10	0859	461	18.9	26	NA		
11	--	11	--	--		27			
12	GUAM	12	1158	452	17.2	28	NA		
	POGO		1221	285	3.0				
13	GUAM	13	1329	203	1.3	29	NA		
	POGO		1350	429	10.5				
	BOSS		1400	224	1.8				
14	POGO	14	1520	494	29.5	30	1519	493	29.3
	BOSS		1528	496	60.5		1527	495	61.2
15	POGO	15	1649	506	74.4	31	1649	506	74.6
	BOSS		1700	26	.02		1700	30	.03
16	INDI	16	1754	408	10.0	32	1753	407	9.8
	POGO		1818	488	27.4		1818	487	7.4
	COOK		1830	495	89.1		1830	494	89.6

3.0 Retrieval Orbit (15 Jan 83 - 9 Feb 83):

Day	Circular Alt NM	Period	Long. ASC Node	Cross Track In N Days	Walk, NM In 8 Days
0	153.2	89'59.8"	64.05°	0	88.2
1	152.7	89'59.1"	64.07°	1.2	112.2
2	152.2	89'58.4"	64.13°	4.8	136.2
3	NA	--	--	--	--
4	151.2	89'56.9"	64.4°	21.0	184.8
5	150.6	89'56.2"	64.61°	33.6	209.4
6	150.1	89'55.5"	64.86°	48.6	238.2
7	149.7	89'54.7"	65.17°	67.2	261.0
8	149.3	89'53.9"	65.53°	88.8	286.8
9	148.9	89'53.2"	65.94°	113.4	313.8
10	NA	--	--	141.0	340.8
11	NA	--	--	171.6	368.4
12	147.9	89'50.8"	67.48°	205.8	396.0
13	NA	--	--	243.0	424.8
14	NA	--	--	283.8	453.6
15	146.9	89'48.3"	69.52°	328.2	483.0
16	146.7	89'47.4"	70.32°	375.6	513.0
17	146.4	90'46.5"	71.17°	427.2	543.0
18	146.1	89'45.6"	72.08°	481.8	573.6
19	NA	--	--	540.0	604.8
20	145.5	89'43.8"	74.08°	601.8	636.0
21	145.2	89'42.8"	75.18°	667.8	667.2
22	144.8	89'41.9"	76.34°	737.4	698.4
23	144.6	89'40.9"	77.57°	811.2	729.6
24	144.6	89'39.9"	78.86°	888.6	760.8
25	143.9	89'38.9"	80.22°	970.2	792.2

NOTE: Average B-factor used for the retrieval orbit was .716.

4.0 Retrieval Orbit RTS Acquisition Schedule:

4.1 Daily Comparison

Day	Rev	POGO Acq.	POGO Dur.	BOSS Acq.	BOSS Dur.
1	14	1520Z	494	1528Z	496
2	30	1519Z	493	1527Z	495
3	46	1519Z	492	1527Z	494
4	62	1518Z	491	1526Z	493
5	78	NA		NA	
6	94	1516Z	488	1524Z	491