"Mission 3A and Mission 3B", from JSC-07896 vol III, "Space Shuttle Baseline Reference Missions", JSC Internal Note No. 73-FM-47, Mar 26, 1973.

Study Guide: One of the early shuttle mission concepts was for a very short military mission in high inclination orbit. The AF was never clear about just WHAT was to be deployed or retrieved but it was probably something likely to get the Russians REALLY mad, which is why the shuttle would have to land immediately afterwards. This mission was a rendezvous design driver for a few years but about 1976-7 the Pentagon said "Never Mind!" and nobody ever thought of it again.

Points to Ponder: The mission profile looks a lot like the very earliest concepts for space station rendezvous, way back in the early 1950's. Note the similarities in terminal guidance techniques (as if the intervening two decades hadn't even happened!). Note also that the "rendezvous sensor" was still not specified and that the shuttle's control axis was therefore not known.

Footnotes to History: References to RCS plume problems appeared at least this early (see them?). The invention of "DAP LOW Z" was one result.

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Volume III

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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SPACE SHUTTLE SYSTEM BASELINE REFERENCE MISSIONS

VOLUME III - MISSION 3A AND MISSION 3B

MISSION PLANNING AND ANALYSIS DIVISION

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2.0 INTRODUCTION

Many aspects of the space shuttle depend upon mission operations. Since the shuttle is to be a workhorse vehicle capable of performing a wide spectrum of earth orbit operations, no single reference mission exists by which to define the total design requirements for the shuttle; on the contrary, the space shuttle must be designed to perform several types of missions.

Originally, the space shuttle request for proposal (RFP) defined three reference missions that were expected to be typical of shuttle operations. Although these missions do not cover the entire spectrum of total system design, they do provide a common basis for systems design and for operational planning. Preliminary profiles for mission 1, a geosynchronous satellite placement mission, and for mission 2, an orbiting element resupply mission, are in reference 1. No mission profile was developed specifically for mission 3.

Since the publication of these preliminary reference mission profiles, two major decisions have been made, requiring that the three baseline missions be reevaluated and updated. First, the shuttle has been changed from a fully-reusable vehicle to a configuration using staged, solid rocket boosters and an external, main propellant tank. Second, reference mission 3 has been redefined as two missions, designated 3A and 3B, that are representative of Air Force requirements upon the shuttle. Mission 3A is defined as a one-revolution satellite deployment and return flight. Mission 3B is defined as a one-revolution satellite retrieval and return flight.

The Mission Planning and Analysis Division (MPAD), with cooperation from the Crew Procedures Division (CPD) and the Spacecraft Design Division (SDD), has begun an effort to design and analyze the baseline missions. The purpose of this effort is to establish requirements for shuttle hardware, software, and operations. As a result, this document, comprising three volumes, is being issued to update the reference missions as currently defined for use with the PRR shuttle configuration. Mission 1, the geosynchronous satellite placement mission, is updated in volume I; mission 2, the orbital element resupply mission, is updated in volume II; missions 3A and 3B, the one-revolution payload deployment and the one-revolution payload retrieval missions, respectively, are defined in this volume.

Each volume contains a summary of the nominal mission profile, including events time line, attitude profile, crew activities (where applicable), and ΔV requirements; a listing of groundrules, guidelines, and assumptions; a detailed analysis of each mission phase, including pertinent trajectory data, mission phase design philosophy, abort constraints, and dispersion analyses; and a discussion of any unresolved problem areas and open items. The space shuttle configuration, aerodynamic data, and engine performance characteristics used in the mission design are given in appendix A of each volume.

The document will soon become a program control document of the Space Shuttle Program Office at the Lyndon B. Johnson Space Center. It will be updated and revised as required to reflect controlled changes to the space shuttle design, mission objectives, and operational requirements. Requests for detailed trajectory printouts or for additional mission data should be directed to the Mission Integration Branch (FM2) of MPAD at JSC. Please note: Due to the fluid state of the current vehicle design, not all the required data were available for the mission designs. Where this occurred, assumptions were made and noted as such.

5.1 Mission 3B Objective and Requirements

For mission 3B, the primary objective of the shuttle is to retrieve a satellite from near earth orbit and to return with it to the launch site (WTR) one revolution after lift-off. (The satellite is 10 feet in diameter, 60 feet long, and weighs 25 000 pounds; it is automated and propulsive; and it is in a 100-n. mi. circular orbit inclined 104°.) To achieve this objective, the space shuttle must be able to meet the following mission requirements:

a. The shuttle must be launched southbound from WTR with no payload on board and inserted at a significantly higher altitude than the standard 50- by 100-n. mi. orbit. The precise insertion targeting parameters are functions of the target orbit and the rendezvous scheme employed.

b. The capability must exist to perform a once-around abort.

c. Disposal of the ET must ensure safe earth impact.

d. The plume from the ET solid rocket motor must not impinge upon the payload. For the purposes of this analysis, the payload is protected by maintaining a specified minimum distance between it and the external tank during tank deorbit. The scheme for this mission and consequently this description depend highly upon the magnitude of the range assumed for payload shielding. The larger the range, the more demanding the shuttle propellant becomes and, because of time line constraints, the less likely the mission becomes. For this study, a minimum of 6 n. mi. is assumed adequate.

e. Tank disposal activities must not seriously hamper rendezvous and retrieval activities.

f. The orbiter insertion and knowledge of payload ephemeris must be stringently accurate.

g. The payload is maneuverable for prephasing.

h. This mission has no lighting constraints; however, beacons on the payload are not desirable. This requirement relates to the type of rendezvous sensors on board the shuttle.

i. Based on the brevity of this mission and the activities that must be compressed into the time line, no ground tracking, orbital navigation, or rendezvous navigation (that is, maneuver solutions from onboard state vectors) are assumed. Rendezvous must be accomplished without the terminal phase or midcourse targeting traditionally supplied by the G&N system. The crew will monitor relative measurement readouts and, in effect, perform the braking phase by using basically a "brute force" technique. In addition, the orbiter is assumed to perform deorbit without a navigational update.

j. The orbiter must have the capability to deorbit with one OMS engine out and yet provide an adequate amount of free fall time. This is based on the 25 000-pound payload on board.

k. No requirement for the use of the radiators is assumed. Instead, adequate water is assumed to be loaded on board.

5.2 Mission Summary

Mission 3B is defined as a payload retrieval and quick return mission. The total mission duration is one revolution, or approximately 2 hours. The sequence of events composing this mission, that is, ascent, orbital operations, and entry through landing, is illustrated on figure 5.2-1. This sequence is time critical: time allotted for one mission event affects the time of and activities of previous and subsequent events.

The space shuttle, with no payload on board, is launched from WTR on a launch azimuth of 198.6°. Prior to lift-off, the payload (target to be retrieved) will have been maneuvered such that a position compatible with rendezvous is attained. The ascent consists of a short vertical rise for tower clearance, pitchover, SRB staging, throttling (both for control of the down-range insertion position relative to the payload and for control of the excess main engine propellant which would otherwise remain after insertion), a 90° vehicle roll, and, finally, orbit insertion at a g.e.t. of 10 minutes 09 seconds. At insertion, the orbiter is at an altitude of approximately 91 n. mi. in an orbit of 80.5 by 96.8 n. mi.

Orbital operations begin immediately with ET disposal. First, the remaining propellants in the ET are dumped, after which the orbiter is on an intercept trajectory with the payload. Then the orbiter performs an RCS separation maneuver to establish a safe clearance with the ET. Finally, at 14 minutes 09 seconds g.e.t., the SRM on the ET is fired to deorbit the ET into the South Pacific Ocean.

After the termination of the ET deorbit maneuver, the orbiter nulls the separation maneuver with the RCS and the rendezvous operations are begun. The rendezvous is performed manually with the RCS by controlling relative range, range rate, and inertial angular rate between the payload target and the orbiter according to a prescribed schedule. The terminal phase control begins at 17 minutes 39 seconds g.e.t. and continues until 31 minutes 33 seconds g.e.t., at which time the orbiter is 100 feet from the payload. After a period of stationkeeping, maneuvering to retrieval position, and passivation of the payload, the payload is retrieved and stowed at 54 minutes 06 seconds g.e.t.

The final orbital event is the orbiter deorbit maneuver scheduled for 59 minutes 06 seconds g.e.t. The deorbit is performed with one OMS engine and requires a ΔV of 396 fps to achieve the targets for entry interface at 400 000-foot altitude. Deorbit is the only nominal 3B mission maneuver requiring use of the OMS.

Entry interface occurs at 1 hour 11 minutes 57 seconds g.e.t. The entry is a controlled guidance phase designed primarily to minimize the weight of the thermal protection system. Thus, the minimum entry target range is selected that will still accommodate expected trajectory dispersions. The entry flight mode selected for this reference mission consists of flying a constant 30° angle of attack until Mach 6 is achieved. From Mach 6 to Mach 0.9 a slow transition to a low angle of attack is performed to satisfy conditions for the terminal area landing phase.

The terminal area phase begins at 1 hour 43 minutes 45 seconds g.e.t. at an altitude of approximately 50 000 feet. The terminal area phase is a controlled guidance maneuver designed for the following: efficient utilization and dissipation of energy within the vicinity of the airfield; establishment of a heading along the runway on a predetermined glide slope; establishment of a flare to accommodate final dispersions; and touchdown and braking to a stop on the runway. Mission 3B is completed at 1 hour 49 minutes 49 seconds g.e.t.

Entry

1 Lift-off from WTR
2 Insertion
3 Propellant dump
4 ET jettison and deorbit
5 Initiate terminal control
6 Theoretical intercept
6' Nominal intercept
7 Max intercept
8 Deorbit preparation
9 Deorbit
10 Entry interface
11 Landing at WTR

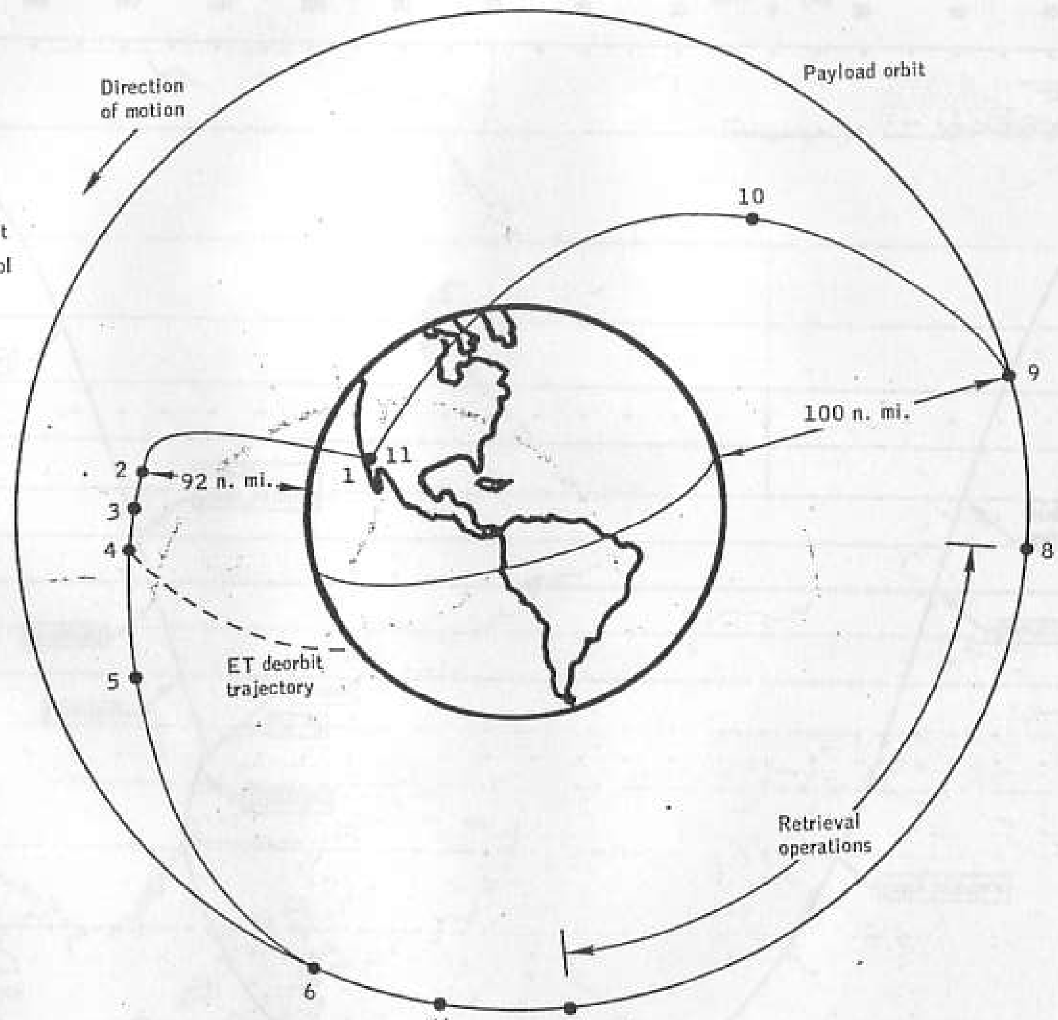

Figure 5.2-1.- Geometrical sketch of mission 3B.

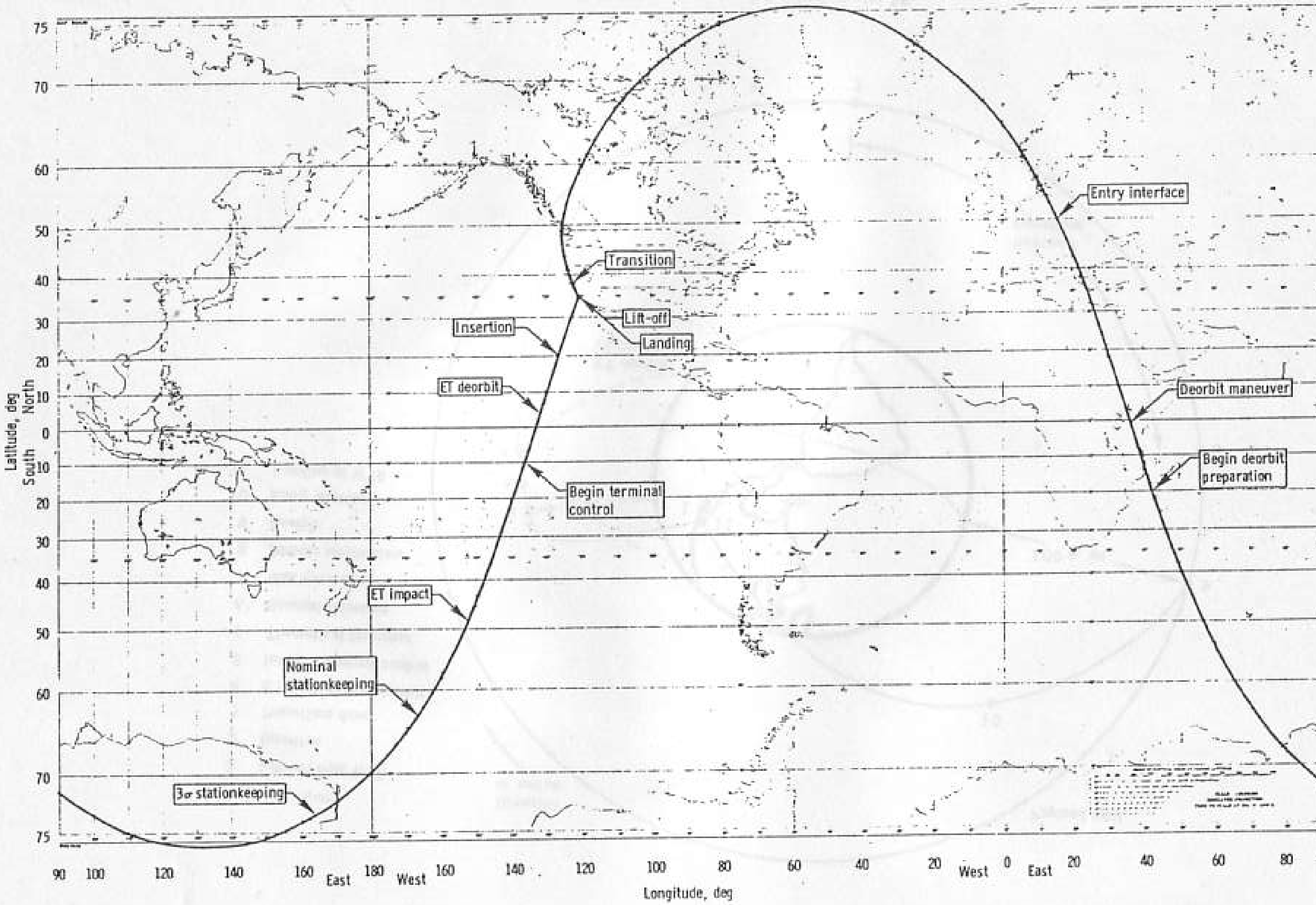

Figure 5.2-3. - Mission 3B groundtrack.

5.5 Rendezvous

No previous rendezvous mission has been flown with time constraints like those imposed by mission 3B; therefore, no flight experience exists upon which to draw. Based upon experience and analysis of past rendezvous missions, however, the design and analytical approach chosen for mission 3B appears to encompass all major operational aspects. The feasibility of this mission can only be determined after extensive dispersion analyses and man-in-the-loop simulations.

5.5.1 Ground rules, guidelines, and assumptions.- The following ground rules, guidelines, and assumptions were used in mission 3B rendezvous design and analysis:

a. The maximum time allowed between insertion and stationkeeping (approximately 100 ft from the payload) is 25 minutes.

b. A minimum range of 6 n. mi. from the payload is adequate to protect against the ET SRM impingement upon the payload.

c. The orbiter inserts on a payload intercept trajectory, with the insertion velocity biased for a nominal propellant dump impulse of 15 fps.

d. The 3σ variation in the propellant dump impulse is $\pm$10 fps.

e. The separation maneuver from the ET is nulled after the ET deorbit burn.

f. Terminal control begins 7 minutes 30 seconds after insertion.

g. The rendezvous sensor is deployed after ET deorbit, requiring 2 minutes to deploy and acquire lock-on.

h. Only the RCS is used for the rendezvous.

i. For the digital analysis, the acceleration level for the RCS was assumed to be 0.4 ft/sec^2 in all axes.

j. Range rate and line-of-sight corrections are made component by component.

k. The rendezvous sensor provides range, range rate, angle, and angular rate information; for this analysis, the information is assumed perfect.

l. The rendezvous is independent of lighting considerations; in addition, the payload is assumed to have no light beacon.

m. No assumption is made as to the type or location of the rendezvous sensor.

n. The actual insertion conditions and the orbiter G&N system knowledge of position and velocity at insertion is assumed to be (1σ) 2000 feet, 2000 feet, 1183 feet, 4 fps, 9 fps, 5.9 fps down range, out of plane, and radially, respectively.

o. The orbiter ascent guidance can control the relative position at insertion between the payload and orbiter to within the accuracy of the estimated state vectors for both vehicles.

p. The 1σ knowledge of the payload position and velocity (based on ground tracking) is assumed to be 3000 feet, 200 feet, 400 feet, 0.2 fps, 0.2 fps, and 3.4 fps down range, out of plane, and radially, respectively.

q. The rendezvous sensor is assumed to have a ranging capability of 10 n. mi.

r. No rendezvous navigation is performed to update the onboard state vectors. The rendezvous is manual, using data directly from the rendezvous sensor.

s. The braking gates have been chosen to nominally provide a 45-second coast period between range rate corrections.

t. The digital analysis has made no attempt to account for propellant usage between the forward and aft tanks nor to account for propellant required for attitude control.

u. The terminal braking has been simulated in a drag-free environment to eliminate from the results the effects of computational accuracy at small relative distances.

5.5.2 Nominal rendezvous trajectory selection.- The selection of the rendezvous trajectory and the resulting insertion targeting has been based upon the following considerations:

a. The 3σ rendezvous time is not to exceed 25 minutes after insertion to provide sufficient time for retrieval operations.

b. The 3σ range to the payload at the time of tank deorbit is not to be less than 6 n. mi. to provide adequate protection from the ET SRM plume impingement upon the payload.

c. The RCS duty cycle is to be minimized for the nominal mission to provide an adequate control authority margin for non-nominal situations, for example, trajectory dispersions, low thrust, and failed sensor.

d. The RCS is used exclusively for terminal control to avoid breaking sensor lock-on.

e. The rendezvous is to be accomplished using manual techniques either in daylight or darkness without a light source on the payload; in addition, because of the time element involved, the rendezvous is to be accomplished without the aid of onboard rendezvous navigation; that is, onboard state vector information is not used to assist in the braking.

f. The effects of dispersions upon the time to acquire rendezvous sensor lock-on are to be minimized to avoid the excessive propellant requirements necessary to meet the 25-minute rendezvous requirement.

The basic rendezvous trajectory employed for mission 3B has been derived from the standard rendezvous profile used on past programs, such as Gemini and Apollo and as currently planned for Skylab and the Apollo-Soyuz Test Project and other shuttle missions. This trajectory is a 130° transfer from a relative elevation angle of approximately 27.5° between the chaser and target spacecraft. This trajectory has the advantage of requiring nearly zero inertial angular rates during the final portion of the trajectory; thus, an efficient manual control technique is provided by maintenance of zero inertial angular rates. Obviously, the time associated with the 130° transfer (approximately 32 min) is not satisfactory for this mission; therefore, a relative position and velocity on this trajectory must be selected so that the time to go (or transfer angle) is less than 32 minutes.

The insertion conditions - position and velocity - are established to be on the selected intercept following the tank disposal operations. This in turn requires the insertion conditions to be biased to account for the ΔV obtained from the nominal tank propellant dump.

Figure 5.5-1 shows the selected insertion point and nominal relative trajectory along with the options available. Illustrated on this figure are the trajectories corresponding to differential altitudes of 10, 15, and 20 n. mi. at the transfer initiation point for a 130° transfer. For each of these trajectories, the relative elevation time history remains the same. As a consequence, the magnitude of the impulsive braking maneuver is a function of range for a constant time-to-go to intercept. As shown on the figure, the trajectory selected is the one corresponding to a differential altitude of 15 n. mi. for a transfer angle of 130°, producing an impulsive braking maneuver of 42 fps. The point on the trajectory for establishing insertion conditions corresponds to a theoretical time to intercept (from insertion) of 18 minutes, or a transfer angle of 73.5°, which is the final 73.5° of the 130° transfer. As a result, the insertion of the orbiter relative to the payload is 6.7 n. mi. behind and 9.4 n. mi. below. This selection is obtained from the convergence of the constraints imposed, namely, the plume impingement range, the maximum time to rendezvous, the RCS acceleration, and the thruster duty cycles during traking.

From solely a nominal trajectory viewpoint, the 10-n. mi. differential altitude would be preferable for the following reasons:

a. It is identical to the final approach being planned for all other shuttle rendezvous.

b. The lower theoretical closing velocity requires less propellant.

c. The slip in rendezvous time due to braking is minimized since for a constant duty cycle the reduced closing velocity allows braking to occur at a later time, that is, at closer range; this in turn more closely approximates an impulsive braking.

The 10-n. mi. differential altitude, however, has the disadvantage of being within the minimum range envelope defined for plume impingement protection and, with regard to dispersions, creates a larger uncertainty in the time required to achieve rendezvous sensor lock-on. As a consequence, the 15-n. mi. approach is employed. The 20-n. mi. differential altitude is indicated on figure 5.5-1 for illustrative purposes only. Its use is not considered primarily because of the high closing velocity required and the rendezvous delay that would result during braking.

The 18-minute theoretical intercept time is basically dictated both by a 25-minute maximum rendezvous time with dispersions and by the 4 minutes required for ET disposal operations. This time appears to provide reasonable time to deploy the rendezvous sensors if deployment is required and to begin terminal control within the manual capability of the crew without excessively large propellant penalties. In addition, the nominal range at ET deorbit is approximately 7.5 n. mi., which, when coupled with dispersions, still adheres to the 6-n. mi. constraint.

During the rendezvous phase, the RCS is used exclusively, maneuvering axis by axis when required. The results of this analysis do not depend upon which axis is chosen for rendezvous braking since all axes were assumed to have the same acceleration capability; however, the profile illustrates braking with the -x RCS thrusters, but acknowledges that z-axis braking could have been shown. If z-axis braking is assumed the orbiter is required to roll to a heads-up position and pitch down approximately 59° following the tank deorbit maneuver. In reality, the braking axis will depend upon the location of the rendezvous sensor, the acceleration capability of the respective thrusters, the field of view associated with each respective axis, and so on. At present no historical preference has been given to one axis over the other.

For a normal (not 3B) rendezvous, the z-axis braking appears to have an advantage in that, for the current design (see appendix A), the z-axis direction has the largest acceleration, the propellant usage would be divided between the forward and aft tanks, some of the rendezvous sensors could possibly be located in the payload bay, and the orbiter would be essentially in a retrieval attitude with the payload bay and manipulator directly facing the payload. Before the principal rendezvous axis is chosen, however, more detailed analyses (including a 6-D vehicle simulation) need to be performed to determine the RCS propellant usage per axis as a function of the principal rendezvous axis.

Past studies on this mission performed by other organizations have suggested the use of the OMS for braking. Operationally, this appears to be very difficult for the following reasons:

a. In the absence of updating the onboard state vector, the thrust direction would be very difficult to establish without prior nulling of the inertial angular rates with the RCS.

b. Sensor lock-on would likely be broken in maneuvering to the burn attitude if it were known.

c. Excessive attitude maneuvering would be required.

d. The delay in taking proper action impacts the total propellant and time requirements.

e. If one OMS engine is assumed, the acceleration capability is similar to the RCS capability.

For the nominal mission, the braking gates (range-range rate) were selected to provide a coast period of approximately 45 seconds between braking corrections based on the assumed vehicle RCS acceleration capability of 0.4 ft/sec^2. This coast time was included to provide a margin for excessive burn times - either for range rate or angular rate corrections - due to dispersions. Additionally, the selection of the braking gates was also based on limiting the 3σ delay in rendezvous time to be no greater than 7 minutes and arriving at a stationkeeping position (100 ft assumed) no later than 25 minutes after insertion. The braking gates are presented in table 5.5-I.

5.5.3 Nominal rendezvous profile.- For this mission the tank deorbit ignition is 4 minutes 00 seconds after insertion. Twenty seconds later the orbiter begins nulling the separation velocity by using the -y RCS to obtain 8 fps. This maneuver nulls the out-of-plane velocity (relative to the payload) that was imparted by the orbiter/ET separation activities (section 5.4). A potential alternative to the nulling maneuver is to insert approximately 1600 feet out of the payload orbital plane with an out-of-plane velocity of 7.8 fps, which is directed so that a node between the orbital planes occurs at the time of separation. Theoretically, the separation maneuver establishes coplanar orbits between the two vehicles.

The null maneuver is completed at 14 minutes 49 seconds g.e.t. At this time, the orbiter initiates deployment of the rendezvous sensor, followed by sensor lock-on. These activities are assumed to require 2 minutes, during which time the orbiter maneuvers to the nominal lock-on attitude (76° above the local horizon assuming x-axis pointed along the line of sight for the rendezvous sensor) and enables local vertical attitude hold. This attitude mode is desirable to expedite payload acquisition and lock-on by the rendezvous sensor. Once lock-on is achieved at 16 minutes 49 seconds g.e.t. (assumed), the orbiter maneuvers to the payload line-of-sight attitude, enables inertial attitude hold, and initiates terminal control at 17 minutes 39 seconds g.e.t. Note that for purposes of this study, the terminal control is assumed to be based strictly on braking gates (range-range rate criteria); however, investigations are currently underway to determine the feasibility of determining a "pseudo midcourse maneuver", from charts or software, directly from sensor data (range, range rate, and angle information) at the beginning of the terminal control phase in order to preserve the intercept time. Subsequent control is accomplished by controlling according to braking gates. Preliminary indications are that the two methods agree in terms of the propellant usage, but the midcourse concept will minimize the slip in rendezvous time. Onboard software may be desirable if the concept proves beneficial.

Nominally the first braking maneuver occurs at 24 minutes 05 seconds g.e.t. and its purpose is to reduce the closing velocity to 40 fps. Subsequent maneuvers null inertial angular rates and reduce the closing velocity to 0.1 fps at 100 feet in steps of 5-fps intervals at the selected range gates. At 31 minutes 33 seconds g.e.t., a 100-foot distance between the payload and orbiter is reached.

Figure 5.5-2 shows the relative motion between the payload and orbiter from insertion to rendezvous (100-ft range). Also included on this figure is the relative motion of the tank following its deorbit maneuver. The ΔV expended for the nominal 3B rendezvous is approximately 45 fps. Of this, approximately 40 fps is used in reducing the closing velocity (range rate) to zero at 100 feet, while 5 fps is required to null the inertial angular rates. The time to stationkeeping (100-ft range) was delayed 3 minutes 24 seconds due to braking, resulting in an arrival time of 21 minutes 24 seconds after insertion.

Figure 5.5-3 shows the thrusting times associated with the nominal rendezvous. Although terminal control begins approximately 7 minutes 30 seconds after insertion, no maneuvering is required for the nominal mission until 13 minutes 56 seconds after insertion. The nominal duty cycle, which is defined as the ratio of the thrusting time to the total time (referenced from the time of the initial braking maneuver), is approximately 21 percent. This parameter is a function of the braking gates employed, and it indicates the delay in rendezvous time that will be encountered; however, a margin must be incorporated to account for off-nominal situations. This margin can then be used for longer thrusting requirements or for brief periods between maneuvering for crew assessment of the situation. Figure 5.5-4 illustrates the range-range rate profile and the associated braking gates. The time history of the relative parameters, for example, range, range rate, and elevation angle, are shown on figure 5.5-5.

The thrusting time line shown during braking and the ΔV of 45 fps can only be achieved from an ideal trajectory. As seen in section 5.5.4, mean RCS ΔV expenditure due to dispersions increases from 45 fps to approximately 83 fps; for a 3σ situation, the ΔV will increase to approximately 143 fps. Additionally, thrusting will likely be required as soon as rendezvous sensor lock-on can be achieved.

5.5.4 Mission 3B rendezvous dispersion analysis.- The purpose of performing a rendezvous dispersion analysis on mission 3B is to evaluate the feasibility of such a mission and to aid in the establishment of vehicle requirements. The feasibility can be evaluated by observing the effect of dispersions on key parameters. Two of these parameters actually serve as constraints because of the nature of the 3B rendezvous; namely,

a. The time to stationkeeping (approximately 100-ft distance from the payload)

b. The minimum range between the payload and the orbiter at the time of ET SRM ignition (insertion plus 4 min)

In addition to these, other key parameters that affect vehicle design and operational requirements were examined. They are

a. RCS ΔV usage during the rendezvous. This knowledge aids in sizing the RCS propellant loading.

b. The RCS duty cycle. This defines requirements on the RCS thrusting frequency and crew activities during the rendezvous.

c. The payload look angle uncertainties. These aid in the definition of the required rendezvous sensor acquisition time, field of view, and scanning capability of the rendezvous sensor.

Prior to the discussion of the dispersions assumed, the methodology used, and

case I varies from approximately 41° to 79°, and case II varies from 27° to 60°.

While the 3B rendezvous dispersion analysis may have some simplifying assumptions, the results are felt to indicate the actual situation. The analysis indicates that with a rendezvous sensor that can provide information (so that range, range rate, and inertial line-of-sight rates can be displayed to the crew), with an acceleration level of 0.4 ft/sec^2, and with the assumed insertion dispersions and vent uncertainties, the rendezvous time constraints and range requirements at insertion plus 4 minutes can be met. To meet these criteria, however, the RCS ΔV must be large enough to handle each rendezvous situation.

In later 3B rendezvous dispersion studies several factors should be considered. They include

1. Errors in the onboard sensors

2. Uncertainties in the braking performance

3. A more detailed analysis of the potential payload acquisition problems

4. Using different thrust accelerations for different axes

5. Selection of the rendezvous braking axis

TABLE 5.5-I.- ASSUMED BRAKING GATES FOR MISSION 3B RENDEZVOUS

Range, ft	RDOT minimum, fps	RDOT maximum, fps
50 635	85.00	95.00
45 615	80.00	90.00
40 855	75.00	85.00
36 365	70.00	80.00
32 135	65.00	75.00
28 160	60.00	70.00
24 455	55.00	65.00
21 010	50.00	60.00
17 830	50.00	55.00
14 910	45.00	50.00
12 255	40.00	45.00
9 860	35.00	40.00
7 730	30.00	35.00
5 860	30.00	30.00
4 255	25.00	25.00
2 910	20.00	20.00
1 830	15.00	15.00
1 010	10.00	10.00
455	5.00	5.00
160	.10	.10

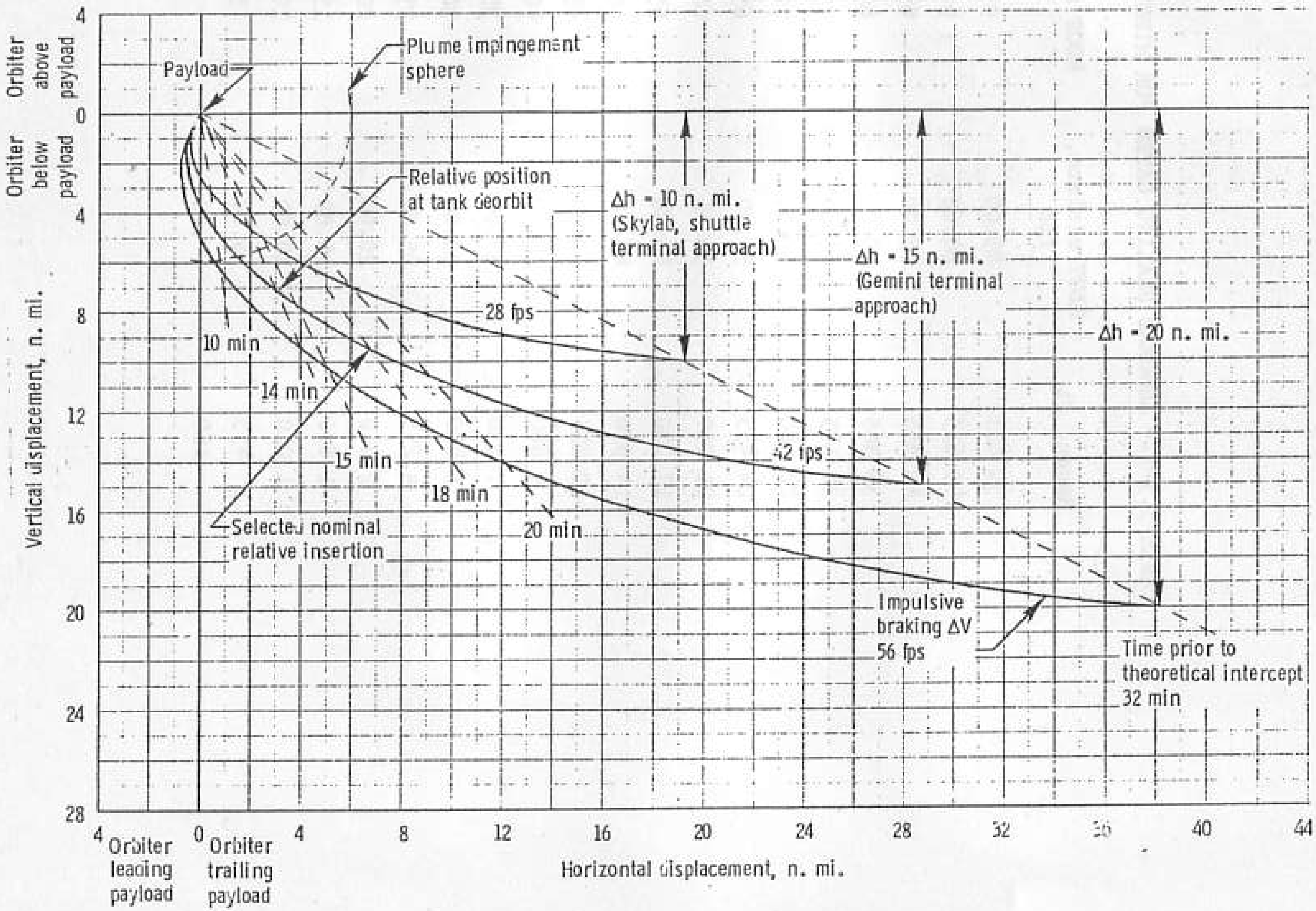

Figure 5.5-1. - Selection criteria for establishment of the mission 3B insertion targets.

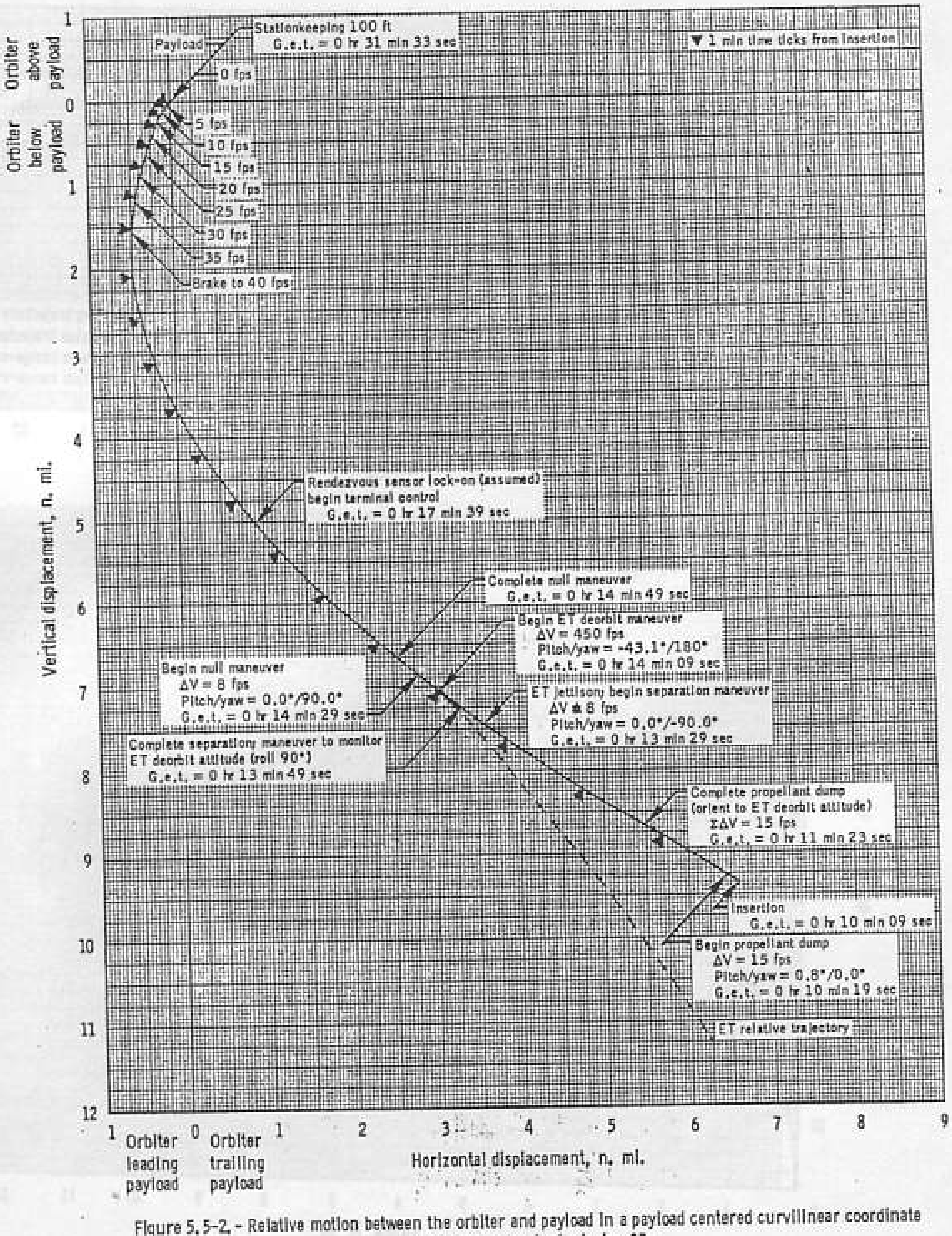

Figure 5.5-2. - Relative motion between the orbiter and payload in a payload centered curvilinear coordinate system for a nominal mission 3B.

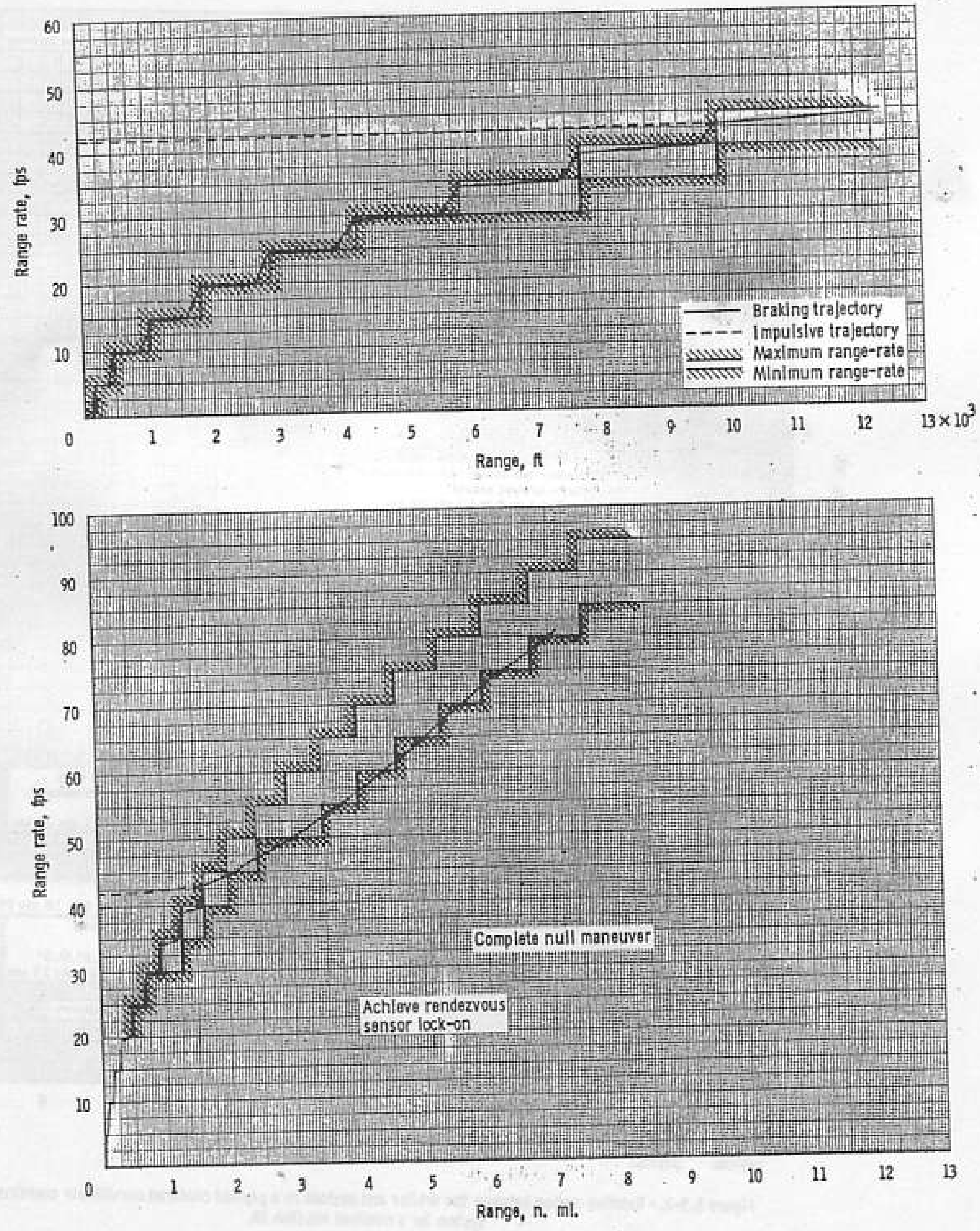

Figure 5.5-4. - Range rate versus range profile during braking phase of mission 3B.

5.9 Existing Problem Areas Related to Mission 3B

Problem areas have been identified that could impact the results of this mission analysis and that could alter the vehicle requirements.

In this section, these problems are categorized by mission phase.

5.9.1 Ascent.-

1. The nominal ascent profile must be shaped to maximize main engine propellant usage rather than minimizing the consumption. This is necessary to satisfy the external tank fuel dump requirements at insertion. The down-range distance incurred from burning off the excessive propellant could result in a possible overshoot of the desired external tank impact footprint.

orbiting payload and interference with the orbiter during the orbiter deorbit phase. This problem is being investigated.

5.9.3 Rendezvous.-

1. Although the analysis of the 3B rendezvous is felt to be realistic, it is by no means a complete analysis. Several simplifying assumptions have been made and additional analyses are required to prove the feasibility of the mission.

2. Re-evaluation of the assumed dispersions should be done, especially for the orbiter insertion dispersions and the vent uncertainty.

3. Accuracies and types of candidate rendezvous sensors should be studied.

4. A more sophisticated analysis of the acquisition problems for candidate rendezvous sensors should be performed, perhaps resulting in a varying time for braking initiation. This includes such problems as deployment of the sensor and sensor acquisition.

5. Uncertainties in braking maneuver execution should be included.

6. A better definition of the plume impingement requirements may change the direction of future studies.

7. Analysis concerning the selection of the rendezvous braking axis should be performed. In addition, different thrust accelerations for the different orbiter axes should be considered as appropriate.

8. Consideration should be given to rendezvous with the payload in a higher orbit to reduce the relative uncertainties due to drag effects, and to provide additional launch opportunities for retrieval without requiring additional payload prephasing (appendix C).

5.9.4 Payload retrieval.-

1. An uncertainty in the retrieval time and therefore in the mission time line exists in the absence of more definitive payload information.

2. The payload attitude during retrieval is currently undefined.

3. A better definition of the required activities between stationkeeping and retrieval needs to be developed.

4. An actual simulation should be performed to verify the retrieval time line.

5.9.5 Orbiter deorbit.-

1. A detailed analysis should be performed to determine the minimum acceptable deorbit preparation time and minimum free fall time; however, the 5-minute dedicated entry preparation is based on experience from previous manned space flights. If

6.0 CONCLUSION

This document has described the trajectory profiles of missions 3A and 3B. The description has been augmented with the objectives, rationale, and analysis of each mission phase. Because of the unique requirements imposed by their respective objectives, both missions have been evaluated in detail. Both appear feasible within the framework of the assumptions, although problem areas or areas requiring further analysis have been identified. Before their feasibility is certain, however, other, different levels of detailed analysis are needed. For example, analysis is needed for crew simulations, training, and crew time lines. Other areas of concern are the simulation of payload deployment and retrieval operations and whether or not these operations are actually feasible within the enclosed time lines. The effect of having the shuttle RCS thrusters deactivated during the deployment and retrieval operation is also unknown.

As a result of the design and analysis of missions 3A and 3B, shuttle design requirements and subsystems that are potentially impacted by these missions have been identified. They are

a. Missions 3A and 3B establish the minimum cross range that the orbiter must be capable of flying during entry. This requirement must be reflected in the thermal protection system.

b. Mission 3A is potentially the mission which sizes the shuttle.

c. Special guidance software will be required to accommodate once-around abort requirements for mission 3A.

d. Mission 3B requires main engine throttling capability for control of both the down range insertion position and the excess main engine propellant remaining post insertion.

e. Mission 3B will require specialized ascent targeting and guidance for controlling the down-range insertion position.

f. The accuracy requirements of the G&N are driven by mission 3B.

g. Mission 3B potentially impacts the ET deorbit SRM design and tank jettison system.

h. Mission 3B potentially impacts the rendezvous sensor requirements in terms of type, range, and accuracy.

i. The RCS is potentially impacted with regard to the tank size, thrust level, and usage, for example, no burn duration constraints.

j. The minimum OMS thrust-to-weight ratio may be impacted by mission 3B requirements.

k. Missions 3A and 3B potentially impact the manipulator design requirements.