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Shuttle Program Standard Maneuver Sequences for Orbiter/Upper-Stage Separation

SSUS-A, SSUS-D, and IUS

(NASA-TM-81058) SHUTTLE PROGRAM STANDARD
MANEUVER SEQUENCES FOR ORBITER/UPPER-STAGE
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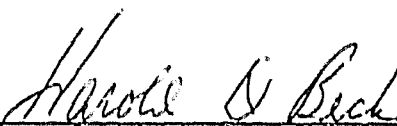
STANDARD MANEUVER SEQUENCES FOR ORBITER/UPPER-STAGE SEPARATION

SSUS-A, SSUS-D, AND IUS

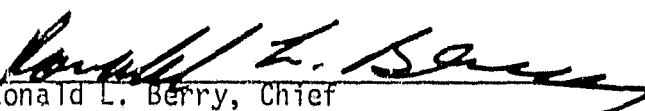
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FOREWORD

This document contains descriptions of standard post-ejection maneuver sequences for the deployment of IUS, SSUS-A, and SSUS-D upper stages from the Space Shuttle Orbiter. The sequences were designed to satisfy requirements for limiting the damage inflicted on the Orbiter by upper-stage exhaust particles, subject to a further requirement for minimizing the impingement of Orbiter thruster plumes on the deployed payload. In all cases it was assumed that the orbital maneuvering system (OMS) engines would be used to apply the Orbiter's major separation velocity increment.

These maneuver sequences represent a "first cut" at the problem of defining a comprehensive, integrated set of standard sequences for upper-stage deployment. As such, they are subject to future modification and refinement. This is particularly true of those applying to the SSUS-A, since the design of that stage requires the remote manipulator system (RMS) to remove a thermal shield and hold it while the Orbiter's major separation velocity increment is being applied. Because of RMS structural limitations, the OMS cannot be used for Orbiter/SSUS-A separations, which will necessitate redesigning the sequence within the primary RCS constraints.

The standard sequences are being published in their current form to serve as a basis for critique by the various disciplines and organizations concerned with Orbiter/upper-stage separation. Aside from any systematic deficiencies, the standard sequences may require modification to satisfy specific payload/flight requirements. However, at the very least they should be useful as points of departure in specific flight design exercises.

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SYMBOLS AND ABBREVIATIONS

CG	center of gravity (center of mass)
HFRMP	High Fidelity Relative Motion Program
IUS	inertial upper stage
I_{xx} , I_{yy} , I_{zz}	moments of inertia
I_{yz} , I_{zx} , I_{xy}	products of inertia
LVLH	local vertical/local horizontal coordinate system
PET	phase elapsed time (measured from instant of upper-stage ejection from cargo bay)
PRCS	primary reaction control system
RCS	reaction control system
SRM	solid-propellant rocket motor
SSUS	spinning solid upper stage
STS	space transportation system
$\overline{\Delta V}$	velocity increment (vector)
ΔV	velocity increment (magnitude)

1.0 INTRODUCTION

Many satellites to be launched by the Space Transportation System (STS) will require the use of upper stages to boost them into orbits beyond the performance envelope of the Orbiter. Three standard stage types (IUS, SSUS-A, and SSUS-D), all using solid rocket motors (SRMs) as their major propulsion units, are currently being developed for this purpose. These stages, with satellites attached (the combination will hereafter be referred to as "the payload"), will be deployed from the Orbiter typically in a geocentric circular orbit at an altitude of about 160 nautical miles.

In the process of deployment, each payload will be directed on a tilt table in the Orbiter's cargo bay, and then ejected by a spring impulse which will produce a relative velocity increment on the order of 0.3 to 3.0 feet per second along the longitudinal axis of the upper stage. The magnitude of the ejection velocity increment will vary with stage type (and to a lesser extent, with payload mass), and the angle it makes with the Orbiter's body-fixed X_B axis will vary from one stage type to another, but it will always (nominally, at least, for all standard stage types) lie in the Orbiter's plane of symmetry.

2.0 SEPARATION MANEUVER ANALYSIS

2.1 POINTING REQUIREMENT AT PAYLOAD EJECTION TIME

The SSUS-A and SSUS-D stages are relatively unsophisticated. Each of these types will be spun up before ejection, and will depend primarily on the conservation of angular momentum to maintain the proper inertial orientation of its SRM thrust vector, i.e., its longitudinal axis. Necessarily then, in the case of any SSUS deployment, the Orbiter's attitude at ejection time will be constrained by a requirement to point the SRM thrust vector in the proper direction with respect to an inertial frame. The proper direction in each case will depend on the characteristics and the objective of the particular satellite that the SSUS is transporting, and must be expected to vary from one deployment to another.

The IUS, having a more sophisticated attitude control system, does not have to be ejected in its SRM burn attitude. However, in some cases at least, the satellite it carries will have to be pointed in some particular direction at ejection time. For instance, the Galileo spacecraft and the TDRS satellites need to be pointed away from the sun to avoid thermal problems. Thus it can be seen that a general deployment technique for any standard upper stage type must be able to accommodate a requirement to point the longitudinal axis of the stage in an arbitrary direction at the time of payload ejection.

2.2 POINTING REQUIREMENT FOR THE SEPARATION MANEUVER

The upper-stage SRMs expel particles in their exhaust plumes which can damage the Orbiter's windows and thermal protection tiles. Evaluation of potential damage from the SRM particles (see Appendix) indicates a need for maneuvering the Orbiter to a considerable distance (on the order of tens of miles) from the upper stage during the interval of time between payload ejection and SRM ignition. The standard length of this time interval is only 45 minutes, which means that a separation velocity increment (ΔV) on the order of tens of feet per second must be applied to the Orbiter soon after ejection of the upper stage. Because of the directionality of the SRM exhaust flow and gravitational effects on the separation trajectory of the Orbiter, the separation velocity increment cannot be applied in

an arbitrary direction. Assuming that SRM ignition occurs 36 minutes after the separation maneuver, analysis indicates that the preferred alignment of the separation velocity increment is approximately 30 degrees above the Orbiter's local vertical local-horizontal (LVLH) X axis at maneuver execution time.

For reasons of propellant economy and also because their use makes it easier to avoid excessive plume impingement on the ejected payload, the Orbiter's OMS engines (fired in unison) are the preferred means of applying the separation velocity increment. The effective line of their combined thrust is essentially fixed in the Orbiter's plane of symmetry, canted downward at an angle of approximately 16 degrees with respect to the X_B axis. During the OMS burn, the Orbiter attitude must be such as to point the body-fixed OMS thrust vector at least approximately in the preferred direction of alignment for the separation velocity increment, as defined in the preceding paragraph with respect to the LVLH reference system.

2.3 RESOLUTION OF CONFLICT BETWEEN POINTING REQUIREMENTS

It is impossible, in the general case, to define a single fixed attitude for the Orbiter that will satisfy both of the pointing requirements delineated in Sections 2.1 and 2.2. This means that an Orbiter rotational maneuver must be executed during the initial phase of the separation sequence, i.e., between the time of payload ejection and the time of OMS ignition.

If the time interval between payload ejection and OMS ignition is known (or fixed arbitrarily), then the preferred direction of the OMS thrust vector (Section 2.2) can be defined with respect to the same inertial frame in which the payload pointing requirement (Section 2.1) is defined. Only two of the Orbiter's three degrees of rotational freedom are used up in satisfying the payload pointing requirement at ejection time. The remaining degree of freedom allows the Orbiter to be rotated arbitrarily about the longitudinal axis of the erected upper stage. This means that it is always possible to define an ejection attitude that will (1) satisfy an arbitrary payload pointing requirement, and (2) cause the inertially-fixed vector representing the preferred direction of the OMS thrust vector to lie in the Orbiter's plane of symmetry.

It follows then that the Orbiter need never have to yaw or roll during the initial separation phase. Proper planning can always arrange it so that, after payload ejection, the preferred OMS thrust direction can be attained by rotating the Orbiter about its own body-fixed Y_B axis, i.e., by performing only a pitch maneuver. This has very definite advantages with regard to (a) avoiding a collision, (b) maintaining visibility of the payload on the part of the flight crew, and (c) preventing excessive plume impingement on the ejected payload.

There remains one major difficulty: the angle of rotation between the ejection attitude and the OMS-burn attitude can be controlled only to a very limited extent by the flight designer. For a given payload pointing requirement, there are only two Orbiter attitudes that will satisfy that requirement and also allow the preferred OMS thrust direction be achieved without having to perform a yaw or a roll maneuver. Once a choice is made between the two available options, the required angle of rotation between the ejection attitude and the OMS-burn attitude is uniquely determined, and the value of that angle may lie anywhere between -180 and $+180$ degrees.

2.4 FLIGHT DESIGN STRATEGY

At this point we must consider a problem that can be explained as follows. Because the nominal length of the total interval between ejection and SRM ignition is fixed at 45 minutes, the time that can be spent in the execution of the post-ejection rotational maneuver is quite limited. The longer the OMS burn is delayed, the shorter the remaining interval before SRM ignition, and the longer the OMS burn itself must be in order to produce the necessary separation distance at SRM ignition time. The longer the OMS burn, the more propellant is consumed, and the longer is the flight path traversed by the Orbiter during the burn. The last point is extremely important in relation to plume impingement on the ejected payload. If excessive impingement is to be avoided, before OMS ignition it is necessary not only to rotate the Orbiter into the proper burn attitude, but also to maneuver it in a translational sense so that the payload position in the Orbiter's body-fixed coordinate system will lie ahead of the Orbiter at OMS ignition time. The magnitude of the required distance is essentially equal to the length of the path traversed by the Orbiter during the OMS burn.

In light of the preceding paragraph it is obvious now that there are important translational as well as rotational aspects to be considered in relation to maneuvering the Orbiter during the initial phase of separation (i.e., from payload ejection to OMS ignition). Furthermore, it is obvious that the time interval allotted to the initial phase must not be too long. On the other hand, the time allotment must not be too short. Otherwise the required rotational rate of the Orbiter might be excessive, large translational velocity increments might be required to attain the necessary separation distance at OMS ignition, or there might be insufficient time to perform the cockpit procedures attending the execution of the OMS burn. Taking all of these considerations into account, for the purpose of designing standard Orbiter/upper-stage separation sequences, a provisional allotment of 9 minutes has been given to the initial phase. This leaves an interval of 36 minutes (minus the length of the OMS burn itself, which may range between 10 and 40 seconds depending on the upper stage type and payload mass) for the Orbiter to coast in its separation trajectory before the nominal time of SRM ignition.

Now we come to the meat of the problem to which this report is addressed: given such a short time (9 minutes) to execute the initial phase of separation, there appears to be no generally-applicable maneuvering technique or algorithm that will accommodate the full range of possible rotation angles (-180 to $+180$ degrees) required of the Orbiter and also satisfy all the other necessary requirements for maintaining payload visibility, avoiding collision and excessive plume impingement, etc. That is to say, in the specific cases that have been studied, it has been found necessary to tailor the initial maneuver sequence (by trial and error) to fit the value of the angle through which the Orbiter is required to rotate between payload ejection and OMS ignition. Not only that, but also it has been found that the maneuver sequence which works for a particular rotation angle and upper stage type generally does not work for another stage type and the same rotation angle. This is because the ejection impulse magnitude -- and equally important or perhaps more so, the ejection angle with respect to the Orbiter's X_B axis -- varies from one stage type to another.

The implications of the preceding paragraph are sobering when applied to flight design and training requirements in the STS operational era, for it would seem necessary to treat each and every upper stage deployment as a special case

requiring its own uniquely-designed maneuver sequence. However, the dilemma is not quite so bad as it might seem, for the following two reasons.

First of all, the initial phase is short enough that the variation of gravitational effects with ejection attitude can be expected to be small. Therefore, given the rotation angle required of the Orbiter, in the process of designing the maneuver sequence it is sufficient to consider only the motion of the payload in the Orbiter's body-fixed reference frame, without having to consider how the Orbiter is oriented relative to the earth. (The design must of course contain adequate tolerances to allow for the small variations due to gravity, as well as those resulting from unavoidable execution errors.) Secondly, analysis indicates that the OMS thrust vector can deviate from its preferred nominal alignment in the LVLH frame by as much as $22\frac{1}{2}$ degrees in any direction (in or out of the geocentric orbit plane) without seriously degrading the effectiveness of the OMS burn, with regard of course to placing the Orbiter in a safe position at SRM ignition time.*

It is possible then to design (for each upper stage type) a set of $360^\circ / 45^\circ = 8$ discrete maneuver sequences, one of which in any given deployment situation can be used to align the OMS thrust vector within $22\frac{1}{2}$ degrees of the preferred nominal direction. This does not mean that, in preparation for a given flight, training would be required for all 8 of the possible maneuver sequences that pertain to a given stage type. Training for a specific deployment would be limited to one particular maneuver sequence that would be selected, prior to the flight, on the basis of the payload pointing requirement. On the basis of expected similarities in the pointing requirements of many SSUS firings, and on the presumption that many IUS ejections will not be constrained as to direction, there is reason to hope that the majority of STS operational deployments may be limited to a small subset (say 2 or 3) of the 8 possibilities that apply to a given stage type. Nevertheless, at this stage of the game it is only prudent to define maneuver sequences to cover all possibilities, which is the subject of the remainder of this report.

*This insensitivity depends on the prior choice of a final separation trajectory that places the Orbiter, at SRM ignition time in the LVLH frame centered on the upper stage, above and behind the upper stage nominally at the maximum distance attainable with a separation velocity increment of fixed magnitude (see Appendix A).

3.0 DEFINITION OF STANDARD ORBITER/IUS SEPARATION SEQUENCES

3.1 ASSUMPTIONS AND DESIGN REQUIREMENTS

1. For trajectory calculation purposes, the mass properties of the Orbiter were assumed to be as listed in Table 3-1 immediately after ejection of the payload from the cargo bay.
2. The payload (IUS+satellite) was modeled as a right circular cylinder 112 inches in diameter and 428 inches in length.
3. The payload will be ejected by springs from a tilt table in the cargo bay, with a relative velocity increment of 0.40 feet per second in the direction of the IUS +X body axis (which is aligned with the IUS solid rocket motor thrust vector).
4. At the time of ejection, the tilt table will be elevated so that the IUS +X axis makes an angle of 58 degrees (measured in the Orbiter's plane of symmetry) with the Orbiter +X body axis.
5. The Orbiter attitude is to be inertially stabilized as nearly as possible with all RCS jets inactive at the time of payload ejection. Orbiter RCS jets are to remain inactive until the payload has cleared all Orbiter-fixed structure by at least TBD feet.
6. The line of action of the nominal spring-ejection impulse is assumed to pass through the payload center of gravity (CG) and through the Orbiter CG, producing no angular perturbation of either vehicle.
7. The payload must be continuously visible to the Orbiter flight crew after ejection until such time as positive separation has been confirmed during or immediately after the Orbiter's final separation maneuver (i.e., the OMS burn).
8. Plume impingement on the payload from Orbiter thrusters is to be minimized.

Table 3-1. Orbiter Mass Properties

PARAMETER	ASSUMED VALUE
Gross weight	200 017 lb
I_{xx}	887 302 slug-ft ²
I_{yy}	6 386 877 slug-ft ²
I_{zz}	6 694 367 slug-ft ²
I_{yz}	-971 slug-ft ²
I_{zx}	247 376 slug-ft ²
I_{xy}	5 622 slug-ft ²
CG Station (X_o)	1 095.3 in
CG Buttock line (Y_o)	0.3 in
CG Water line (Z_o)	377.4 in

3.2 MANEUVER SEQUENCE DESCRIPTIONS

As discussed previously, 8 discrete Orbiter/IUS initial separation phase maneuver sequences have been designed. In Sections 3.2.1 through 3.2.8, each sequence will be described by a table listing the schedule of events and by figures depicting the motion of the payload relative to the Orbiter. Three types of relative motion plots are shown to clarify each sequence: (1) overall view of the Orbiter/IUS initial separation phase including OMS plume contours, (2) an enlarged view of the early stage of the sequence, and (3) the IUS position relative to the RCS plume contours during RCS translational maneuvers. All of the data presented in this section were obtained by use of the HP-9825A High Fidelity Relative Motion Program (HFRMP), version 03M (References 1-3).

All rotational and translational maneuvers of the Orbiter were simulated with finite-thrust models. The Primary Reaction Control System (PRCS) was used for Orbiter attitude control. To minimize plume impingement on the payload, the +Z PRCS jets (i.e., those which expel propellant directly upward relative to the body-fixed frame) were inhibited. Attitude-control deadbands were set at 0.5 degrees per axis.

Figure 3-1 presents the nominal RCS propellant consumption and range of applicability for each of the eight standard Orbiter/IUS separation sequences. Figures 3-2 through 3-9 present an overall view of the Orbiter/IUS initial separation phase depicting the motion of the payload relative to the Orbiter for each sequence. In each sequence, a 35-second OMS burn is assumed to begin 9 minutes after ejection (9:00 P.E.T.). The plume contours for the OMS are overlaid on the relative motion plots. Examination of the figures will show that OMS plume impingement on the payload should be completely negligible even allowing for RCS maneuver execution errors that might produce a trajectory dispersion of 300 feet or so at the time of OMS ignition.

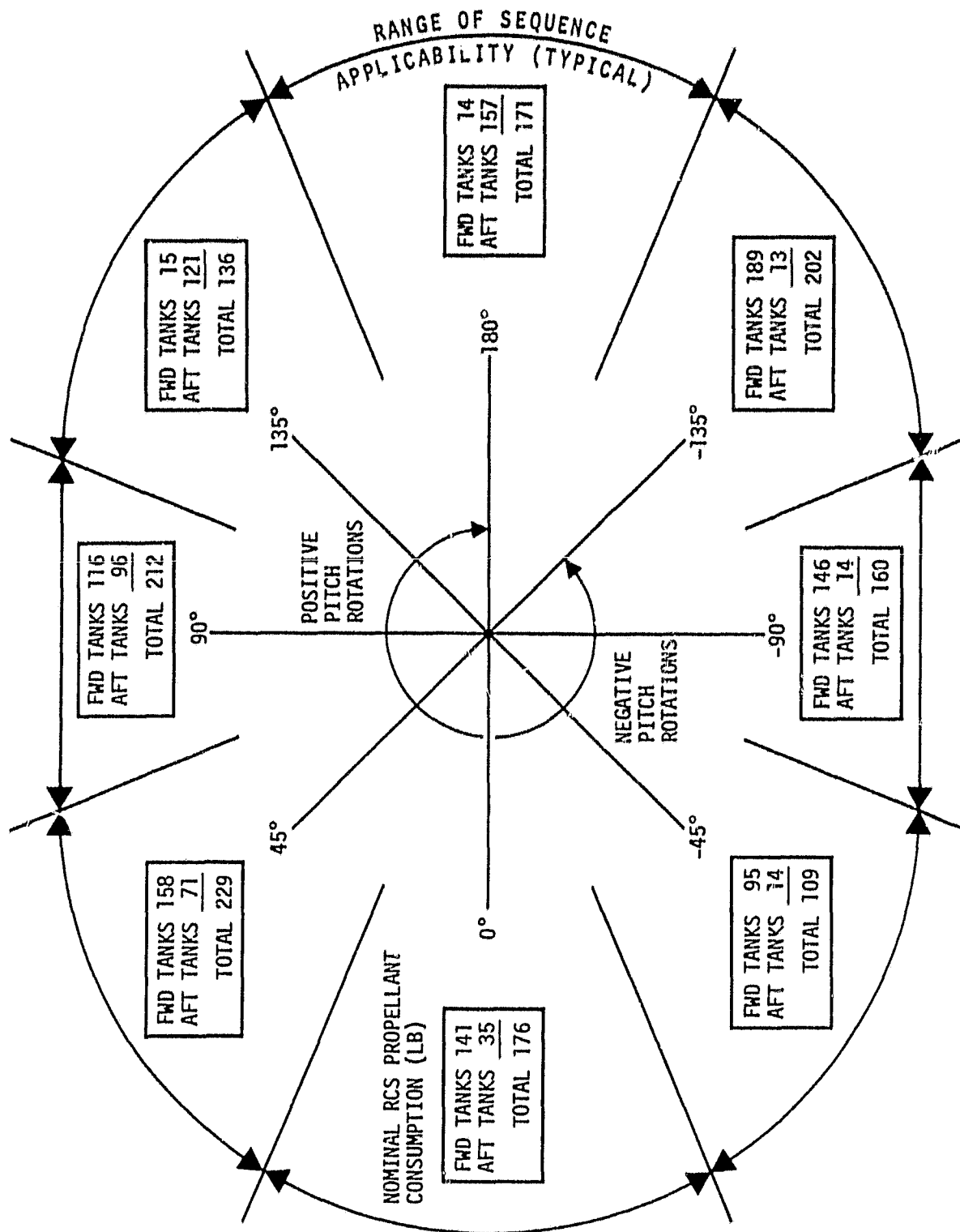
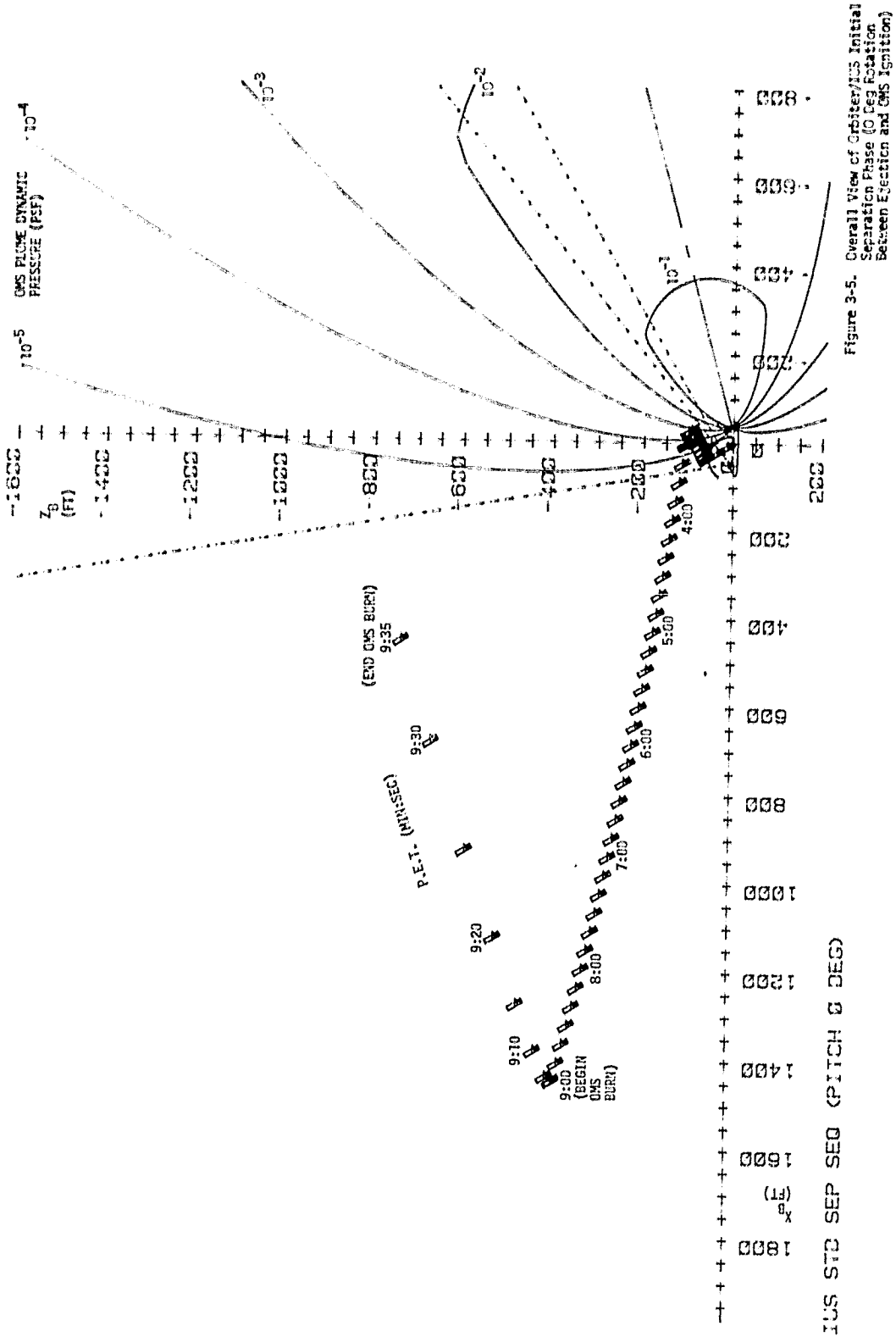
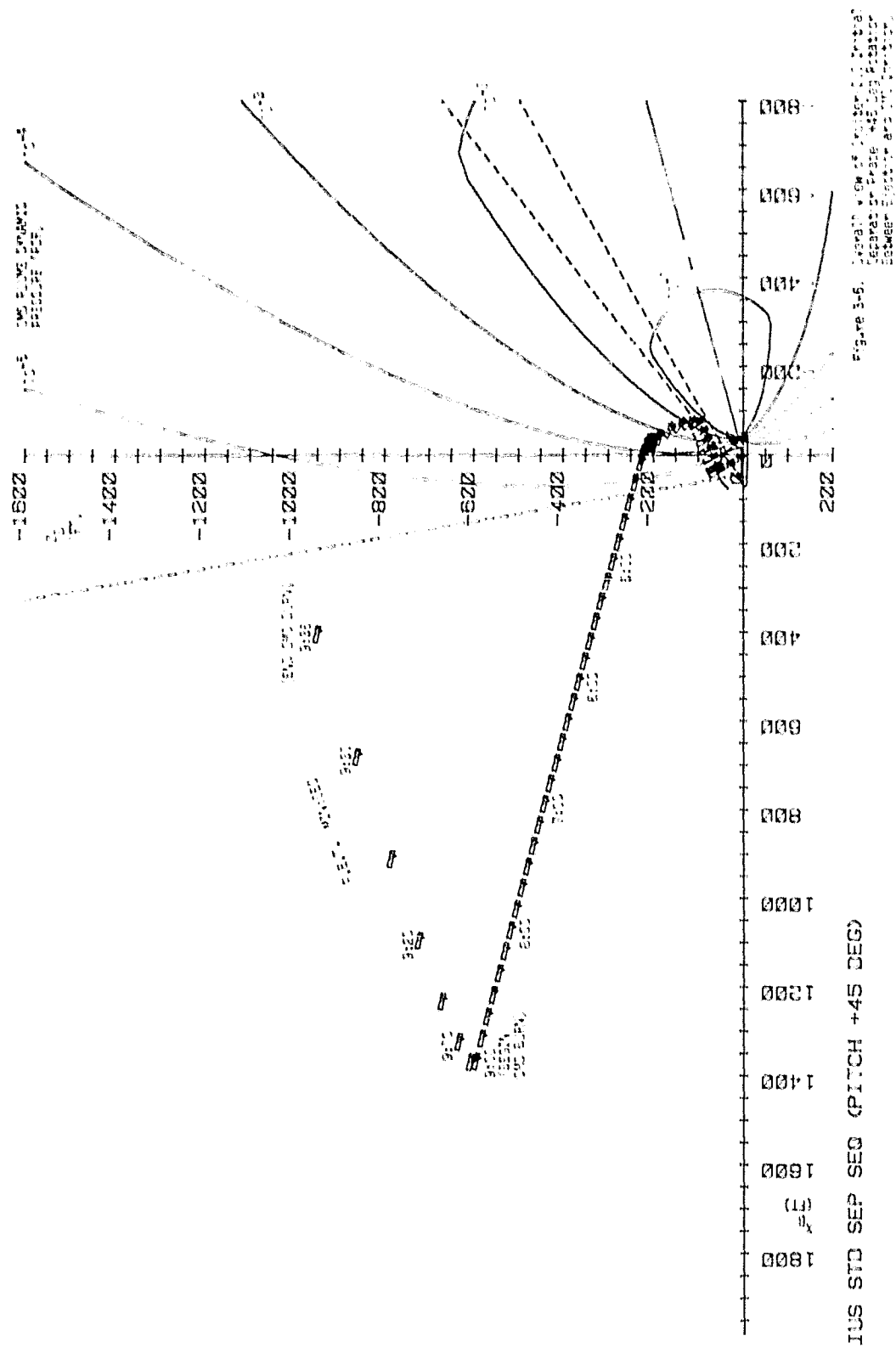
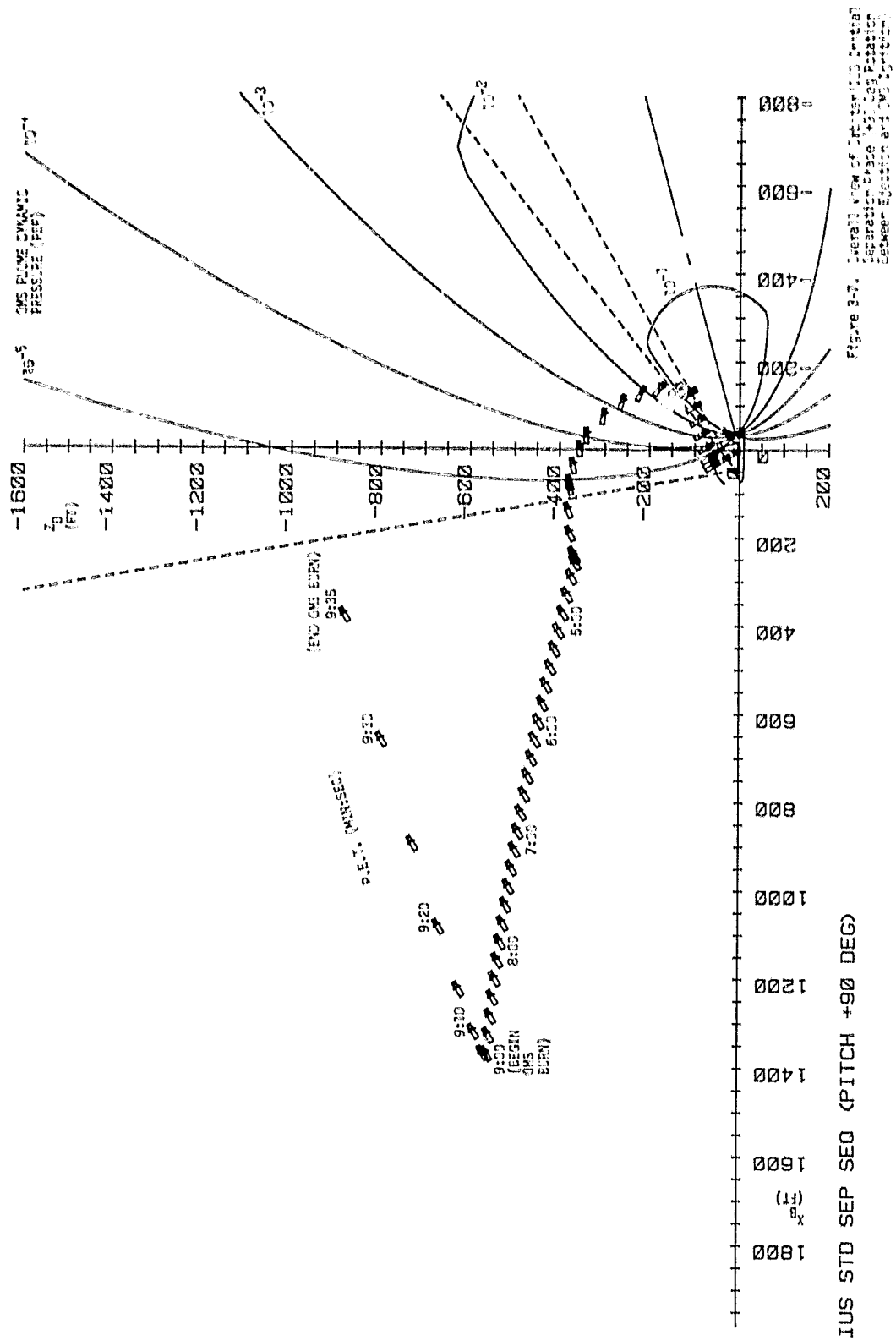
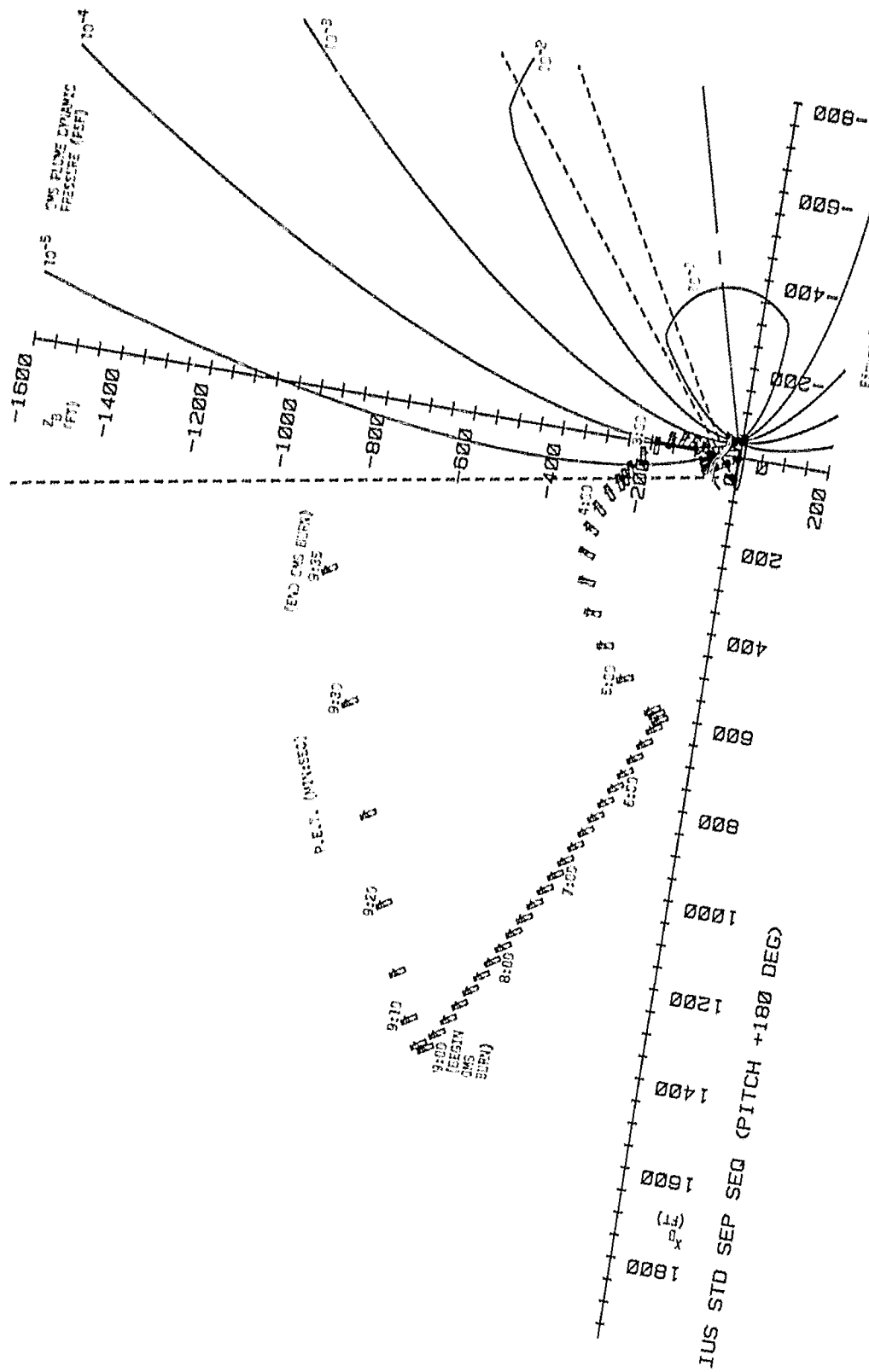


Figure 3-1. Nominal RCS Propellant Consumption for Eight Standard Orbiter/IUS Separation Sequences at Intervals of 45 Degrees in the Angle of Rotation Between Ejection and OMS Ignition









3.2.1 Orbiter Rotation Angle = -135 Degrees

Table 3-2 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/IUS initial separation sequence and Figure 3-10 presents an enlarged view of the early stage (to 6:50 P.E.T.). Examination of Figure 3-10 will show that the payload is visible through the pilot's overhead window almost continuously during the first 4 minutes and 20 seconds of the sequence.

The Orbiter's motion was uncontrolled from 0:00 P.E.T. until 1:00 P.E.T., at which time a 4-second -X RCS burn was initiated to provide a velocity component in the Orbiter +Z body-fixed direction without using the inhibited +Z RCS jets. Examination of Figure 3-10 will show that the payload is clear of the Orbiter structure at the time of the RCS burn. Immediately after the burn, a pitch rate of -1.0 deg/sec was established. At 2:30 P.E.T. a second -X RCS burn was performed for 5 seconds while maintaining the pitch rate. This burn was required to prevent the payload from going too far behind and too close to the Orbiter ($Z_B \rightarrow 0$ FT). The burn time was selected to minimize plume impingement. At approximately 3:19 P.E.T., the -135 degree angle of pitch from the nominal ejection attitude has been reached, an inertial hold initiated, and a 16-second -X RCS burn performed. This final translation places the Orbiter approximately 1400 feet behind and 650 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 3-2).

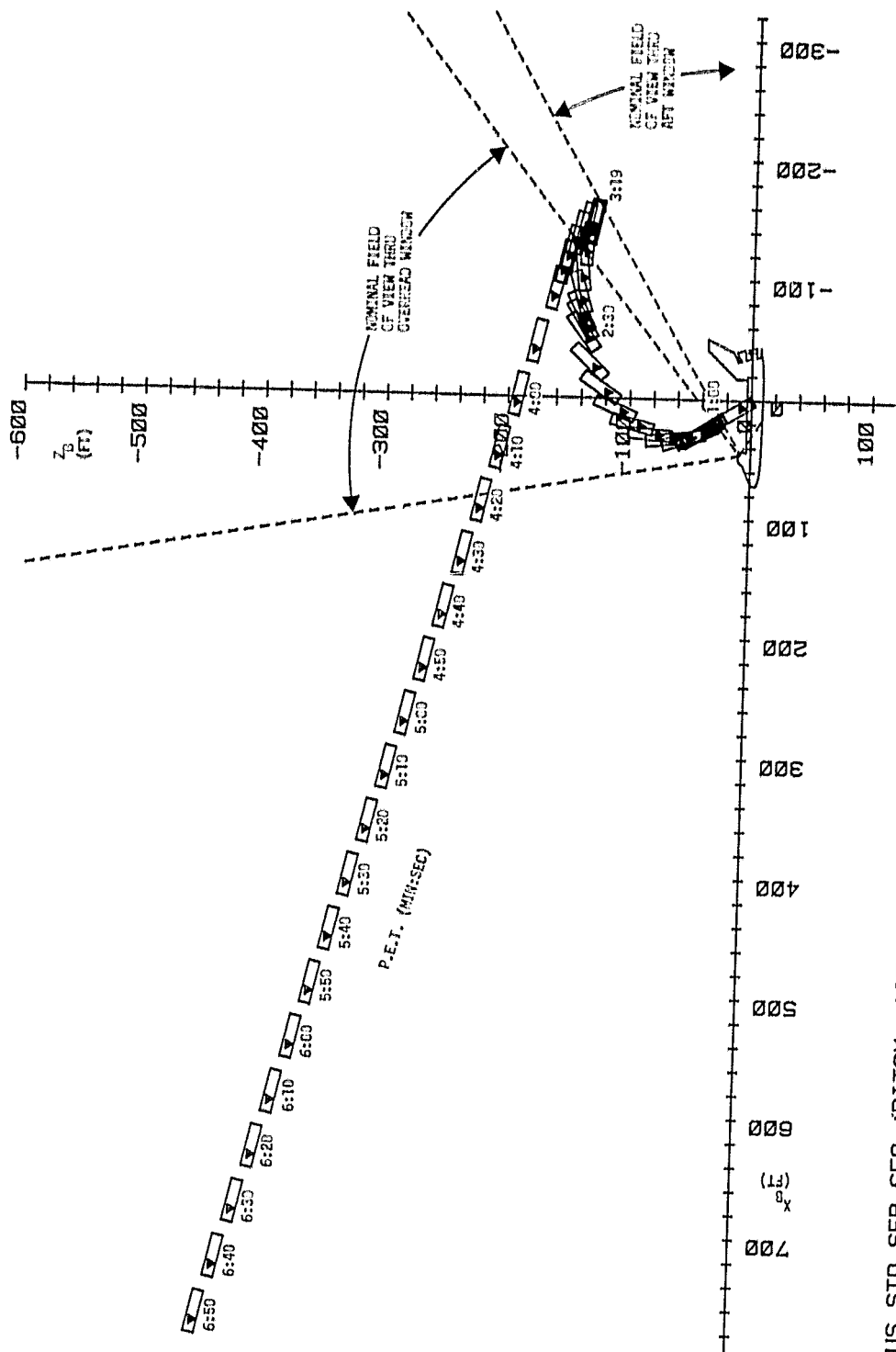
Figure 3-11 illustrates the IUS positions relative to the -X RCS plume during the 3 translational maneuvers. Because of the high degree of plume expansion, impingement cannot be avoided entirely. However, the RCS plume dynamic pressure experienced by the payload should be no greater than approximately 4×10^{-4} pounds per square foot during the 25-second total duration of the 3 RCS translational maneuvers because impingement during the second and third burns is negligible.

Table 3-2. Orbiter/IUS Initial Separation Sequence
(-135 Deg Rotation Between Ejection and
OMS Ignition)

1. Initiate 4-second -X RCS burn at 1:00 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at -1.0 deg/sec immediately after -X burn.
3. At 2:30 P.E.T., burn - X for 5 seconds while maintaining -1.0 deg/sec pitch rate.
4. Begin inertial hold upon reaching -135 deg angle of pitch from nominal ejection attitude (at approximately 3:19 P.E.T.).
5. Execute 16-second -X burn immediately after beginning inertial hold.
6. Trim to nominal OMS burn attitude, if necessary.
7. Begin 35-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	189
Aft Tanks	13
Total	202



IUS STD SEP SEQ (PITCH -135 DEG)

Figure 3-10. Enlarged View of Orbiter/US Initial Separation Phase (-135 Deg Rotation Between E-Section and OMS Igniter)

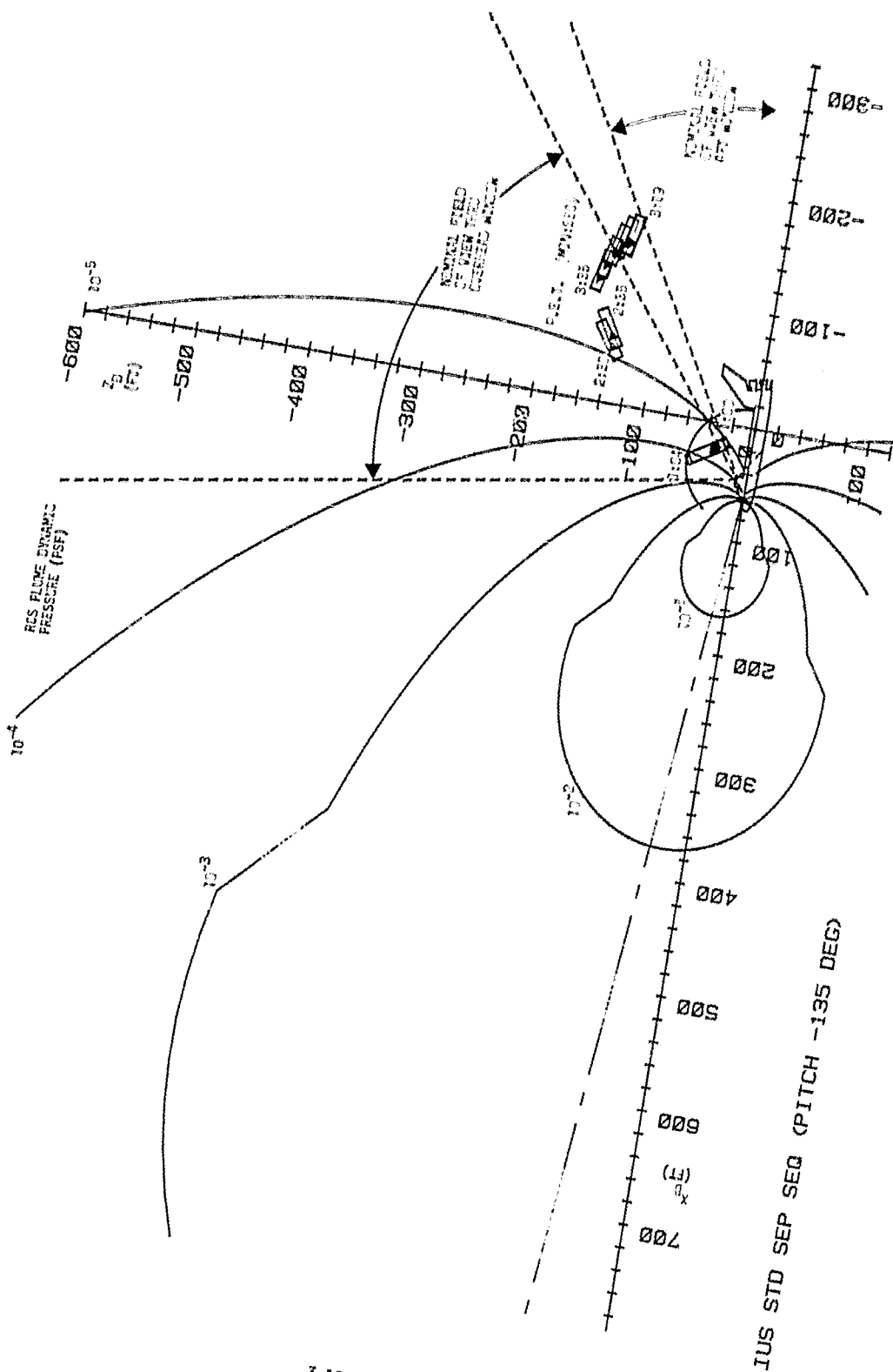


Figure 3-17. IUS STD SEP SEQ (PITCH -135 DEG) (1/10/60)

3.2.2 Orbiter Rotation Angle = -90 Degrees

Table 3-3 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/IUS initial separation sequence and Figure 3-12 presents an enlarged view of the early stage (to 6:10 P.E.T.). Examination of Figure 3-12 will show that the payload is visible through the pilot's overhead window almost continuously during the first 3 minutes and 10 seconds of the sequence.

The initiation of this sequence is similar to the -135 degree rotation case with a -1.0 deg/sec pitch rate established immediately upon completion of the 4-second -X RCS burn at 1:00 P.E.T. At approximately 2:34 P.E.T., the -90 degree angle of pitch rotation from the nominal ejection attitude has been reached, an inertial hold initiated, and a 15-second -X RCS burn performed. This burn is shorter than the third burn required in the -135 degree rotation case because the payload is not as far behind the Orbiter at the completion of the pitch maneuver. This final RCS translation places the Orbiter approximately 1500 feet behind and 450 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 3-3).

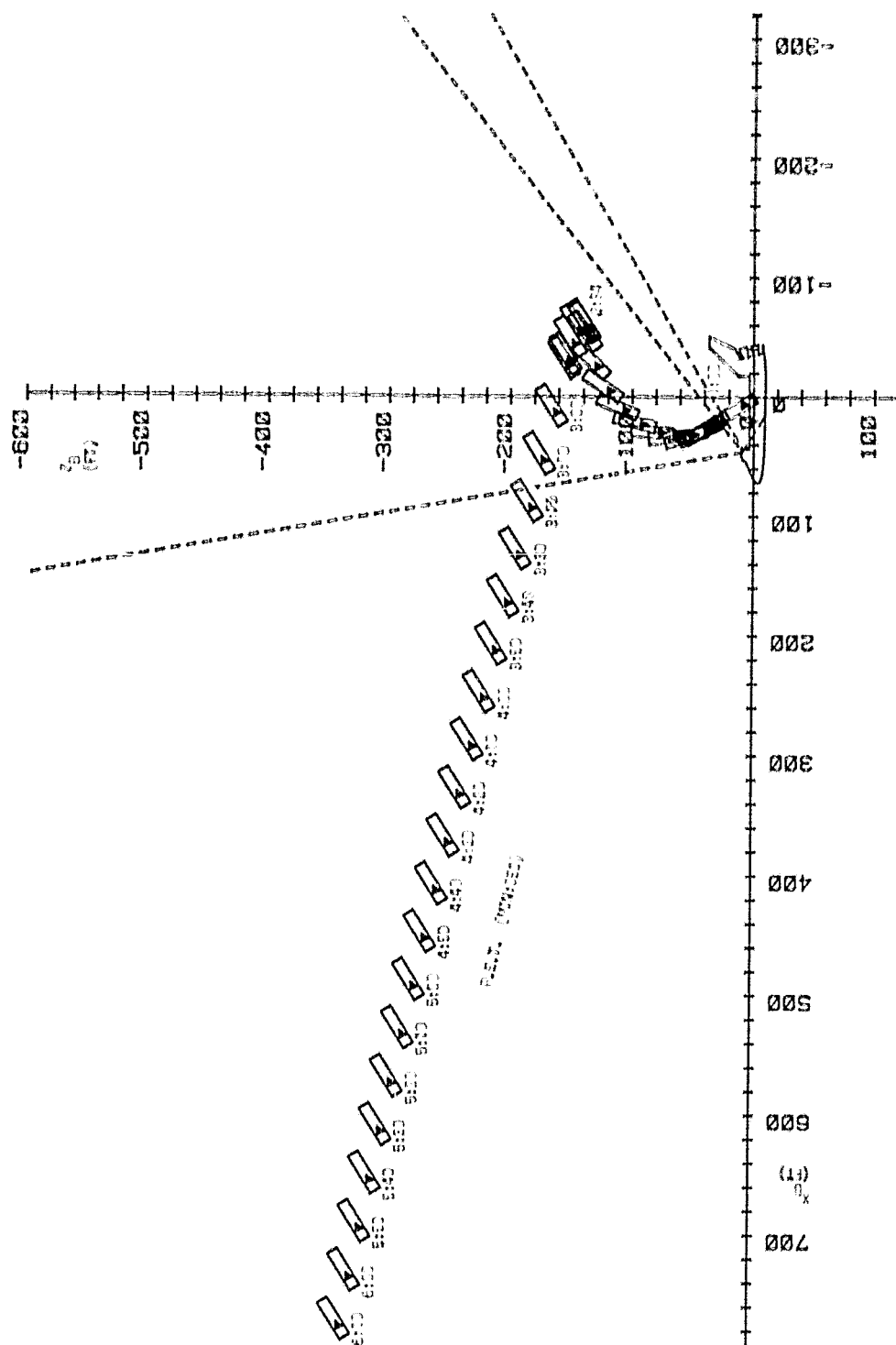
Figure 3-13 illustrates the IUS positions relative to the -X RCS plume during the 2 translational maneuvers. The payload should experience plume impingement of not greater than approximately 3×10^{-5} pounds per square foot during the second (2:34 P.E.T.) translational maneuver. The RCS plume dynamic pressure experienced by the payload should be no greater than approximately 4×10^{-4} pounds per square foot during the 19-second total duration of the 2 RCS translational maneuvers.

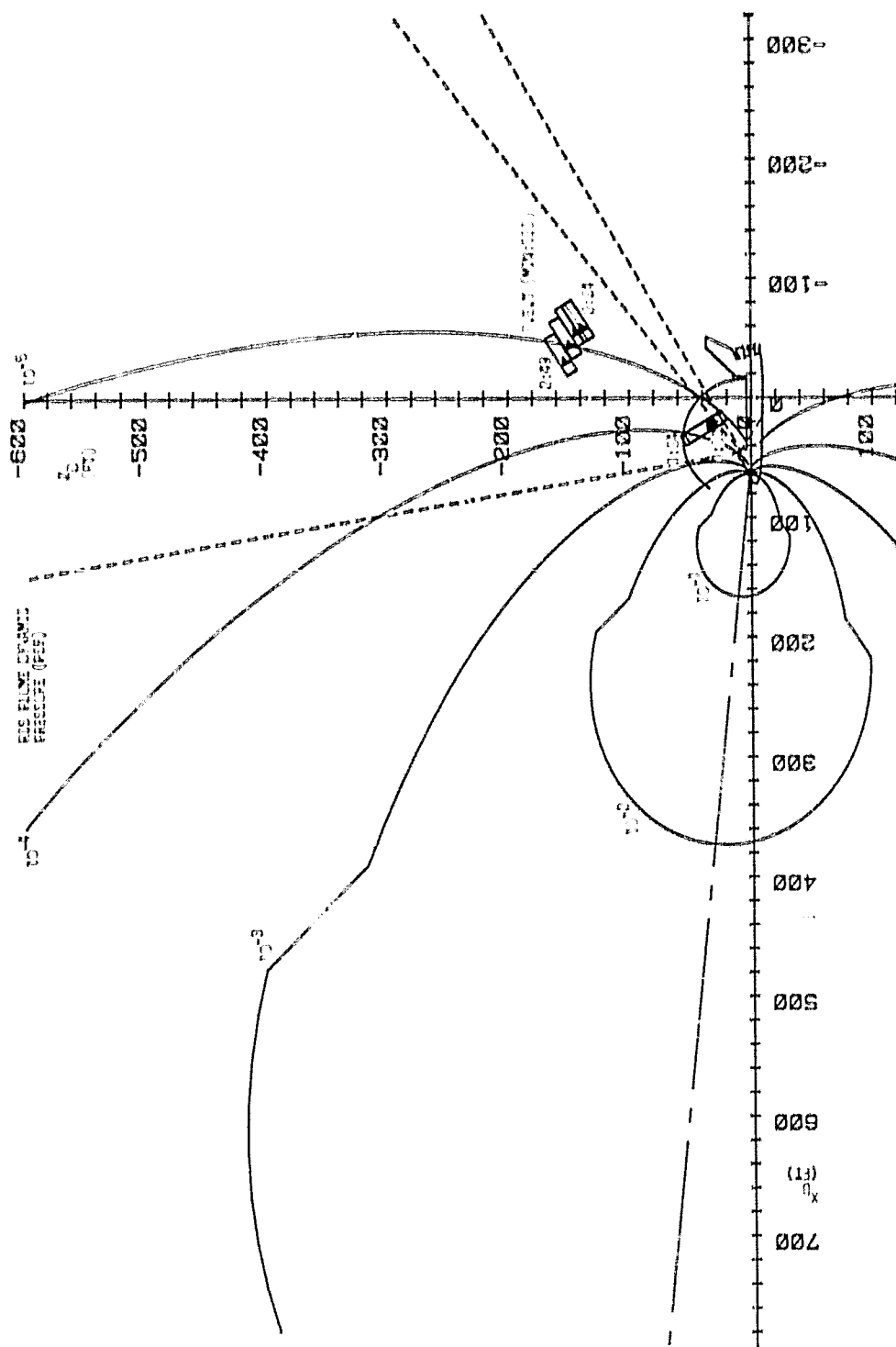
Table 3-3. Orbiter/IUS Initial Separation Sequence
(-90 Deg Rotation Between Ejection and
OMS Ignition)

1. Initiate 4-second -X RCS burn at 1:00 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at -1.0 deg/sec immediately after -X burn.
3. Begin inertial hold upon reaching -90 deg angle of pitch from nominal ejection attitude (at approximately 2:34 P.E.T.).
4. Execute 15-second -X burn immediately after beginning inertial hold.
5. Trim to nominal OMS burn attitude, if necessary.
6. Begin 35-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	146
Aft Tanks	<u>14</u>
Total	160





3.2.3 Orbiter Rotation Angle = -45 Degrees

Table 3-4 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/IUS initial separation sequence and Figure 3-14 presents an enlarged view of the early stage (to 6:10 P.E.T.). Examination of Figure 3-14 will show that the payload is visible through the pilot's overhead window almost continuously during the first 2 minutes of the sequence.

The initiation of this sequence is similar to the -135 and -90 degree rotation cases with a -1.0 deg/sec pitch rate established immediately upon completion of the 4-second -X RCS burn at 1:00 P.E.T. At approximately 1:49 P.E.T., the -45 degree angle of pitch rotation from the nominal ejection attitude has been reached, an inertial hold initiated, and an 8-second -X RCS burn performed. This burn is considerably shorter than the burns required in the -135 and -90 degree rotation cases because the payload is approximately directly overhead of the Orbiter at the completion of the pitch maneuver. This final RCS translation places the Orbiter approximately 1250 feet behind and 400 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 3-4).

Figure 3-15 illustrates the IUS positions relative to the -X RCS plume during the 2 translational maneuvers. Unlike the previous cases, plume impingement of the same magnitude (approximately 5×10^{-4} pounds per square foot) is also experienced during the second (1:49 P.E.T.) translational maneuver.

Table 3-4. Orbiter/IUS Initial Separation Sequence
(-45 Deg Rotation Between Ejection and
OMS Ignition)

1. Initiate 4-second -X RCS burn at 1:00 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at -1.0 deg/sec immediately after -X burn.
3. Begin inertial hold upon reaching -45 deg angle of pitch from nominal ejection attitude (at approximately 1:49 P.E.T.).
4. Execute 8-second -X burn immediately after beginning inertial hold.
5. Trim to nominal OMS burn attitude, if necessary.
6. Begin 35-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	95
Aft Tanks	<u>14</u>
Total	109

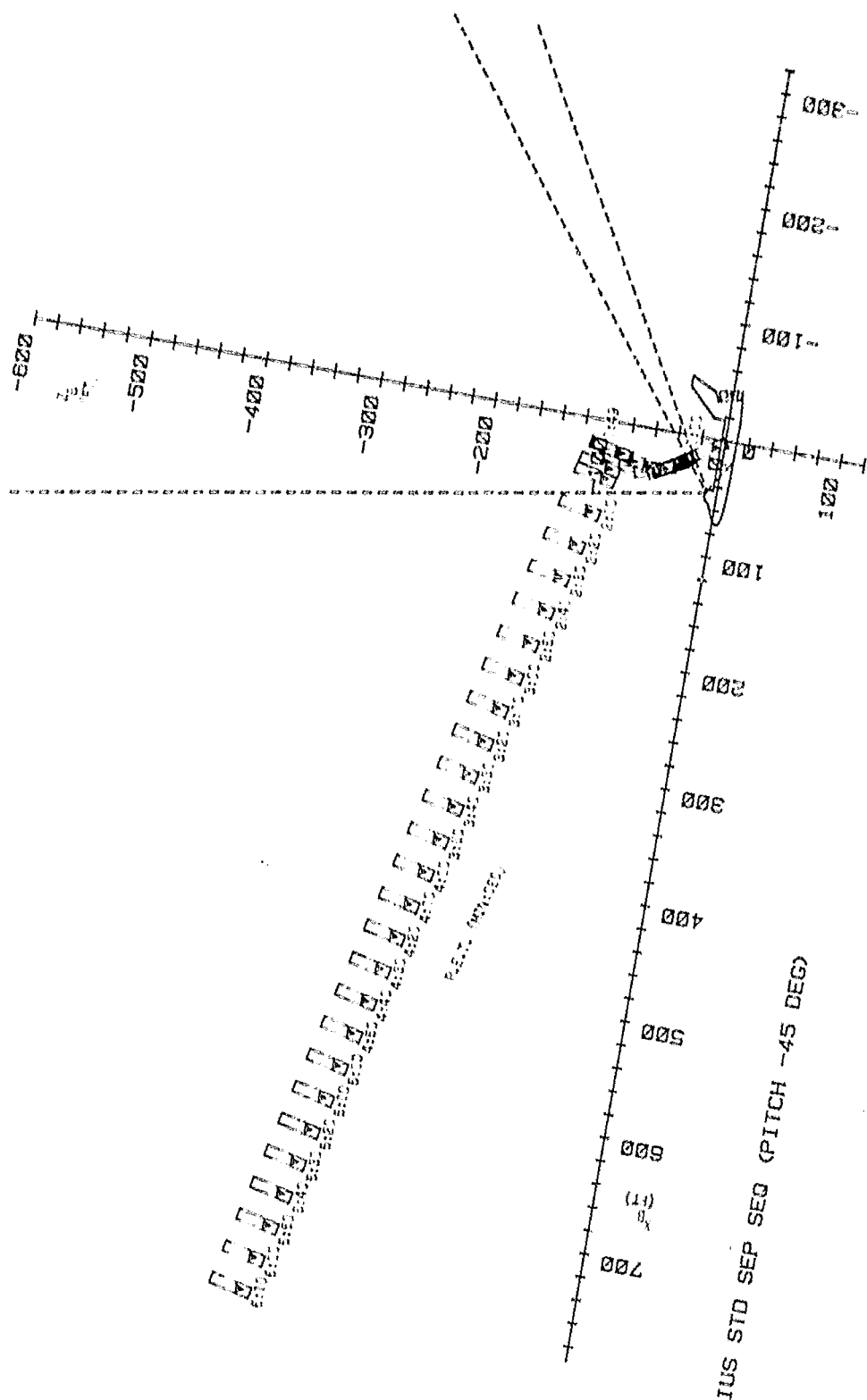
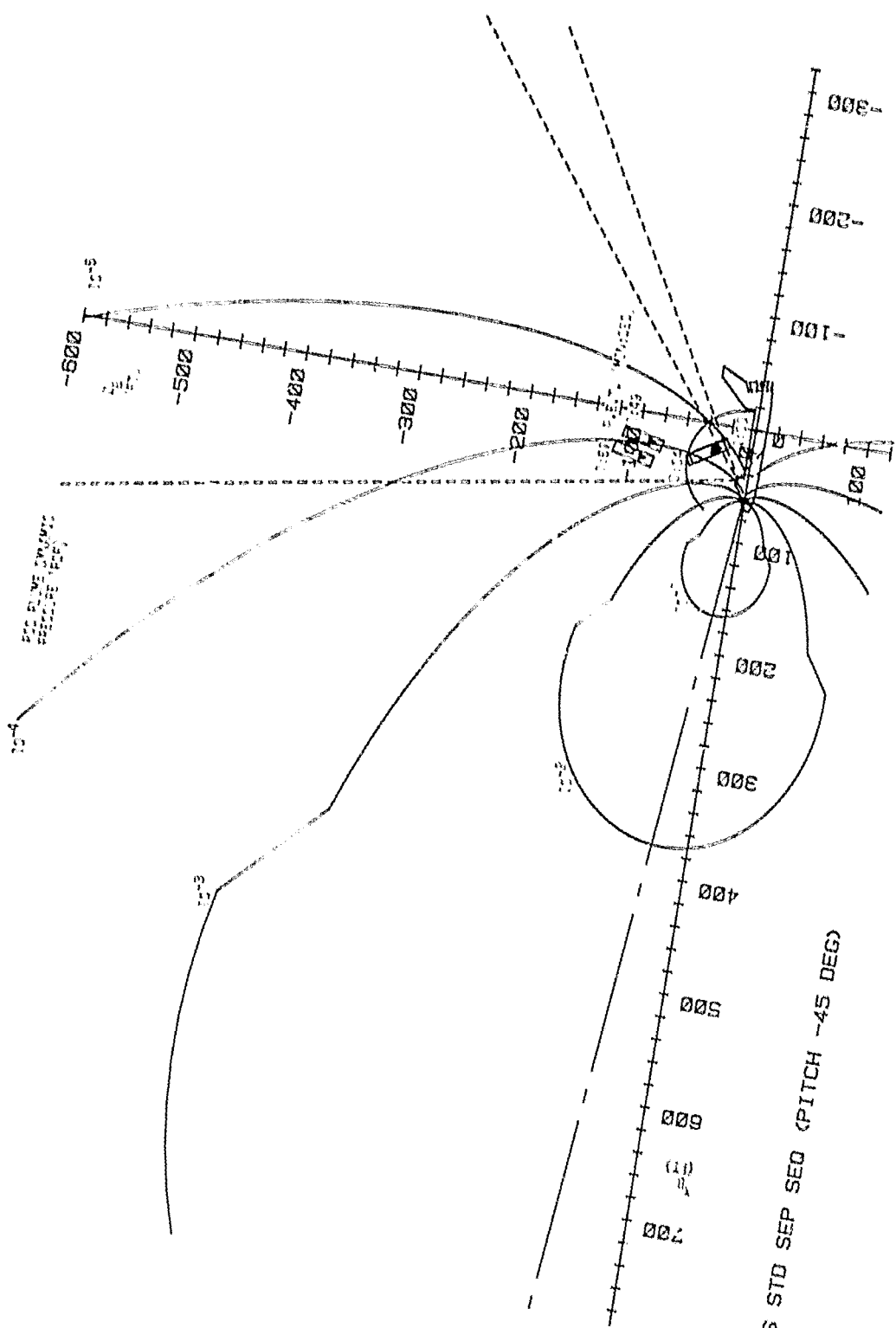


FIGURE 2-74
 Standard Separation Sequence
 (Pitch -45 DEG)



IUS STD SEP SEQ (PITCH -45 DEG)

3.2.4 Orbiter Rotation Angle = 0 Degrees

Table 3-5 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/IUS initial separation sequence and Figure 3-16 presents an enlarged view of the early stage (to 6:20 P.E.T.). Examination of Figure 3-16 will show that the payload is visible through the pilot's overhead window almost continuously during the first 3 minutes and 30 seconds of the sequence.

This, and the remaining sequences (positive Orbiter rotation angles), are initiated differently than the previous cases (negative Orbiter rotation angles). The Orbiter's motion was uncontrolled from 0:00 P.E.T. until 1:00 P.E.T. at which time an inertial hold was initiated and held until 2:00 P.E.T. At this time the first RCS translational maneuver was performed. Unlike the previous cases, this burn is performed using the +X RCS jets (5 seconds) which cause the payload to move aft and slightly upward relative to the Orbiter.

At first this would appear to be counter productive because the aft motion must be stopped by a -X RCS burn of sufficient length to place the payload at least 1200 feet ahead of the Orbiter at the time of OMS ignition (9:00 P.E.T.). However, the vertical displacement at ignition time must also be considered because the Orbiter positive pitch rotation (+0 degrees in this case) will tend to decrease the vertical distance between the Orbiter and payload. Since the +Z RCS jets have been inhibited, the vertical displacement must be accomplished using the +X RCS jets both of which produce a small +Z velocity component. By alternating the +X RCS burns, sufficient vertical displacement can be reached while maintaining the proper horizontal displacement and minimizing the plume impingement.

The second translational maneuver is a 19-second -X RCS burn at 3:00 P.E.T. which places the Orbiter approximately 1400 feet behind and 400 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 3-5).

Figures 3-17 and 3-18 illustrate the IUS positions relative to the +X RCS and -X RCS plume, respectively, during the 2 translational maneuvers. The maximum plume dynamic pressure experienced by the payload should be no greater than approximately 7×10^{-5} pounds per square foot during the 24-second total duration of the 2 RCS translational maneuvers.

Table 3-5. Orbiter/IUS Initial Separation Sequence
(0 Deg Rotation Between Ejection and OMS
Ignition)

1. Begin inertial hold at 1:00 P.E.T.
2. Initiate 5-second +X RCS burn at 2:00 P.E.T.
3. Initiate 19-second -X burn at 3:00 P.E.T.
4. Trim to nominal OMS burn attitude, if necessary.
5. Begin 35-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	141
Aft Tanks	<u>35</u>
Total	176

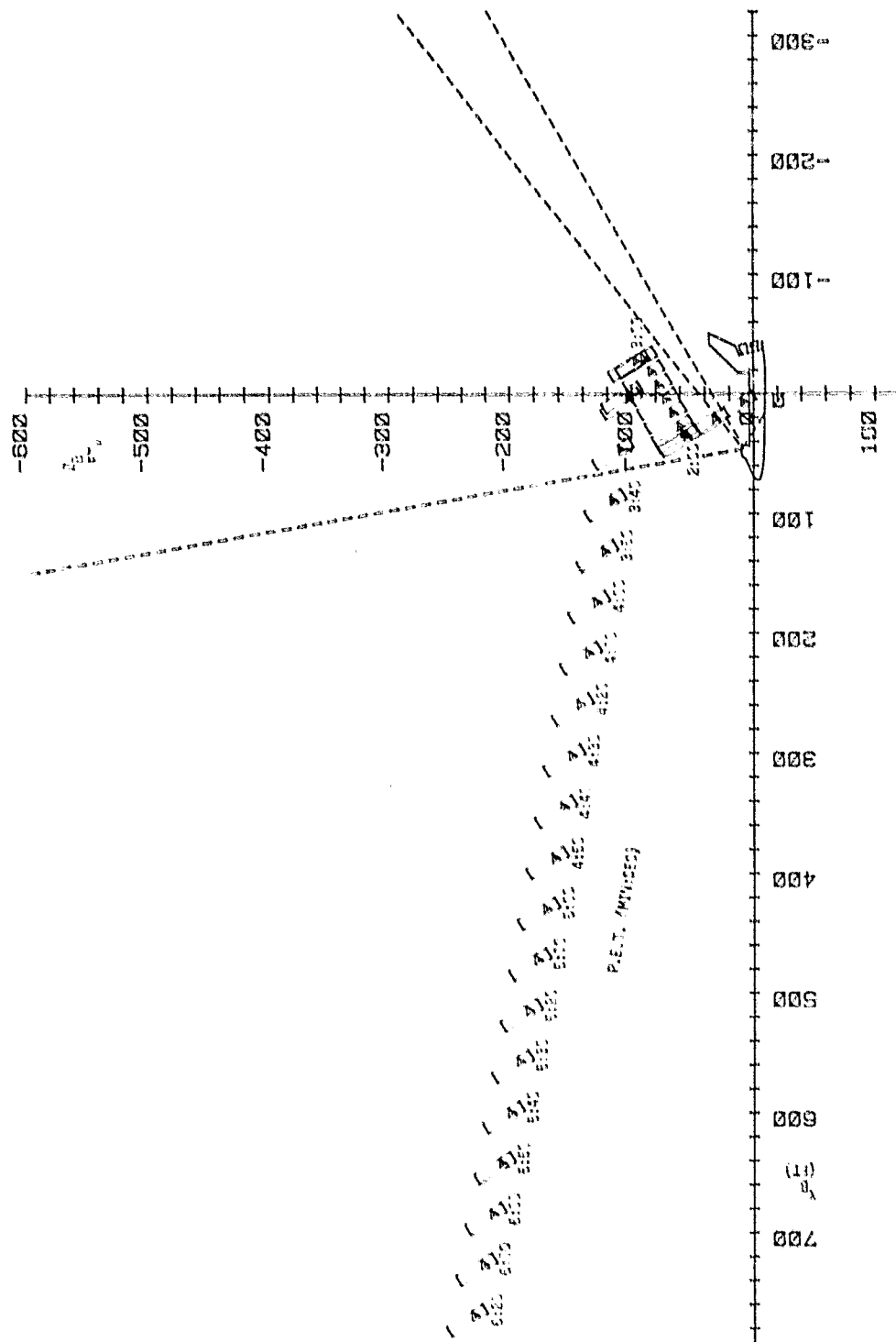


Figure 2-15. Bridge and Gun Positioning Data
Separating Data Using Position
Between Bridge and Gun Position

IUS STD SEP SEQ (PITCH 0 DEG)

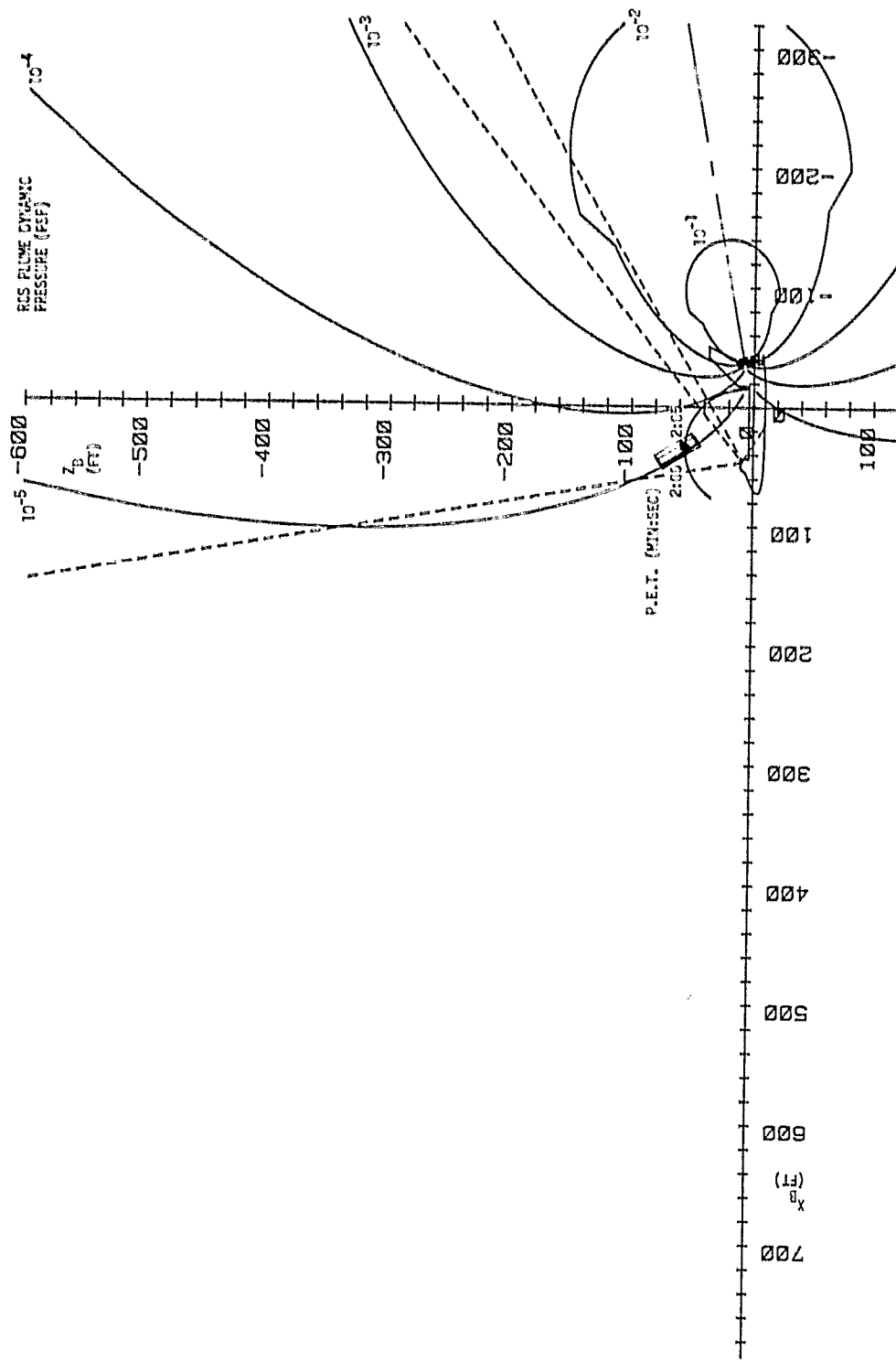


Figure 3-17. IUS Positions Relative to +X RCS Plane
 (0 deg Rotation Between Ejection and
 CWS Ignition)

IUS STD SEP SEQ (PITCH +0 DEG)

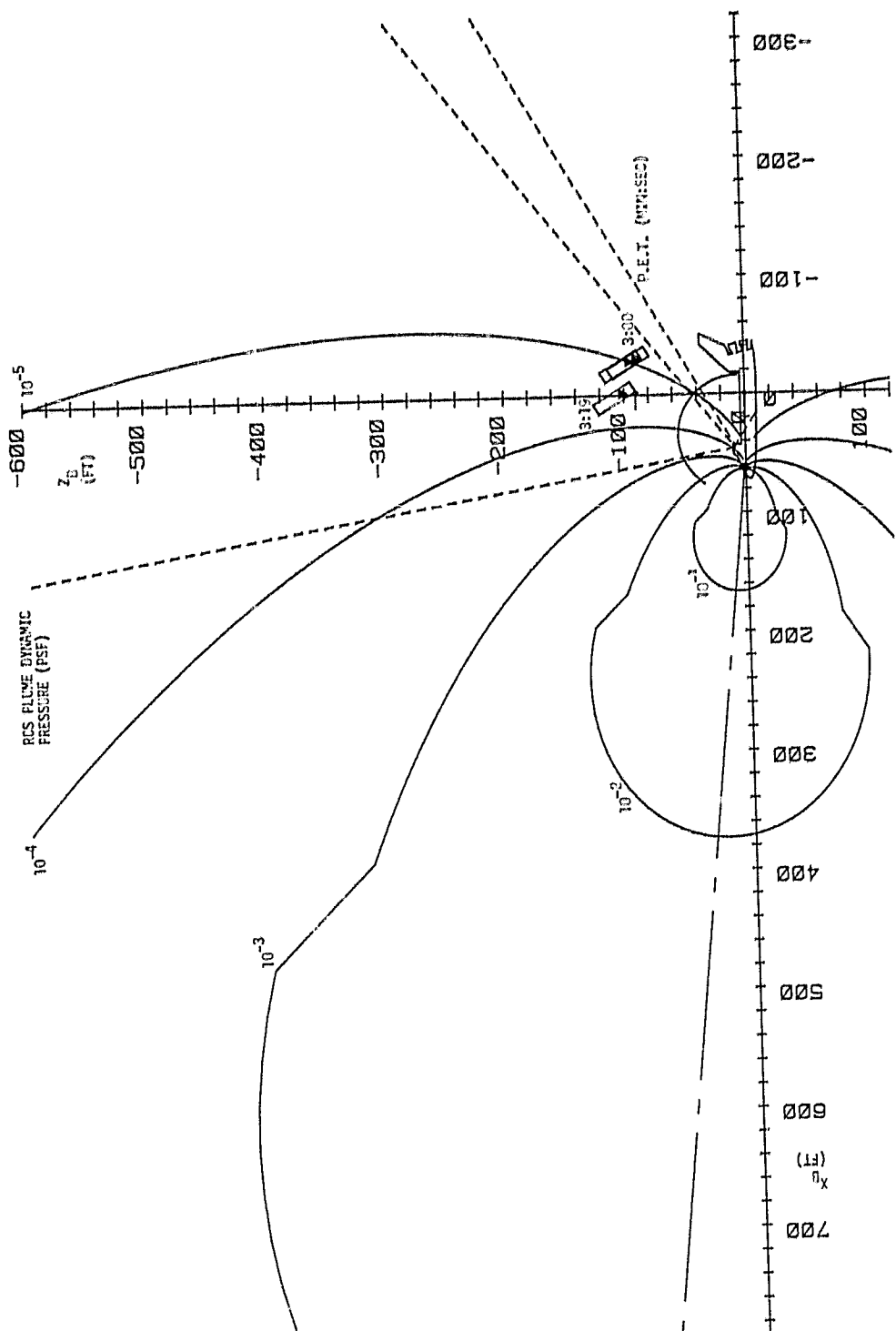


Figure 3-18. IUS Positions Relative to X-RIS Plume
(0 Deg Rotation Between Ejection and
OMS Ignition)

IUS STD SEP SEQ (PITCH +0 DEG)

3.2.5 Orbiter Rotation Angle = +45 Degrees

Table 3-6 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/IUS initial separation sequence and Figure 3-19 presents an enlarged view of the early stage (to 7:00 P.E.T.). Examination of Figure 3-19 will show that the payload is visible through the pilot's overhead window almost continuously during the first 4 minutes and 20 seconds of the sequence.

The initiation of this sequence is similar to the 0 degree rotation case but with an 8-second +X RCS burn at 2:00 P.E.T. followed at 3:00 P.E.T. by initiating a +1.0 deg/sec pitch rate. At approximately 3:47 P.E.T., the +45 degree angle of pitch rotation from the nominal ejection attitude has been reached and a 20-second -X RCS burn performed. This final RCS translation places the Orbiter approximately 1350 feet behind and 600 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 3-6).

Figures 3-20 and 3-21 illustrate the IUS positions relative to the +X RCS and -X RCS plume, respectively, during the 2 translational maneuvers. The maximum plume dynamic pressure experienced by the payload should be no greater than approximately 7×10^{-5} pounds per square foot during the 28-second total duration of the 2 RCS translational maneuvers.

Table 3-6. Orbiter/IUS Initial Separation Sequence
(+45 Deg Rotation Between Ejection and
OMS Ignition)

1. Begin inertial hold at 1:00 P.E.T.
2. Initiate 8-second +X RCS burn at 2:00 P.E.T.
3. Start +1.0 deg/sec pitch maneuver at 3:00 P.E.T.
4. Resume inertial hold upon reaching +45 deg angle of pitch from nominal ejection attitude (at approximately 3:47 P.E.T.).
5. Execute 20-second -X burn immediately after resuming inertial hold.
6. Trim to nominal OMS burn attitude, if necessary.
7. Begin 35-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	158
Aft Tanks	71
Total	229

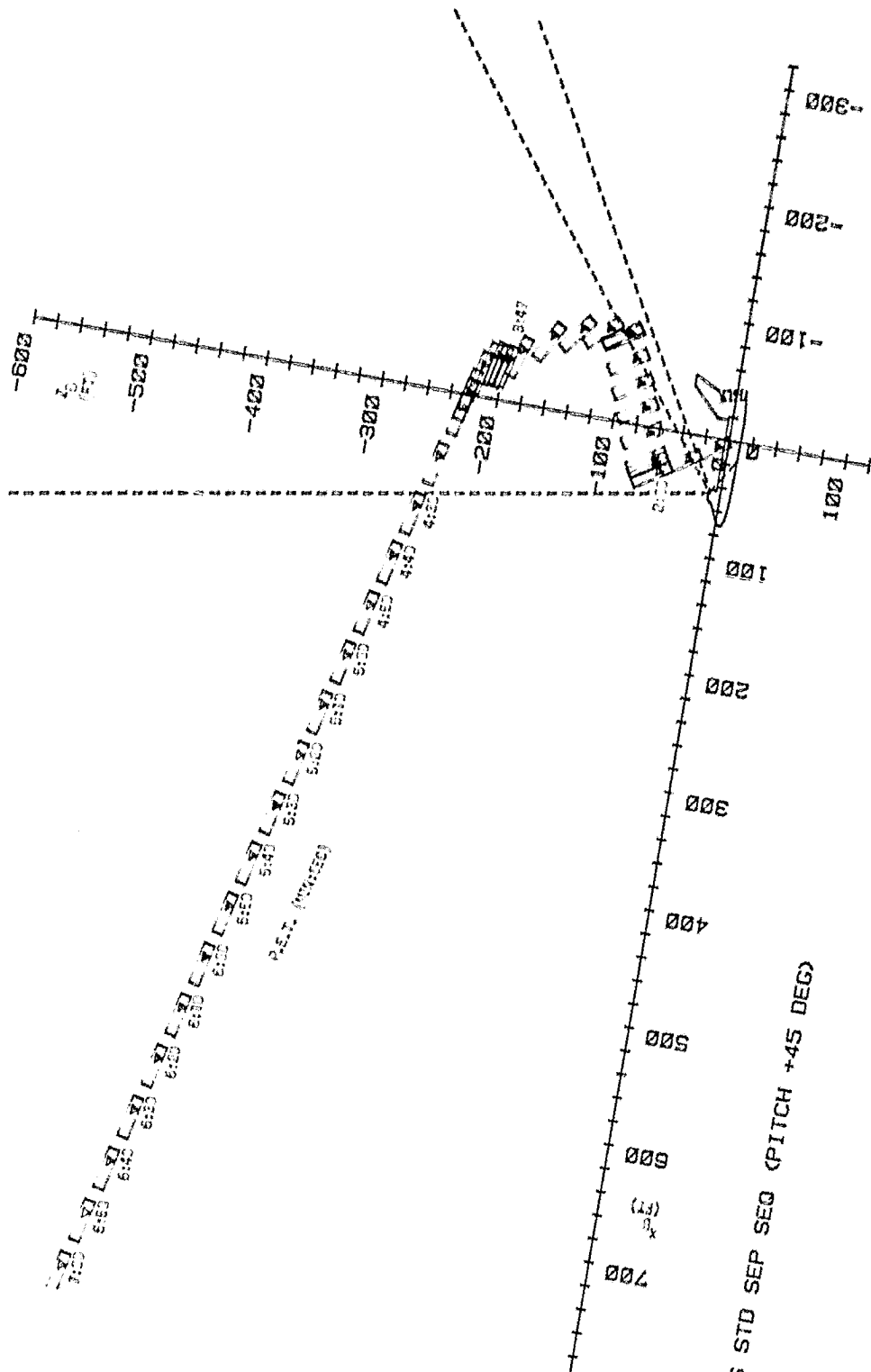
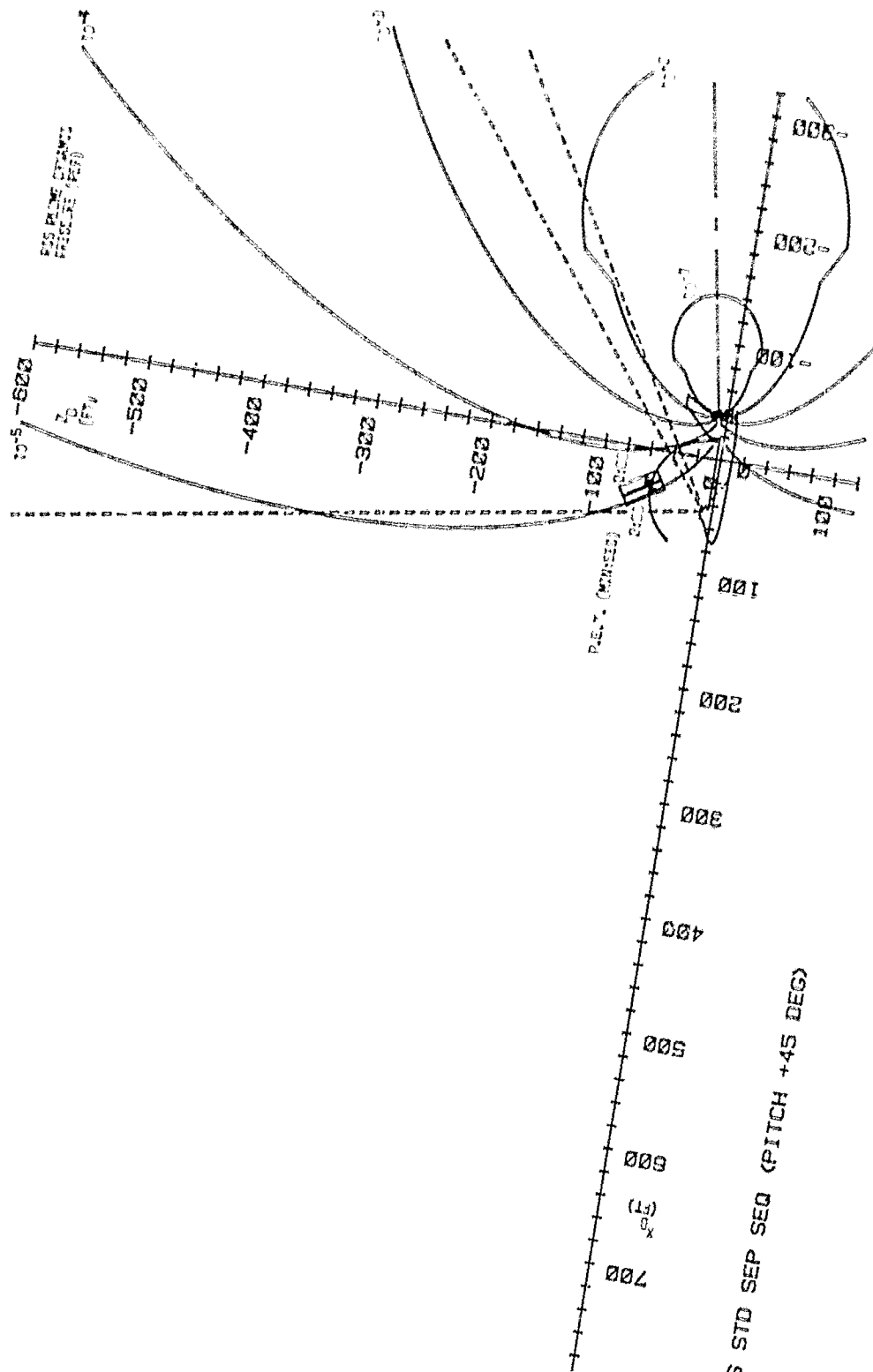


Figure 2-79. Estimated view of the "MUSSEY" data points.



IUS STD SEP SEQ (PITCH +45 DEG)

FIGURE 3-40

10000
9000
8000
7000
6000
5000
4000
3000
2000
1000
0
-1000
-2000
-3000
-4000
-5000
-6000
-7000
-8000
-9000
-10000

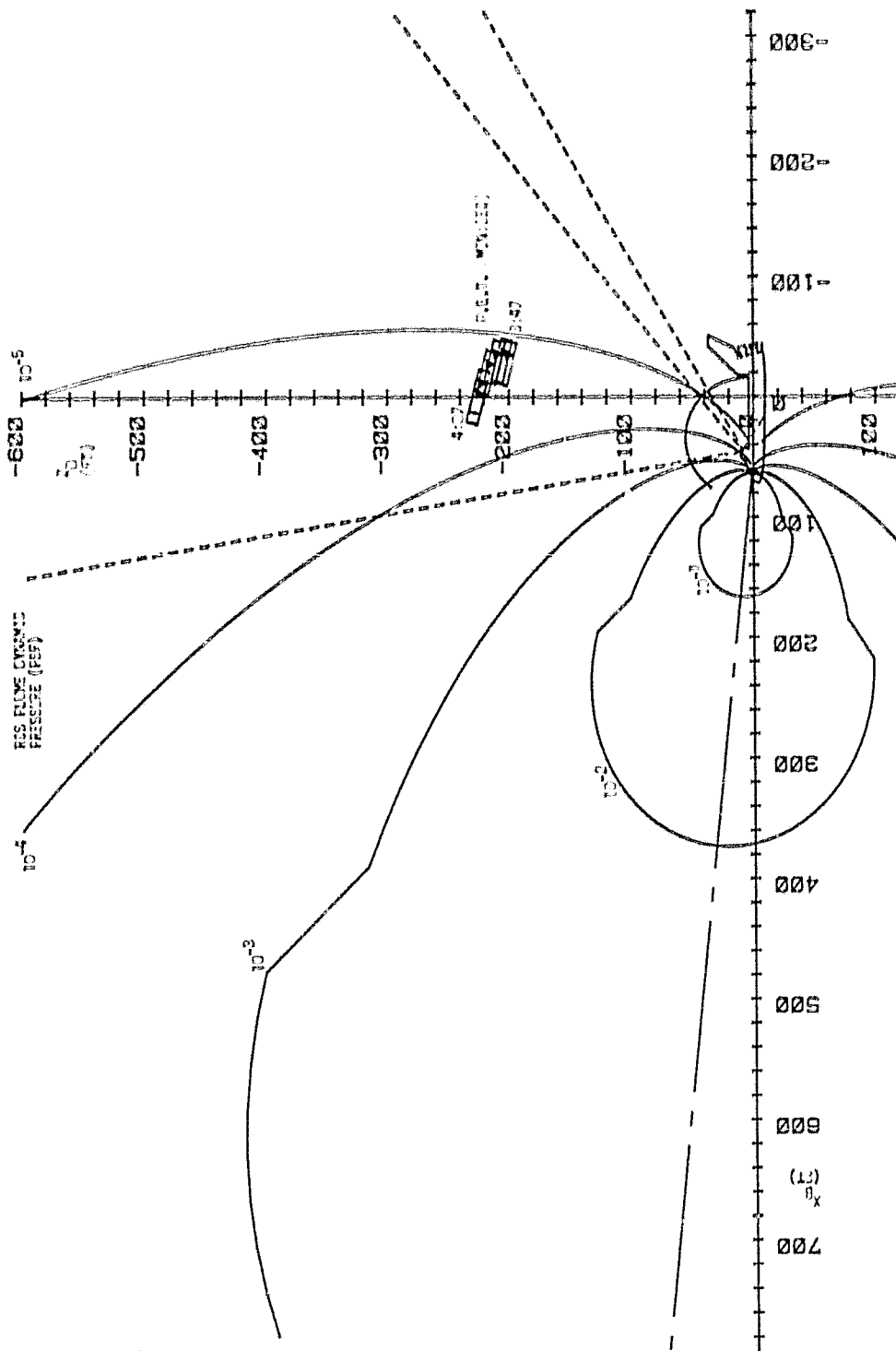


Figure 3-20. RDS Flow Command Pressure vs X_0 (in) for IUS STD SEP SEQ (PITCH +45 DEG) at 1000 and 2000 PSF.

IUS STD SEP SEQ (PITCH +45 DEG)

3.2.6 Orbiter Rotation Angle = +90 Degrees

Table 3-7 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/IUS initial separation sequence and Figure 3-22 presents an enlarged view of the early stage (to 6:40 P.E.T.). Examination of Figure 3-22 will show that the payload is visible through the pilot's overhead window almost continuously during the first 4 minutes and 15 seconds of the sequence. For approximately 40 seconds (from P.E.T. = 2:40 to P.E.T. = 3:20) the payload is outside of the pilot's nominal field of view. However, the payload can probably be seen directly through the window if the pilot moves his head to an appropriate position. Also, the payload should be visible to the CCTV camera mounted on the forward bulkhead.

The initiation of this sequence is similar to the 0 and +45 degree rotation cases but with a 12-second +X RCS burn (see Figure 3-23) at 2:00 P.E.T., followed at 3:00 P.E.T. by initiating a +1.0 deg/sec pitch rate. At 4:00 P.E.T. a 14-second -X RCS burn is performed while maintaining the pitch rate. The burn occurs at this time because less plume impingement would be experienced by the payload than if the burn were performed at the end of the pitch maneuver at 4:32 P.E.T. (see Figure 3-24). This final RCS translation places the Orbiter approximately 1400 feet behind and 550 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 3-7).

Figures 3-23 and 3-24 illustrate the IUS position relative to the +X RCS and -X RCS plume, respectively, during the 2 translational maneuvers. The maximum plume dynamic pressure experienced by the payload should be no greater than approximately 9×10^{-5} pounds per square foot during the 26-second total duration of the 2 RCS translational maneuvers.

Table 3-7. Orbiter/IUS Initial Separation Sequence
(+90 Deg Rotation Between Ejection and
OMS Ignition)

1. Begin inertial hold at 1:00 P.E.T.
2. Initiate 12-second +X RCS burn at 2:00 P.E.T.
3. Start +1.0 deg/sec pitch maneuver at 3:00 P.E.T.
4. At 4:00 P.E.T., burn -X for 14 seconds while maintaining +1.0 deg/sec pitch rate.
5. Resume inertial hold upon reaching +90 deg angle of pitch from nominal ejection attitude (at approximately 4:32 P.E.T.).
6. Trim to nominal OMS burn attitude, if necessary.
7. Begin 35-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	116
Aft Tanks	96
Total	212

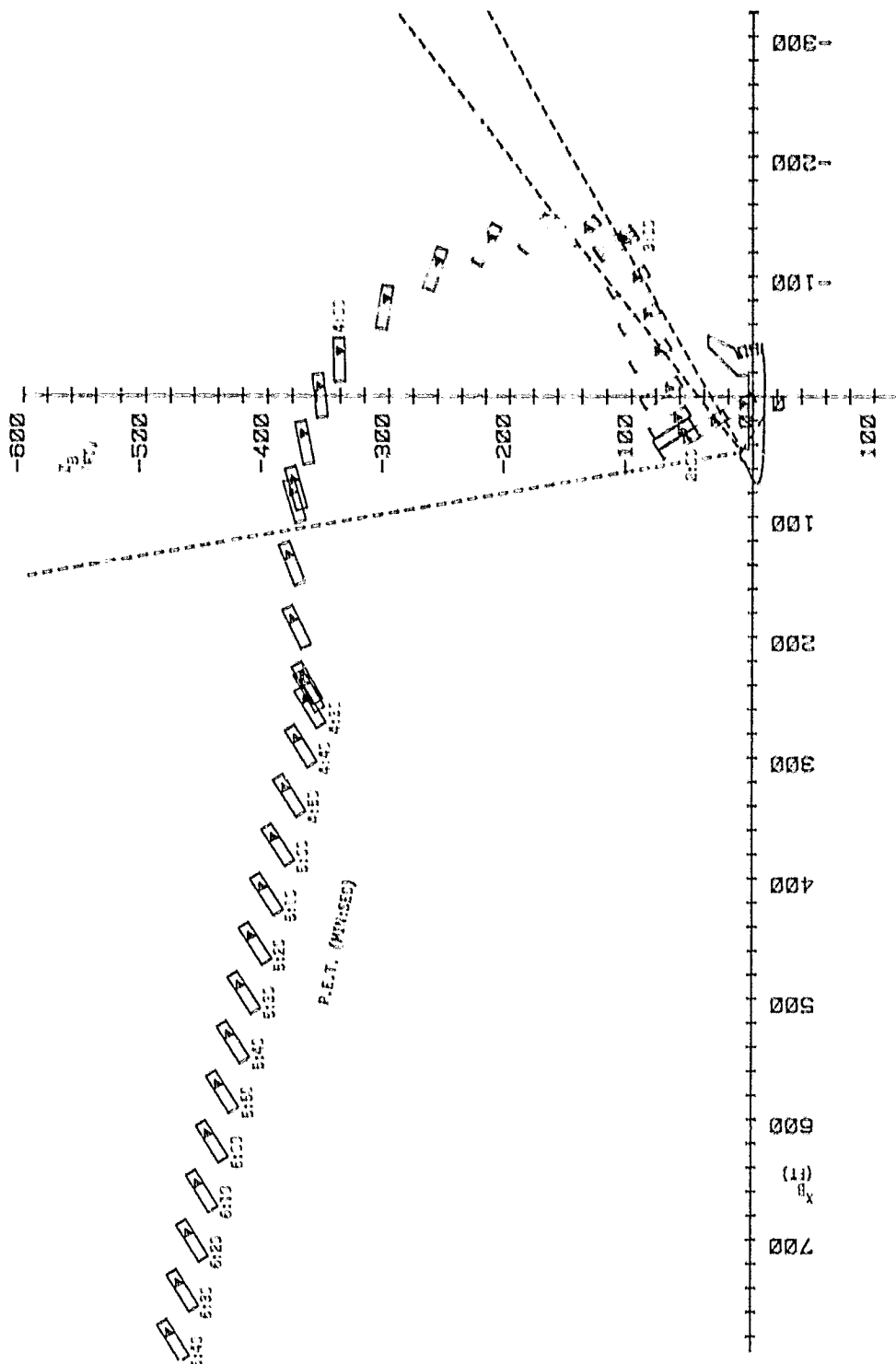


Figure 5-22. Ranges and P.E.T. for IUS STD SEP SEC (PITCH +90 DEG)

IUS STD SEP SEC (PITCH +90 DEG)

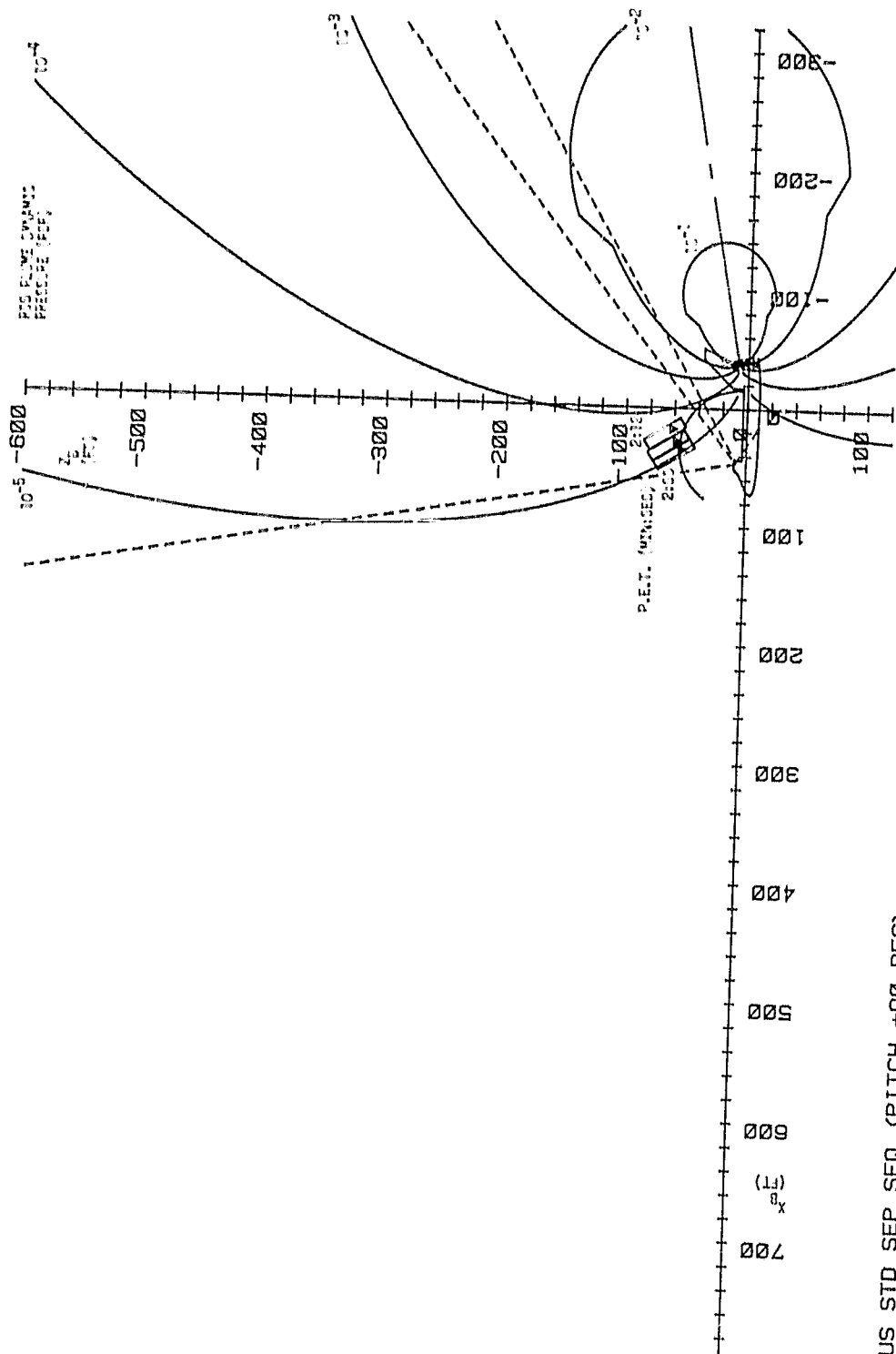


Figure 3-23. P.T. Error Relative to Y P.T. Error
+90 deg Rotation Between Electron and
D.C. Igniter.

IUS STD SEP SEQ (PITCH +90 DEG)

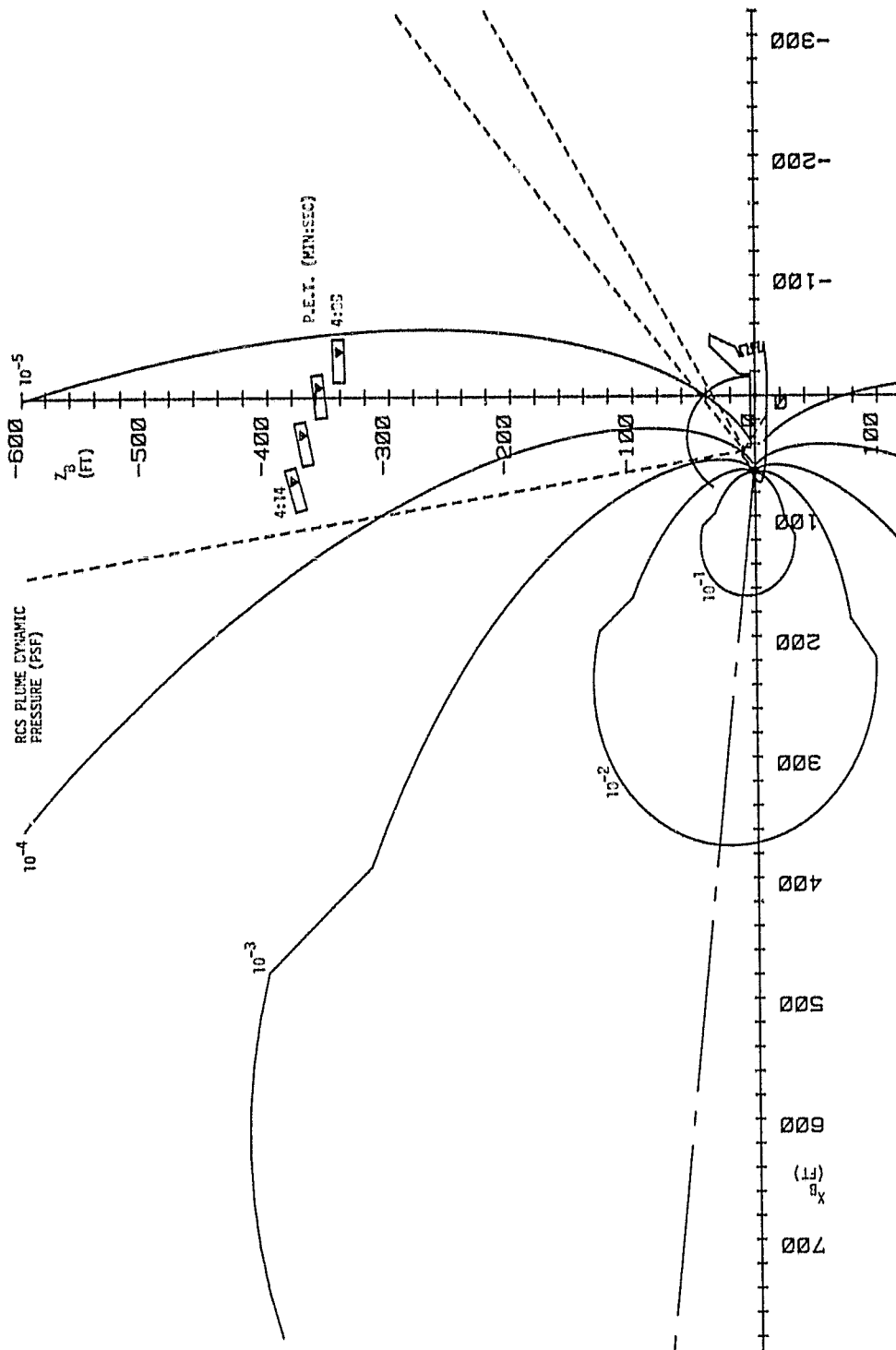


Figure 3-24. IUS Positions Relative to -X RCS Plume (+90 Deg Rotation Between Ejection and OMS Ignition)

IUS STD SEP SEQ (PITCH +90 DEG)

3.2.7 Orbiter Rotation Angle = +135 Degrees

Table 3-8 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/IUS initial separation sequence and Figure 3-25 presents an enlarged view of the early stage (to 6:00 P.E.T.). Examination of Figure 3-25 will show that the payload is visible through the pilot's overhead window almost continuously during the first 3 minutes and 40 seconds of the sequence.

The initiation of this sequence is similar to the other positive rotation cases but with a single 16-second +X RCS burn at 2:00 P.E.T. This burn is sufficient to place the Orbiter approximately 1350 feet behind and 225 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 3-8).

Figure 3-26 illustrates the IUS positions relative to the +X RCS plume during the translational maneuver. The maximum plume dynamic pressure experienced by the payload should be no greater than approximately 2×10^{-4} pounds per square foot.

Table 3-8. Orbiter/IUS Initial Separation Sequence
(+135 Deg Rotation Between Ejection and
OMS Ignition)

1. Begin inertial hold at 1:00 P.E.T.
2. Initiate 16-second +X RCS burn at 2:00 P.E.T.
3. Start +1.0 deg/sec pitch maneuver immediately after completion of +X burn.
4. Resume inertial hold upon reaching +135 deg angle of pitch from nominal ejection attitude (at approximately 4:33 P.E.T.).
5. Trim to nominal OMS burn attitude, if necessary.
6. Begin 35-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	15
Aft Tanks	121
Total	136

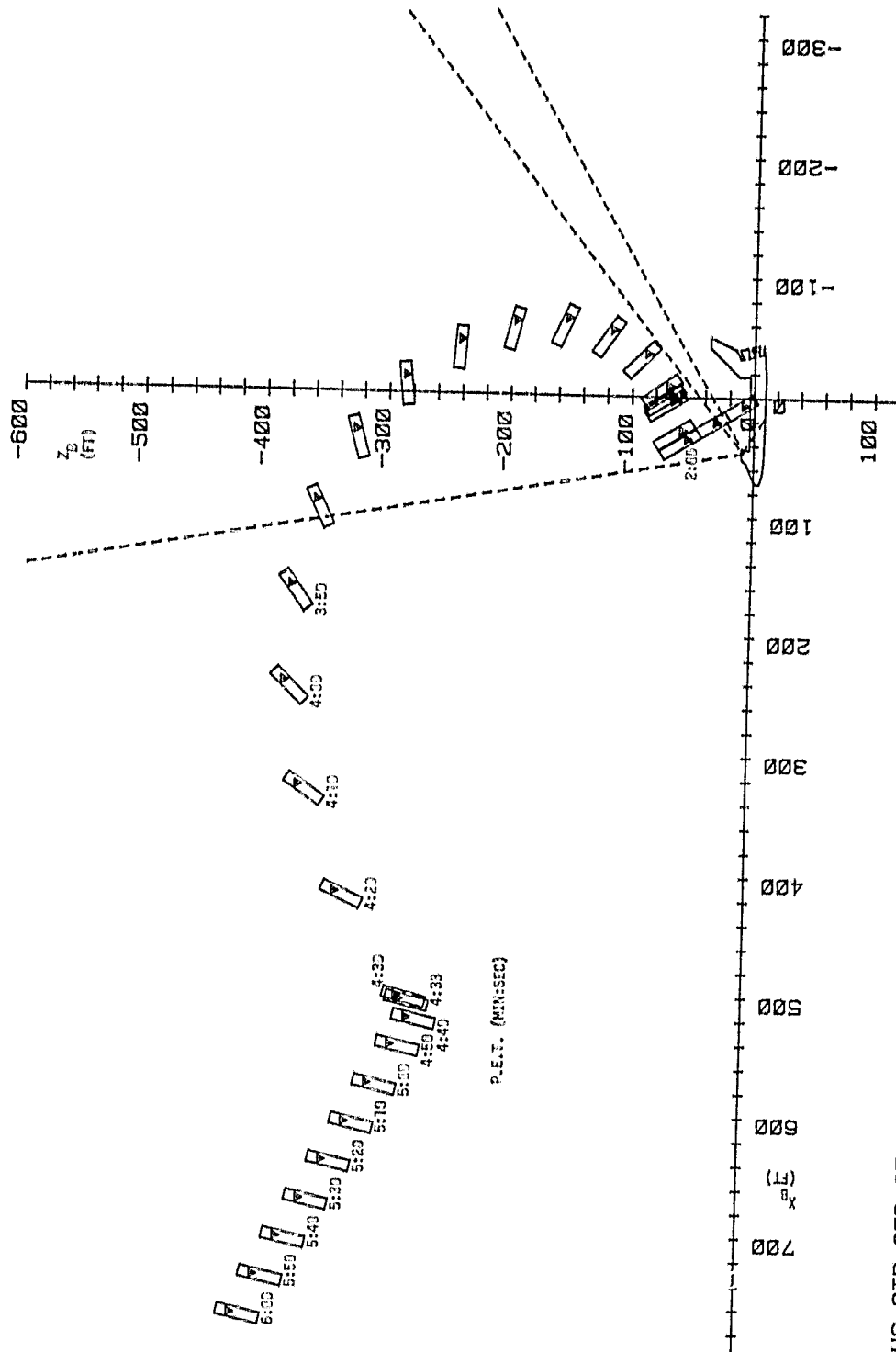


Figure 3-25. Enlarged View of Orbiter/IUS Initial Separation Phase (+135 Deg Rotation Between Ejection and OMS Ignition)

IUS STD SEP SEQ (PITCH +135 DEG)

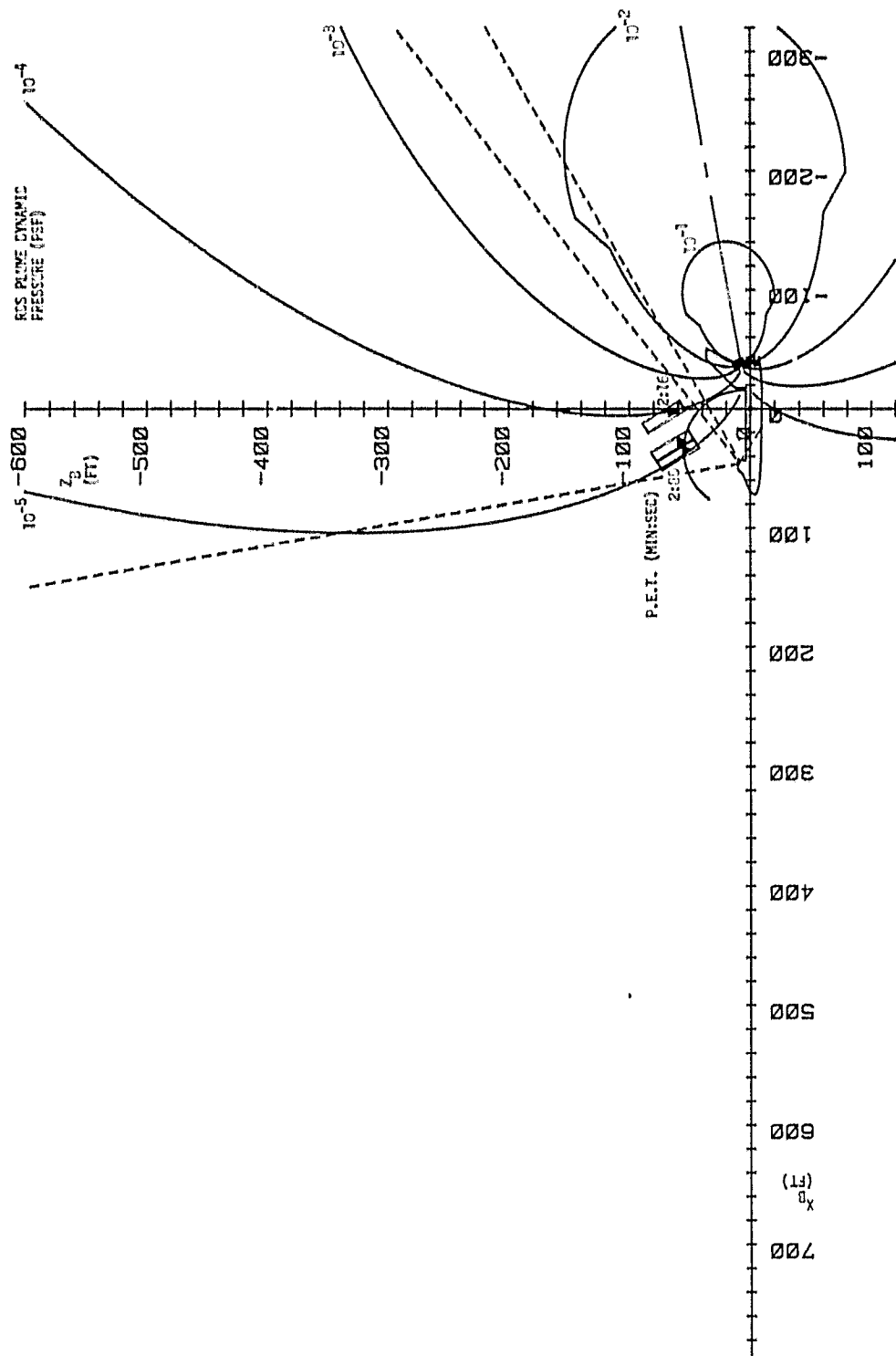


Figure 3-26. IUS Position Relative to +135 Plume
(+135 deg Rotation Between Ejection and
CNS Ignition)

IUS STD SEP SEQ (PITCH +135 DEG)

3.2.8 Orbiter Rotation Angle = +180 Degrees

Table 3-9 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/IUS initial separation sequence and Figure 3-27 presents an enlarged view of the early stage (to 6:00 P.E.T.). Examination of Figure 3-27 will show that the payload is visible through the pilot's overhead window almost continuously during the first 3 minutes and 30 seconds of the sequence.

Of the positive rotation sequences, this one is unique because two +X RCS translation maneuvers are required to place the Orbiter approximately 1400 feet behind and 475 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 3-9). As in the other positive rotation cases, the first +X RCS burn (12-second) occurs at 2:00 P.E.T. Immediately upon completion of the burn, the +1.0 deg/sec pitch rate is initiated. At 3:30 P.E.T., the second +X RCS burn is performed for 10 seconds. This burn is required to control the vertical displacement as well as the horizontal displacement at OMS ignition by preventing the payload from looping below ($Z_B \rightarrow +$) the Orbiter at the completion of the 180-degree pitch maneuver. The burn is performed at this time to minimize plume impingement on the payload (see Figure 3-28).

Figure 3-28 illustrates the IUS positions relative to the +X RCS plume during the translational maneuvers. The maximum plume dynamic pressure experienced by the payload should be no greater than approximately 5×10^{-5} pounds per square foot during the 22-second total duration of the 2 RCS translation maneuvers.

Table 3-9. Orbiter/IUS Initial Separation Sequence
(+180 Deg Rotation Between Ejection and
OMS Ignition)

1. Begin inertial hold at 1:00 P.E.T.
2. Initiate 12-second +X RCS burn at 2:00 P.E.T.
3. Start +1.0 deg/sec pitch maneuver immediately after completion of +X burn.
4. At 3:30 P.E.T., burn +X for 10 seconds while maintaining +1.0 deg/sec pitch rate.
5. Resume inertial hold upon reaching +180 deg angle of pitch from nominal ejection attitude (at approximately 5:14 P.E.T.).
6. Trim to nominal OMS burn attitude, if necessary.
7. Begin 35-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	14
Aft Tanks	157
Total	171

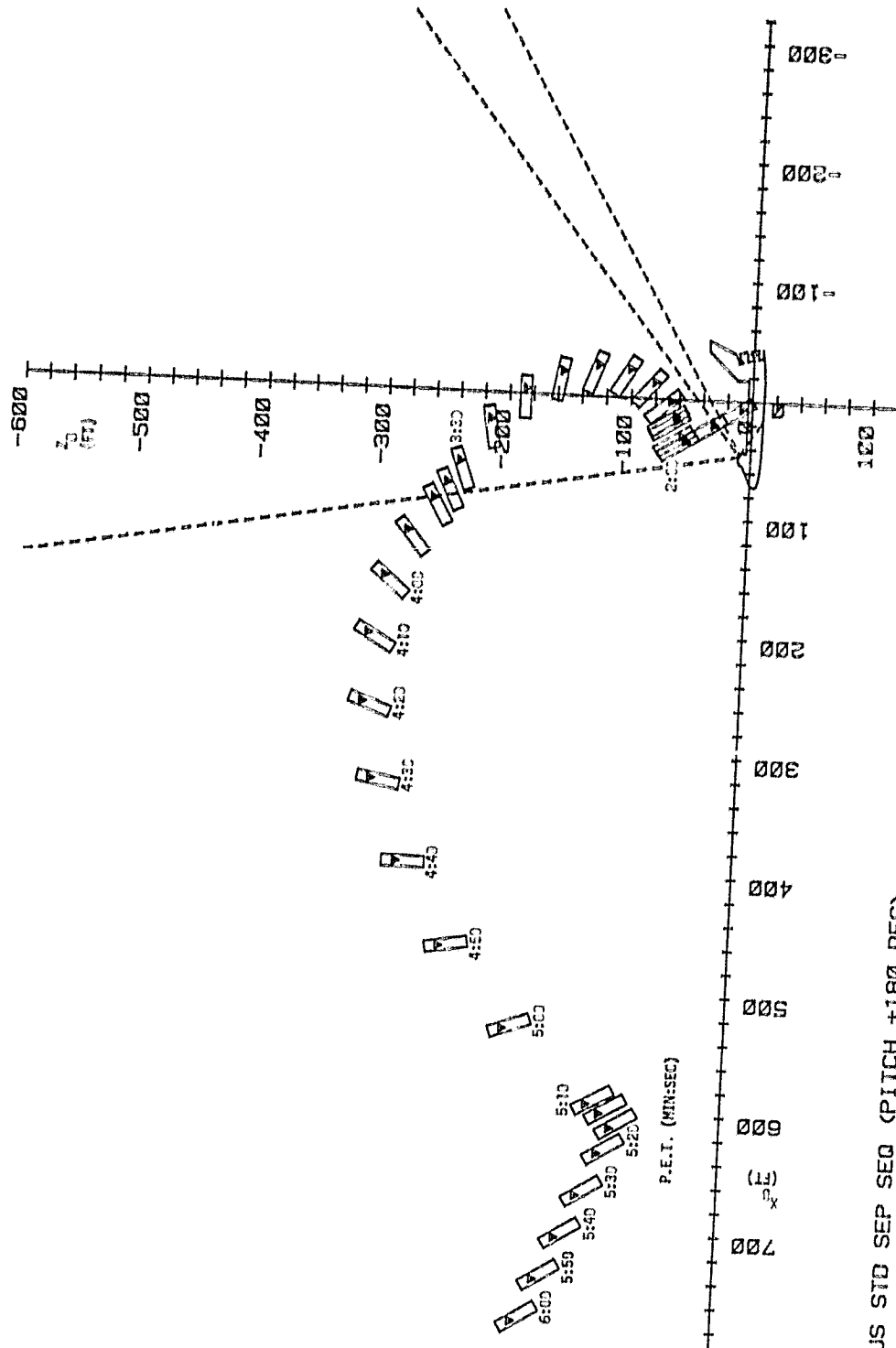
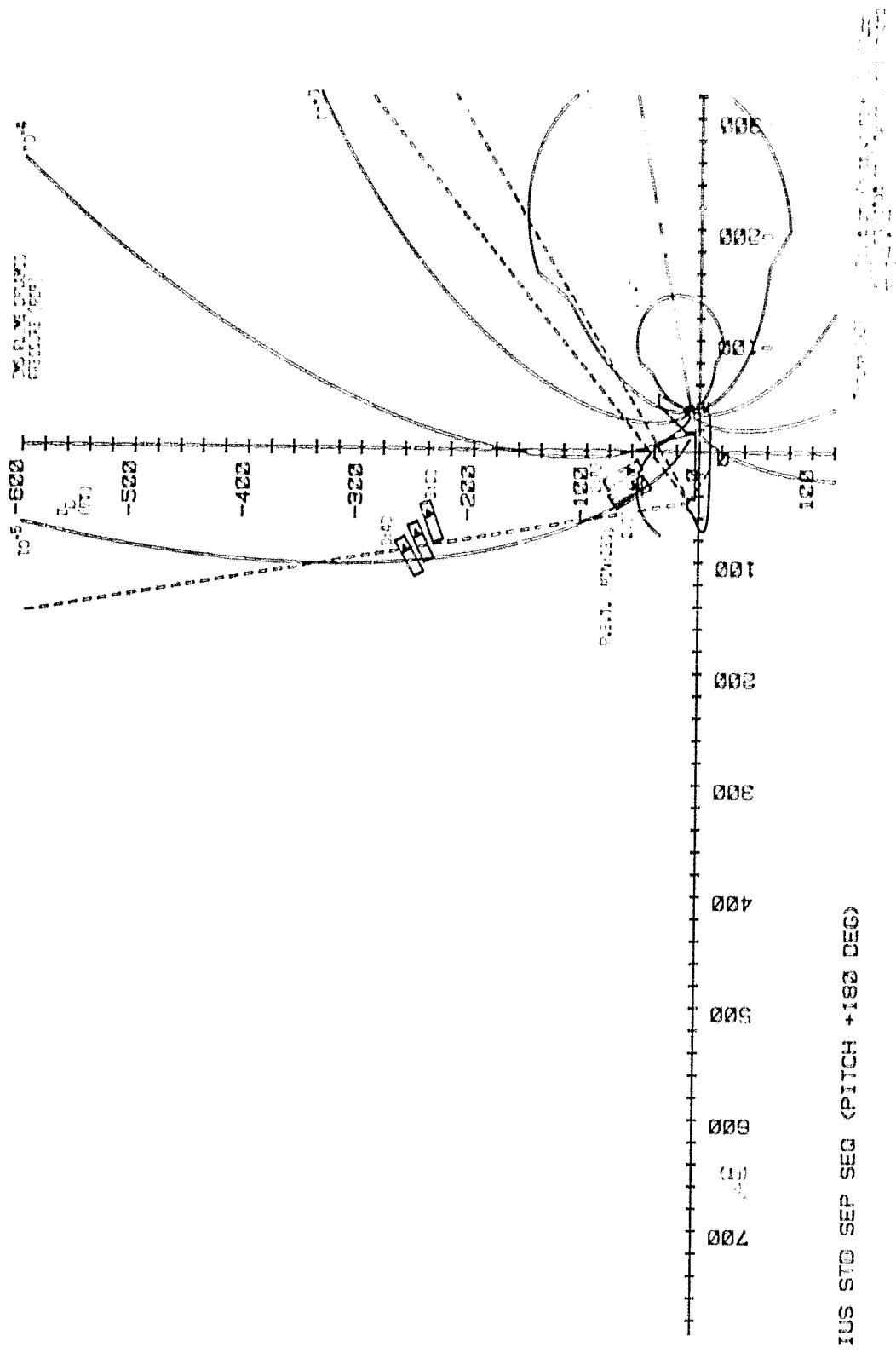


Figure 3-27. IUS STD SEP SEQ (Pitch +180 DEG)
Separation Error (ft) vs. P.E.T. (MIN:SEC)
Between Ejection and 300 ft/sec

IUS STD SEP SEQ (PITCH +180 DEG)



4.0 DEFINITION OF STANDARD ORBITER/SSUS-A SEPARATION SEQUENCES

4.1 ASSUMPTIONS AND DESIGN REQUIREMENTS

1. For trajectory calculation purposes, the mass properties of the Orbiter were assumed to be as listed in Table 3-1 immediately after ejection of the payload from the cargo bay.
2. The payload (SSUS-A+satellite) was modeled as a right circular cylinder 106 inches in diameter and 346 inches in length.
3. The payload will be ejected by springs from a tilt table in the cargo bay, with a relative velocity increment of 1.25 feet per second in the direction of the SSUS-A +X body axis (which is aligned with the SSUS-A SRM thrust vector).
4. At the time of ejection, the tilt table will be elevated so that the SSUS-A +X axis makes an angle of 45 degrees (measured in the Orbiter's plane of symmetry) with the Orbiter +X body axis.
5. The Orbiter attitude is to be inertially stabilized as nearly as possible with all RCS jets inactive at the time of payload ejection. Orbiter RCS jets are to remain inactive until the payload has cleared all Orbiter-fixed structure by at least TBD feet.
6. The line of action of the nominal spring-ejection impulse is assumed to pass through the payload center of gravity (CG), producing no angular perturbation of that vehicle. For the nominal pre-ejection location of the SSUS-A that is represented in the figures that follow, the line of action has a moment arm of some 91 inches about the Orbiter CG and would produce a small positive Orbiter pitch rate. However, since the actual location of the SSUS-A in the cargo bay may deviate considerably from that shown, the Orbiter's reactive angular rate impulse was modeled as if it were zero.
7. The payload must be continuously visible to the Orbiter flight crew after ejection until such time as positive separation has been confirmed during or immediately after the Orbiter's final separation maneuver (i.e., the OMS burn).
8. Plume impingement on the payload from Orbiter thrusters is to be minimized.

4.2 MANEUVER SEQUENCE DESCRIPTIONS

Eight discrete Orbiter/SSUS-A initial separation phase maneuver sequences have been designed. In Section 4.2.1 - 4.2.8 each sequence will be described by a table listing the schedule of events and by figures depicting the motion of the payload relative to the Orbiter. Three types of relative motion plots are shown to clarify each sequence: (1) an overall view of the Orbiter/SSUS-A initial separation phase including OMS plume contours, (2) an enlarged view of the early stage of the sequence, and (3) the SSUS-A position relative to the RCS plume contours during RCS translational maneuvers. All of the data presented in this section were obtained by use of the HP-9825A HFRMP, Version 03M (References 1-3).

All rotational and translational maneuvers of the Orbiter were simulated with finite-thrust models. The PRCS was used for Orbiter attitude control. To minimize plume impingement on the payload, the +Z PRCS jets (i.e., those which expel propellant directly upward relative to the body-fixed frame) were inhibited. Attitude-control deadbands were set at 0.5 degrees per axis.

Figure 4-1 presents the nominal RCS propellant consumption and range of applicability for each of the eight standard Orbiter/SSUS-A separation sequences. Figures 4-2 through 4-9 present an overall view of the Orbiter/SSUS-A initial separation phase depicting the motion of the payload relative to the Orbiter for each sequence. In each sequence, a 28-second OMS maneuver is assumed to begin 9 minutes after ejection (P.E.T. = 9:00). The plume contours for the OMS are overlaid on the relative motion plots. Examination of the figures will show that OMS plume impingement on the payload should be completely negligible even allowing for RCS maneuver execution errors that might produce a trajectory dispersion of 300 feet or so at the time of OMS ignition.

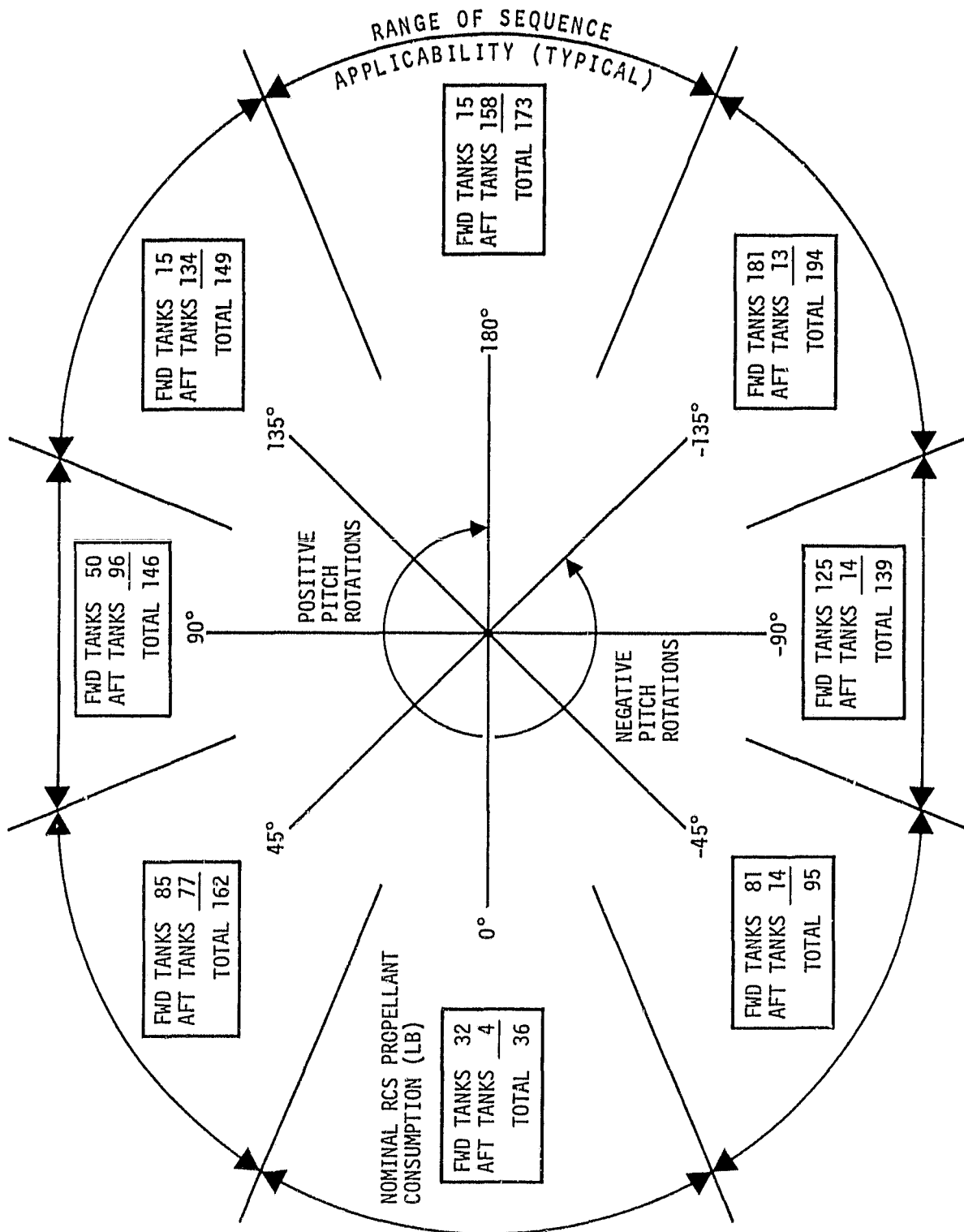
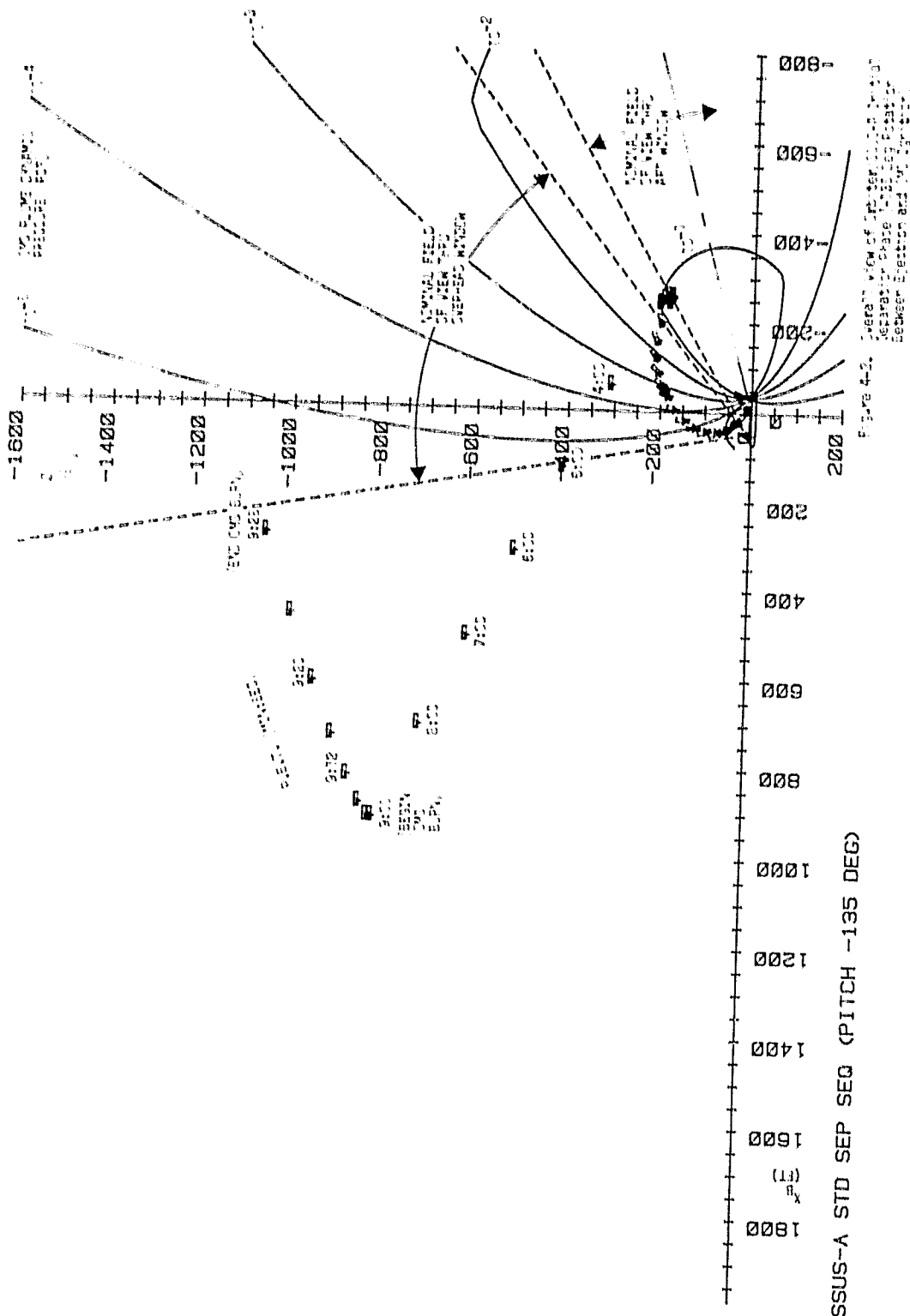
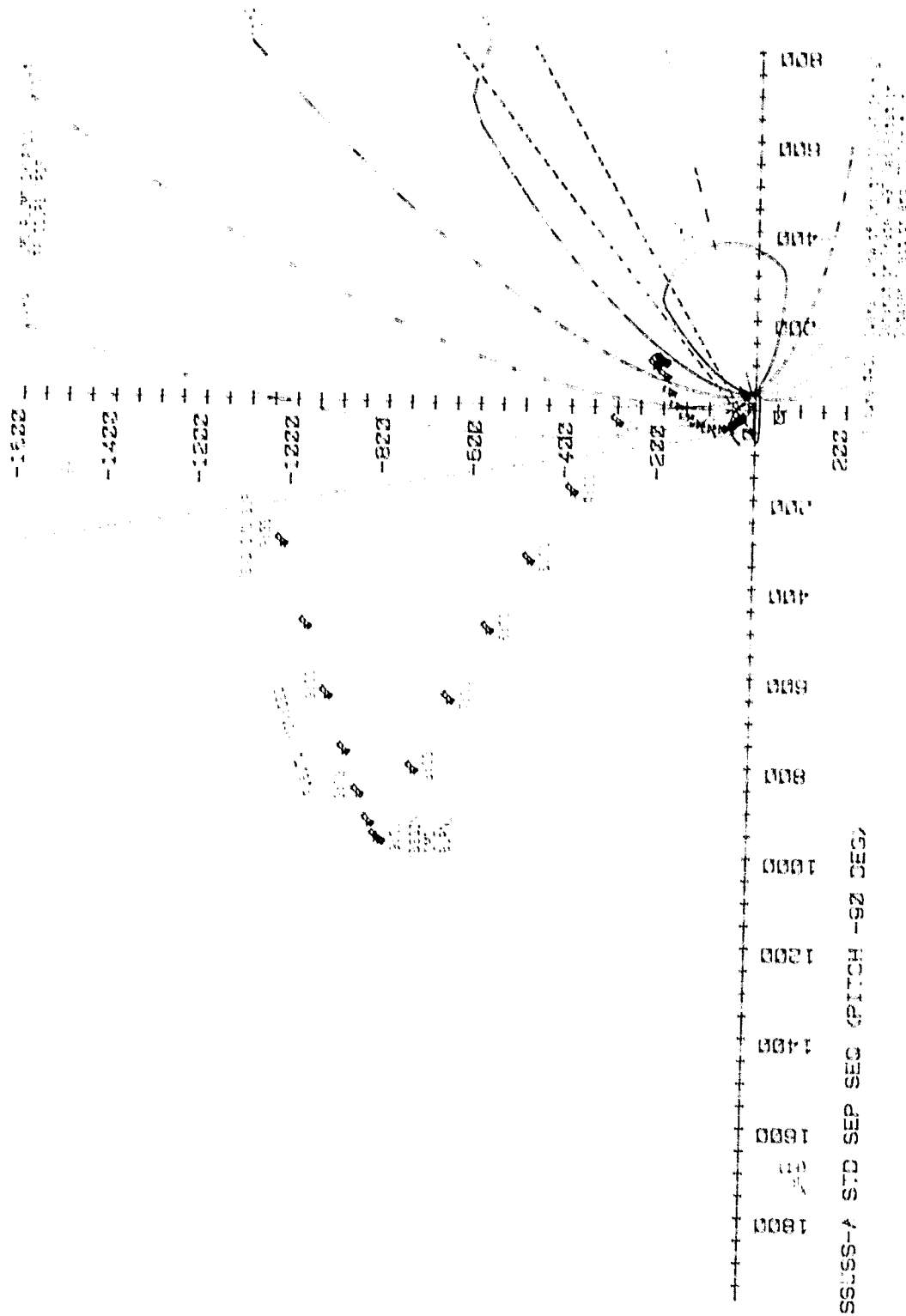


Figure 4-1. Nominal RCS Propellant Consumption for Eight Standard Orbiter/SSUS-A Separation Sequences at Intervals of 45 Degrees in the Angle of Rotation Between Ejection and OMS Ignition





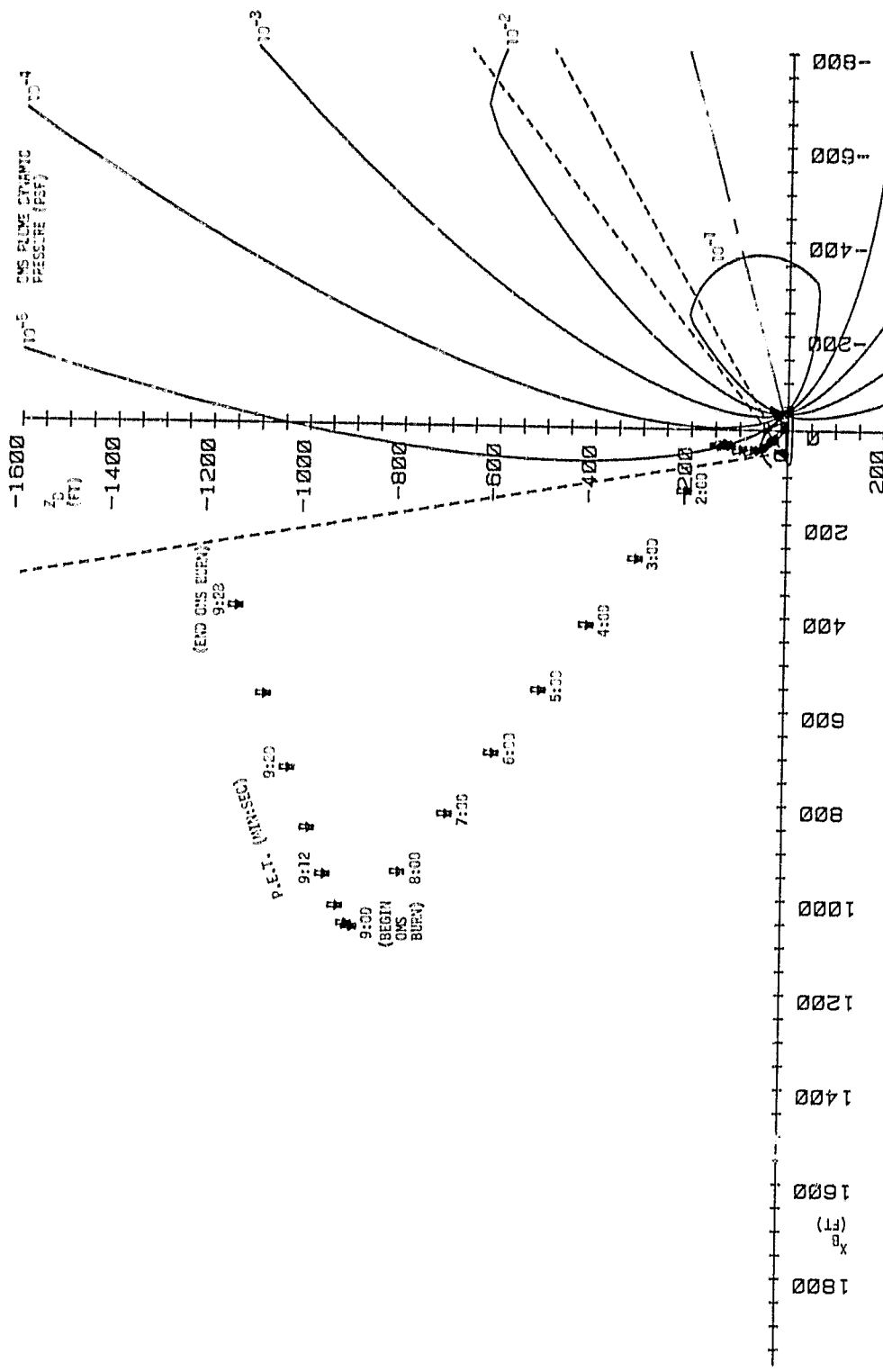


Figure 4-4. Overall View of Orbiter/SSUS-A Initial Separation Phase (-45 Deg Pitch Between Ejection and ONS Ignition)

SSUS-A STD SEP SEQ (PITCH -45 DEG)

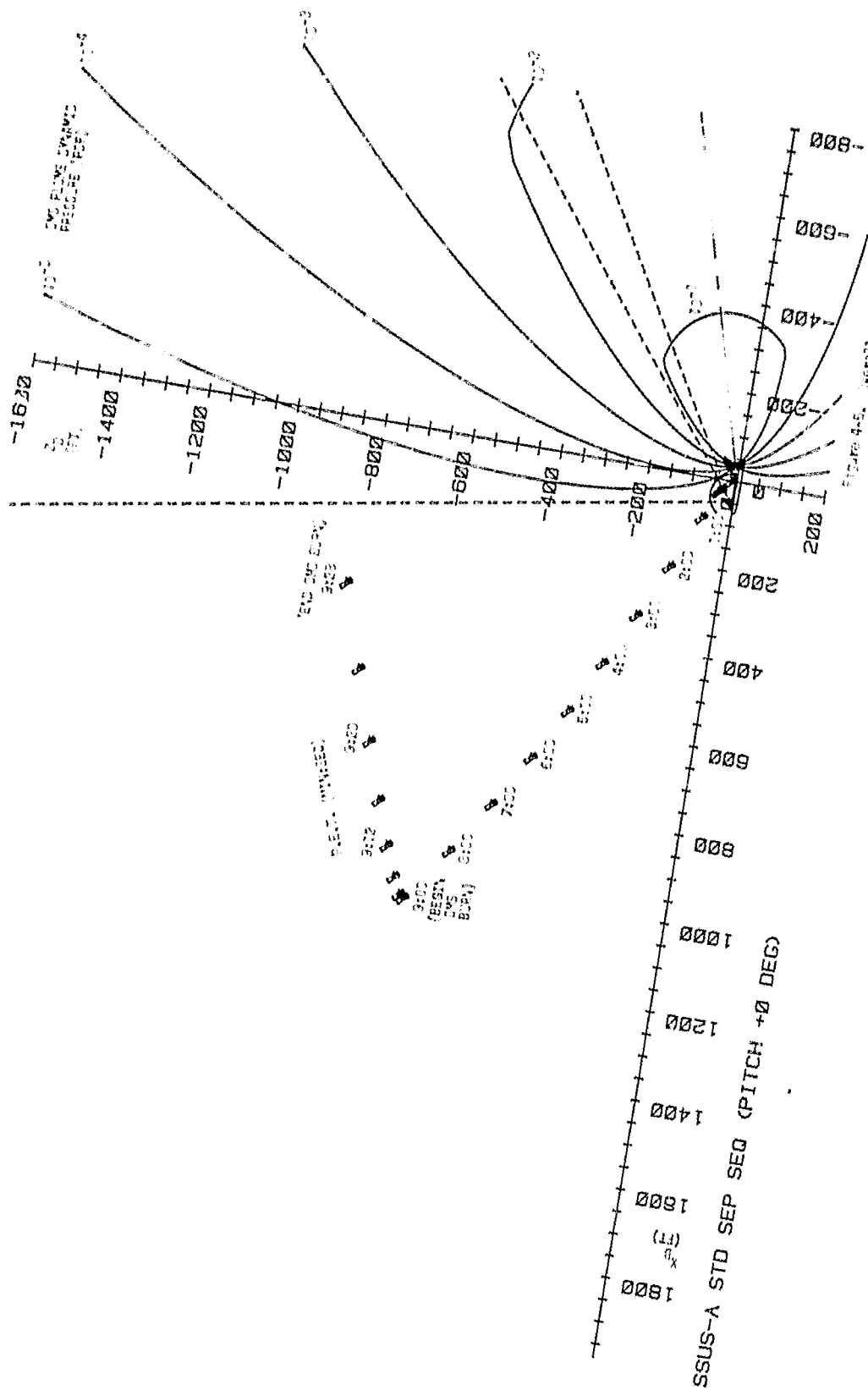
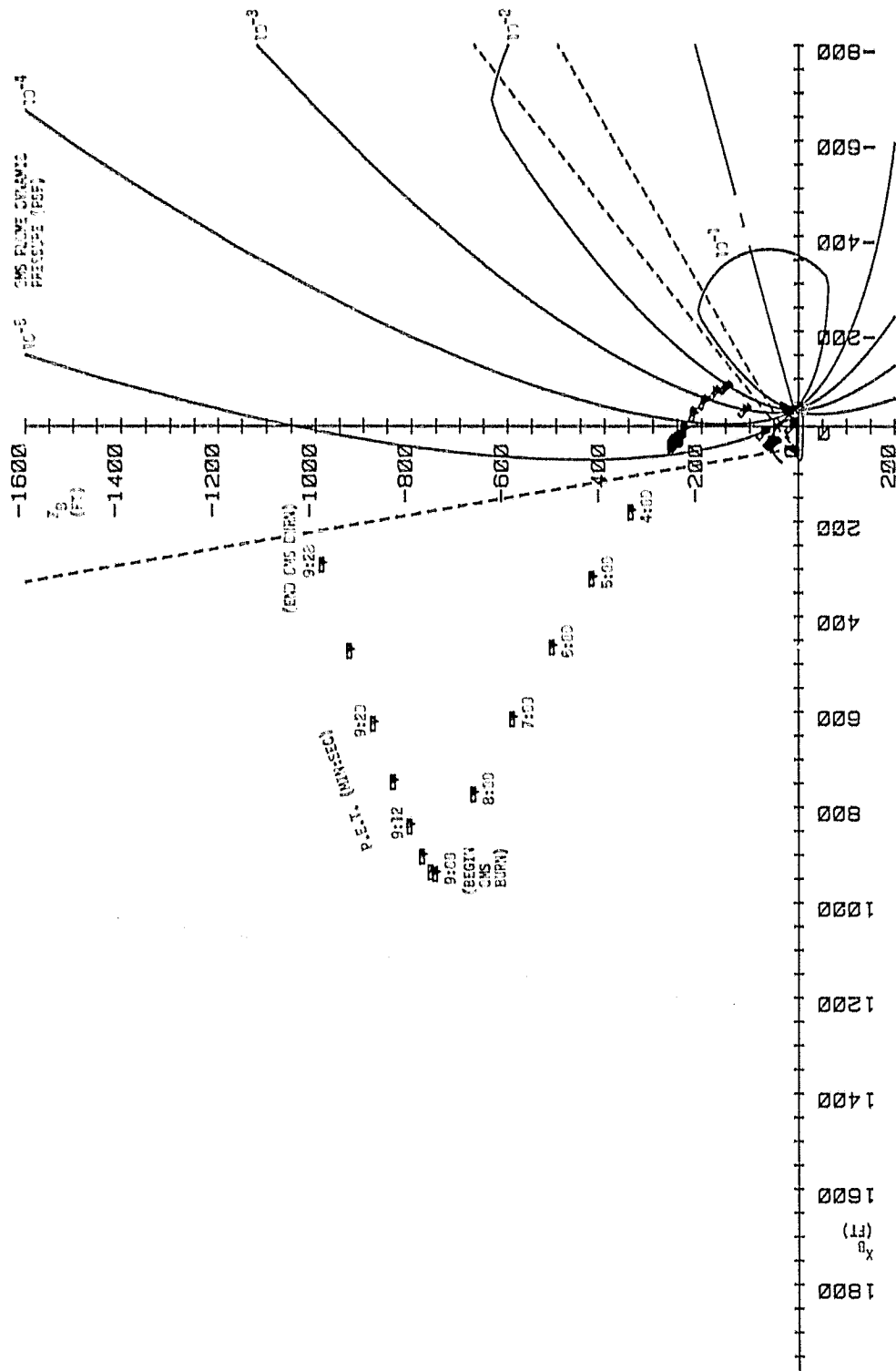


Figure 4-5
 Detail view of the
 SSUS-A STD SEP SEQ
 curves. The curves are
 plotted for a pitch of
 +0 degrees.



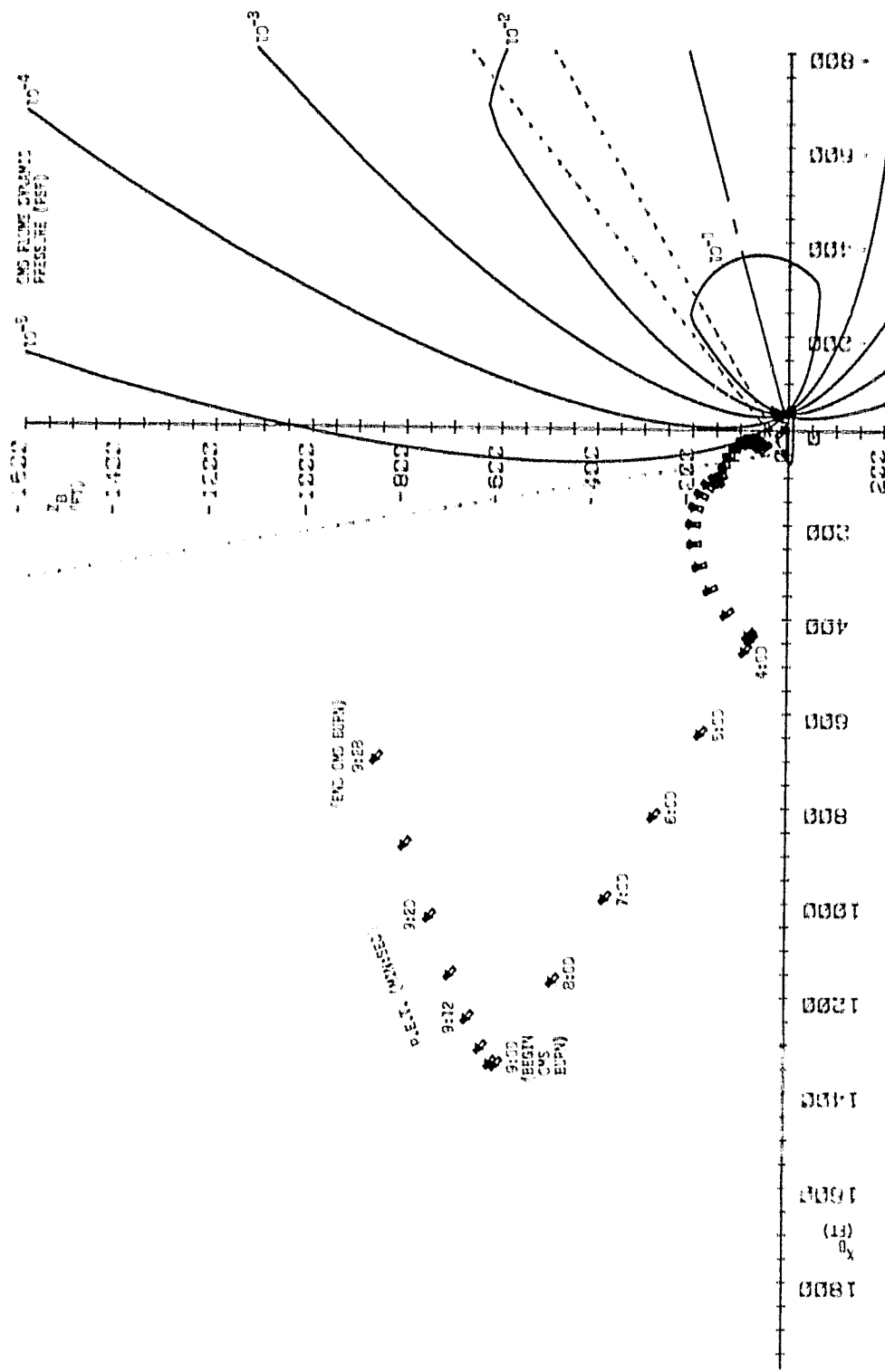


Figure 4-9. Overall View of Inhibitor (SSUS-A Initial Separation Plane) (180 deg Rotation Between Ejection and COS Ignition)

SSUS-A STD SEP SEC (PITCH +180 DEG)

4.2.1 Orbiter Rotation Angle = -135 Degrees

Table 4-1 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-A initial separation sequence and Figure 4-10 presents an enlarged view of the early stage (to 6:00 P.E.T.). Examination of Figure 4-10 will show that the payload is visible through the pilot's overhead window almost continuously during the first 5 minutes of the sequence.

The Orbiter's motion was uncontrolled from 0:00 P.E.T. until 0:25 P.E.T., at which time a 4-second -X RCS burn was initiated to provide a velocity component in the Orbiter +Z body-fixed direction without using the inhibited +Z RCS jets. Examination of Figure 4-10 will show that the payload is clear of the Orbiter structure at the time of the RCS burn. Immediately after the burn, a pitch rate of -1.0 deg/sec was established. At 1:40 P.E.T. a second -X RCS burn was performed for 5 seconds while maintaining the pitch rate. This burn was required to prevent the payload from going too far behind and too close to the Orbiter ($Z_B \rightarrow 0$ FT), blocking the view of the payload behind the OMS pods. The burn time was selected to minimize plume impingement. At approximately 2:44 P.E.T., the -135 degree angle of pitch from the nominal ejection attitude has been reached, an inertial hold initiated, and a 15-second -X RCS burn performed. This final translation places the Orbiter approximately 925 feet behind and 800 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 4-2).

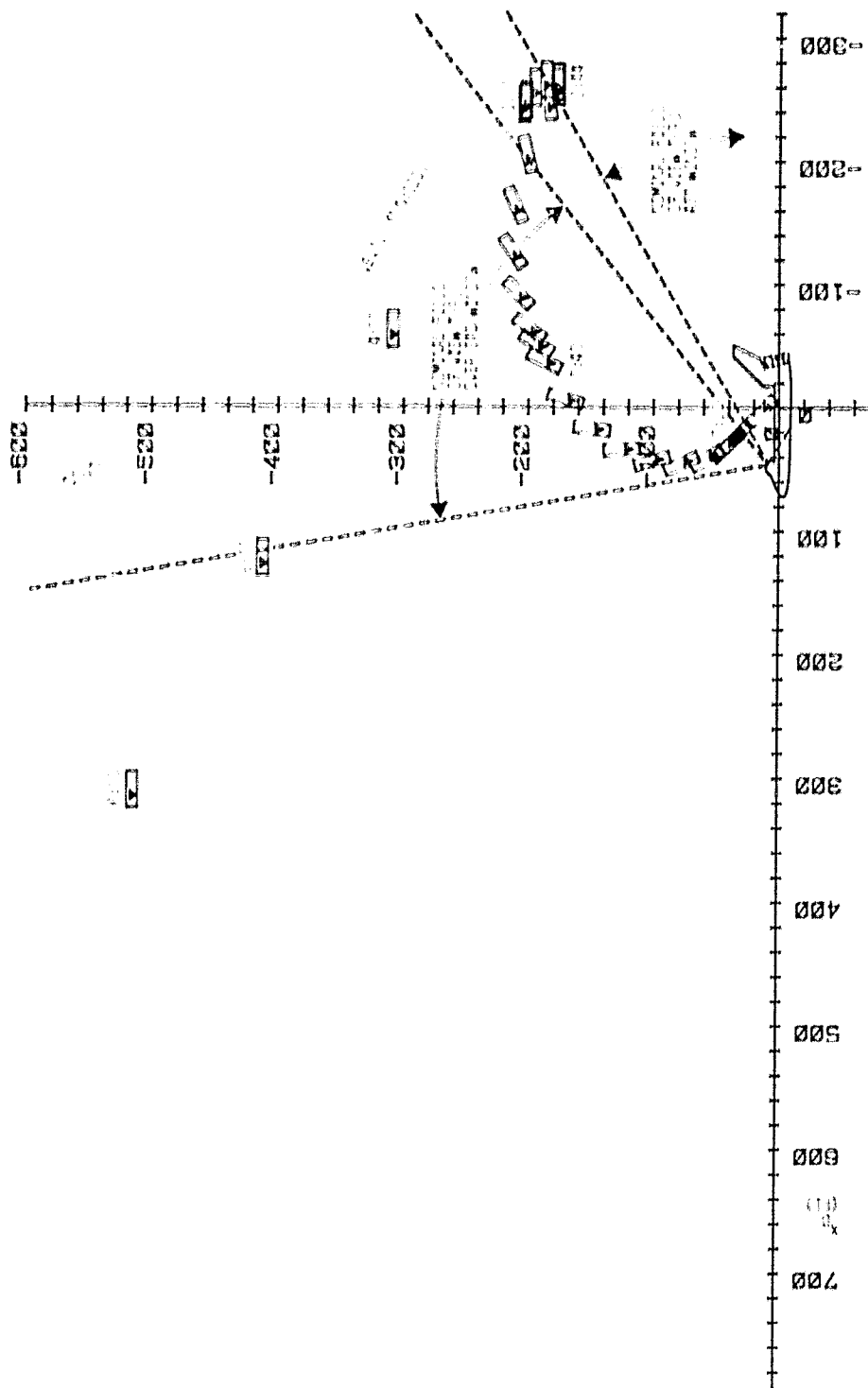
Figure 4-11 illustrates the SSUS-A positions relative to the -X RCS plume during the 3 translational maneuvers. Because of the high degree of plume expansion, impingement cannot be avoided entirely. However, RCS plume dynamic pressure experienced by the payload should be no greater than approximately 5×10^{-4} pounds per square foot during the 24-second total duration of the 3 RCS translational maneuvers.

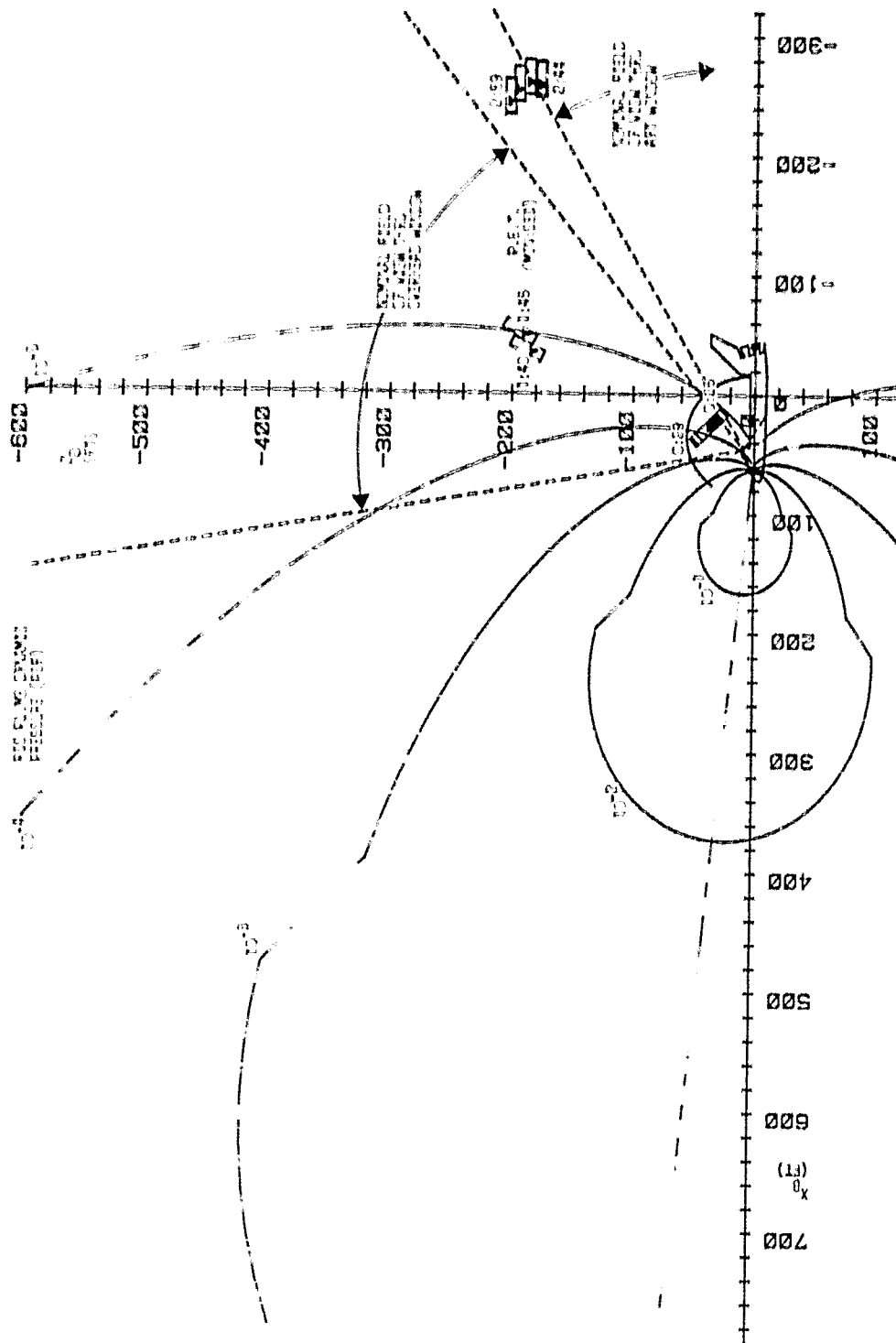
Table 4-1. Orbiter/SSUS-A Initial Separation Sequence
 (-135 Deg Rotation Between Ejection and
 OMS Ignition)

1. Initiate 4-second -X RCS burn at 0:25 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at -1.0 deg/sec immediately after -X burn.
3. At 1:40 P.E.T., burn -X for 5 seconds while maintaining -1.0 deg/sec pitch rate.
4. Begin inertial hold upon reaching -135 deg angle of pitch from nominal ejection attitude (at approximately 2:44 P.E.T.).
5. Execute 15-second -X burn immediately after beginning inertial hold.
6. Trim to nominal OMS burn attitude, if necessary.
7. Begin 28-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION

Fwd Tanks	181
Aft Tanks	<u>13</u>
Total	194





100-443887-100

SSUS-A STD SEP SEQ (PITCH -135 DEG)

4.2.2 Orbiter Rotation Angle = -90 Degrees

Table 4-2 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-A initial separation sequence and Figure 4-12 presents an enlarged view of the early stage (to 6:00 P.E.T.). Examination of Figure 4-12 will show that the payload is visible through the pilot's overhead window almost continuously during the first 3 minutes and 30 seconds of the sequence.

The initiation of this sequence is similar to the -135 degree rotation case with a -1.0 deg/sec pitch rate established immediately upon completion of the 4-second -X RCS burn at 0:25 P.E.T. At approximately 1:59 P.E.T., the -90 degree angle of pitch rotation from the nominal ejection attitude has been reached, an inertial hold initiated, and a 12-second -X RCS burn begun. This burn is less than the final translational maneuver required in the -135 degree rotation case because the payload is not as far behind the Orbiter at the completion of the pitch maneuver. This burn places the Orbiter approximately 975 feet behind and 800 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 4-3).

Figure 4-13 illustrates the SSUS-A positions relative to the -X RCS plume during the 2 translational maneuvers. The payload should experience plume impingement not greater than approximately 5×10^{-4} pounds per square foot during the 16-second total duration of the 2 RCS translational maneuvers.

Table 4-2. Orbiter/SSUS-A Initial Separation Sequence
(-90 Deg Rotation Between Ejection and
OMS Ignition)

1. Initiate 4-second -X RCS burn at 0:25 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at -1.0 deg/sec immediately after -X burn.
3. Begin inertial hold upon reaching -90 deg angle of pitch from ejection attitude (at approximately 1:59 P.E.T.).
4. Execute 12-second -X burn immediately after beginning inertial hold.
5. Trim to nominal OMS burn attitude, if necessary.
6. Begin 28-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	125
Aft Tanks	14
Total	139

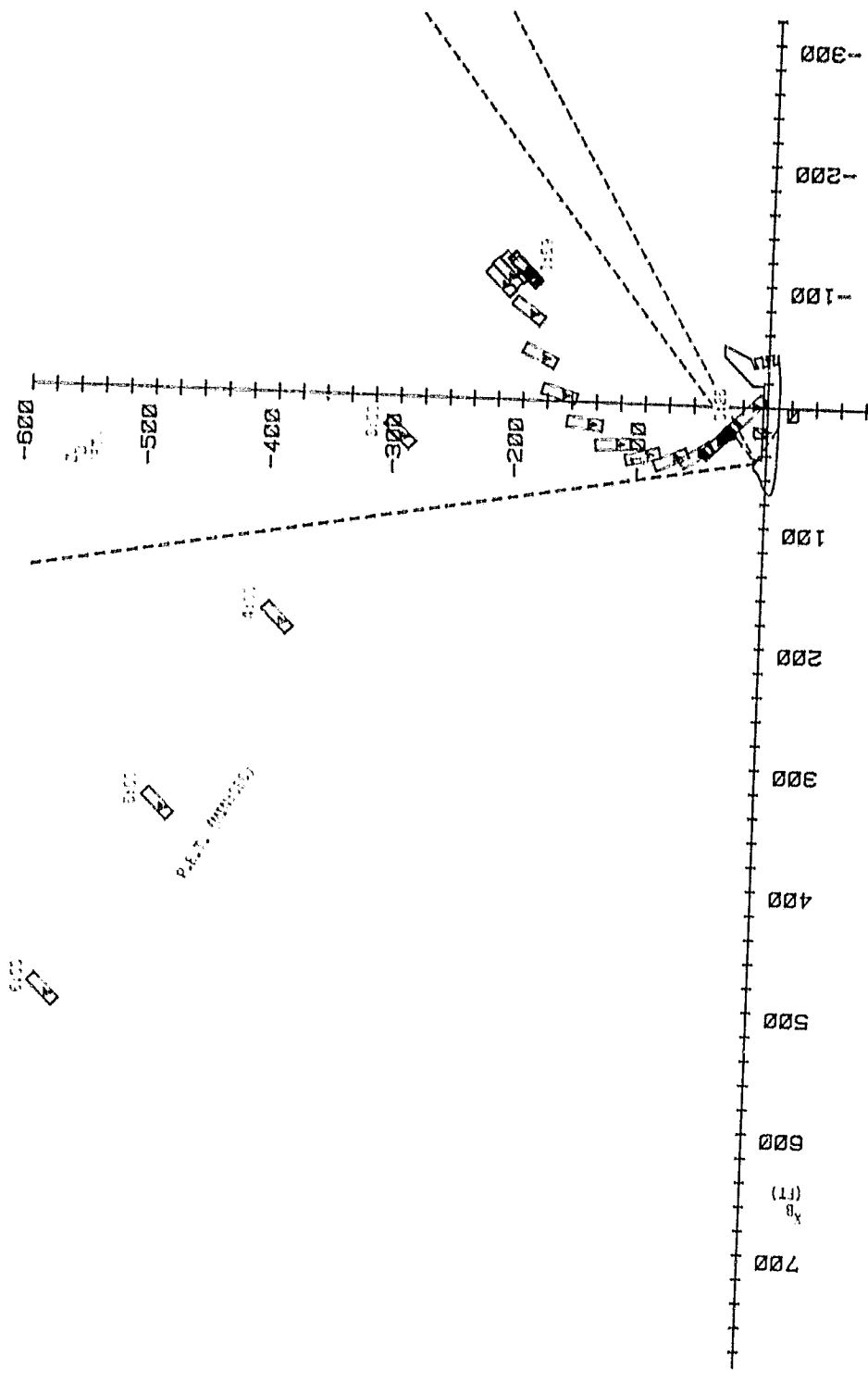


Figure 4-12. Enlarged view of target area. Labels on plot are: 100, 200, 300, 400, 500, 600, 700. Between Station and 100 ft.

SSUS-A STD SEP SEQ (PITCH -90 DEG)

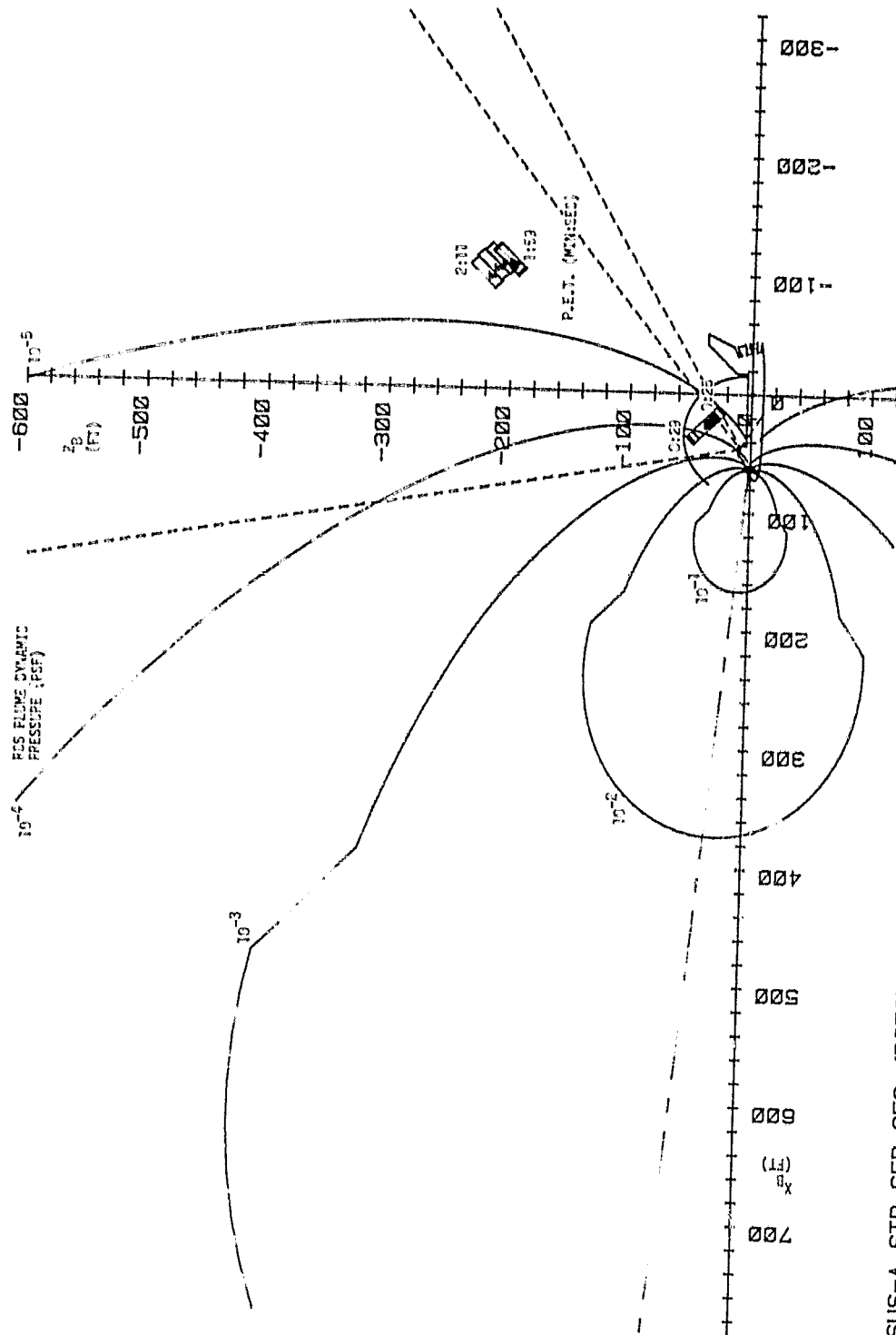


Figure 4-13. SSUS-A Positions Relative to -X FCC plume (-90 Deg Rotation Between Electron and QWS Ionizers)

SSUS-A STD SEP SEQ (PITCH -90 DEG)

4.2.3 Orbiter Rotation Angle = -45 Degrees

Table 4-3 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-A initial separation sequence and Figure 4-14 presents an enlarged view of the early stage (to 5:00 P.E.T.). Examination of Figure 4-14 will show that the payload is visible through the pilot's overhead window almost continuously during the first minute and a half of the sequence.

The initiation of this sequence is similar to the -135 and -90 degree rotation cases with a -1.0 deg/sec pitch rate established immediately upon completion of the 4-second -X RCS burn at 0:25 P.E.T. At approximately 1:14 P.E.T., the -45 degree angle of pitch rotation from the nominal ejection attitude has been reached, an inertial hold initiated, and a 6-second -X RCS burn begun. This burn is considerably shorter than the burns required in the -135 and -90 degree rotation cases because the payload is approximately directly overhead of the Orbiter at the completion of the pitch maneuver. This final RCS translation places the Orbiter approximately 1100 feet behind and 900 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 4-4).

Figure 4-15 illustrates the SSUS-A positions relative to the -X RCS plume during the 2 translational maneuvers. The payload should experience plume impingement not greater than approximately 5×10^{-4} pounds per square foot during the 10-second total duration of the 2 RCS translational maneuvers.

Table 4-3. Orbiter/SSUS-A Initial Separation Sequence
 (-45 Deg Rotation Between Ejection and
 OMS Ignition)

1. Initiate 4-second -X RCS burn at 0:25 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at -1.0 deg/sec immediately after -X burn.
3. Begin inertial hold upon reaching -45 deg angle of pitch from nominal ejection attitude (at approximately 1:14 P.E.T.).
4. Execute 6-second -X RCS burn immediately after beginning inertial hold.
5. Trim to nominal OMS burn attitude, if necessary.
6. Begin 28-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	81
Aft Tanks	<u>14</u>
Total	95

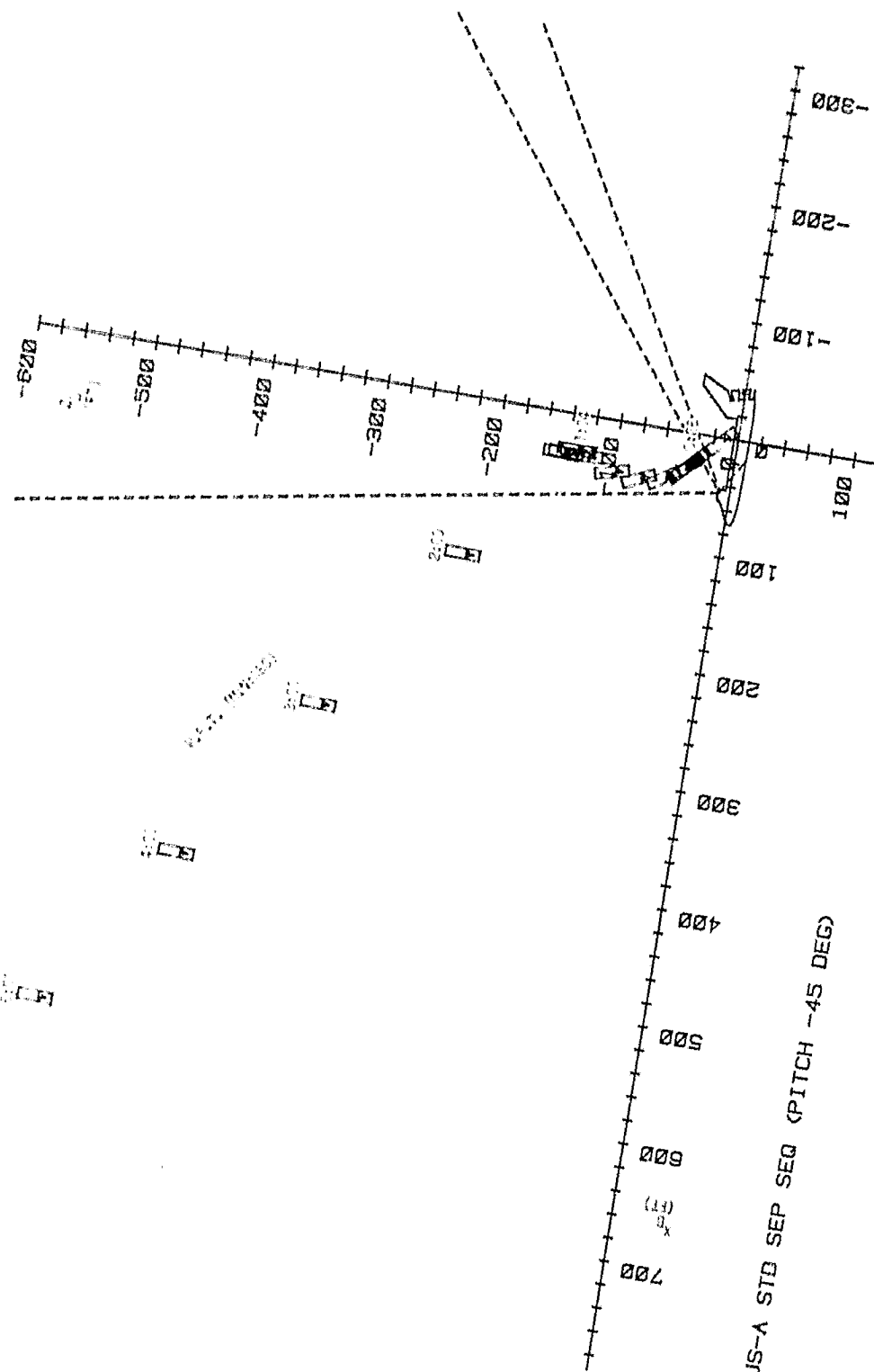
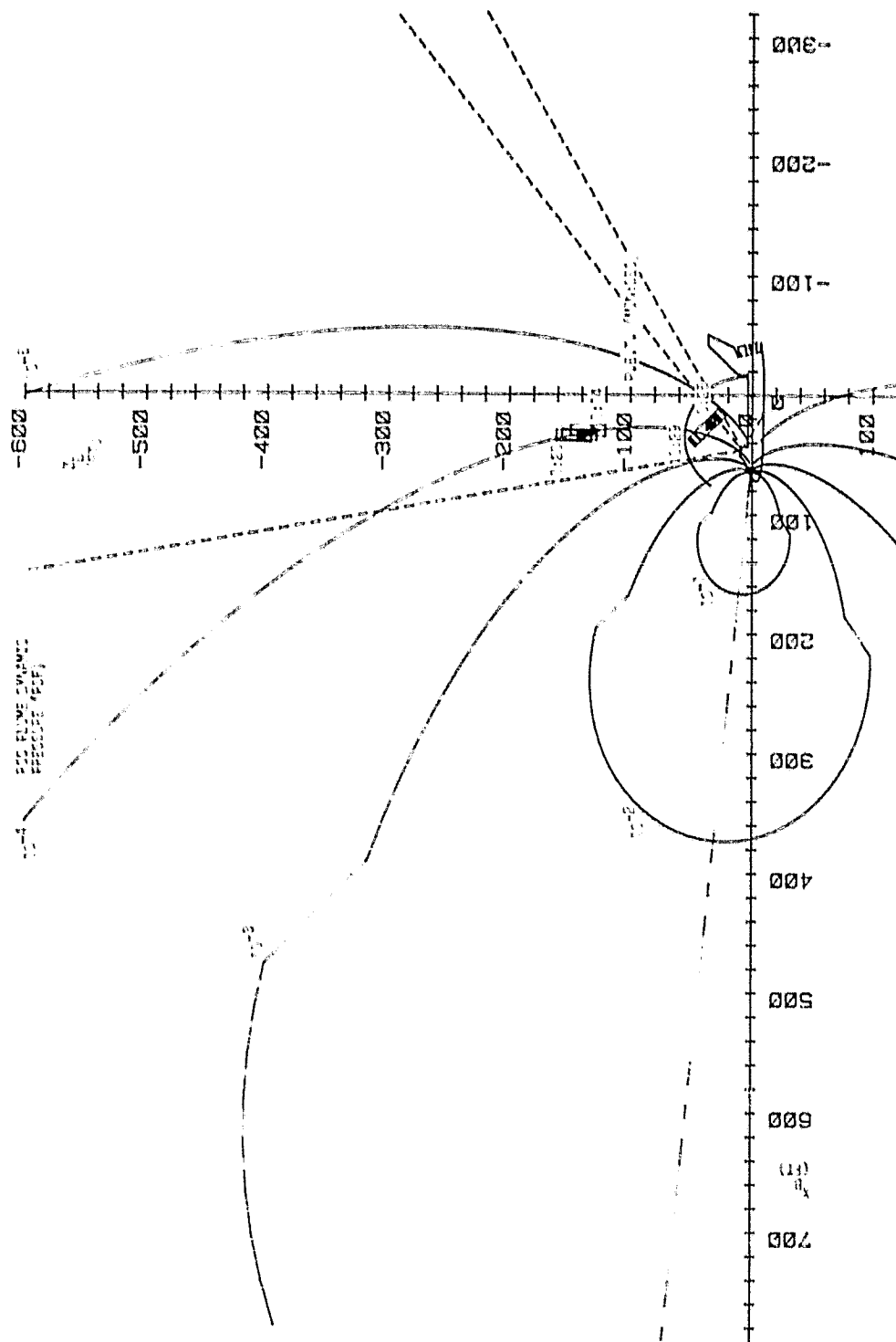


Figure 4-74. Separation of Vehicle and Target at Pitch -45 Degrees



1. 1
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3. 3
4. 4
5. 5
6. 6
7. 7
8. 8
9. 9
10. 10
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86. 86
87. 87
88. 88
89. 89
90. 90
91. 91
92. 92
93. 93
94. 94
95. 95
96. 96
97. 97
98. 98
99. 99
100. 100

4.2.4 Orbiter Rotation Angle = 0 Degrees

Table 4-4 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-A initial separation sequence and Figure 4-16 presents an enlarged view of the early stage (to 6:00 P.E.T.).

This sequence is unique in that the initial 4-second -X RCS burn at 0:25 P.E.T. provides sufficient translation to place the Orbiter approximately 975 feet behind and 550 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 4-5).

Figure 4-17 illustrates the SSUS-A positions relative to the -X RCS plume during the translational maneuver. The maximum plume dynamic pressure experienced by the payload should be no greater than approximately 5×10^{-4} pounds per square foot.

Table 4-4. Orbiter/SSUS-A Initial Separation Sequence
(0 Deg Rotation Between Ejection and OMS
Ignition)

1. Initiate 4-second -X RCS burn at 0:25 P.E.T.
2. Begin inertial hold immediately after -X burn.
3. Trim to nominal OMS burn attitude, if necessary.
4. Begin 28-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	32
Aft Tanks	<u>4</u>
Total	36

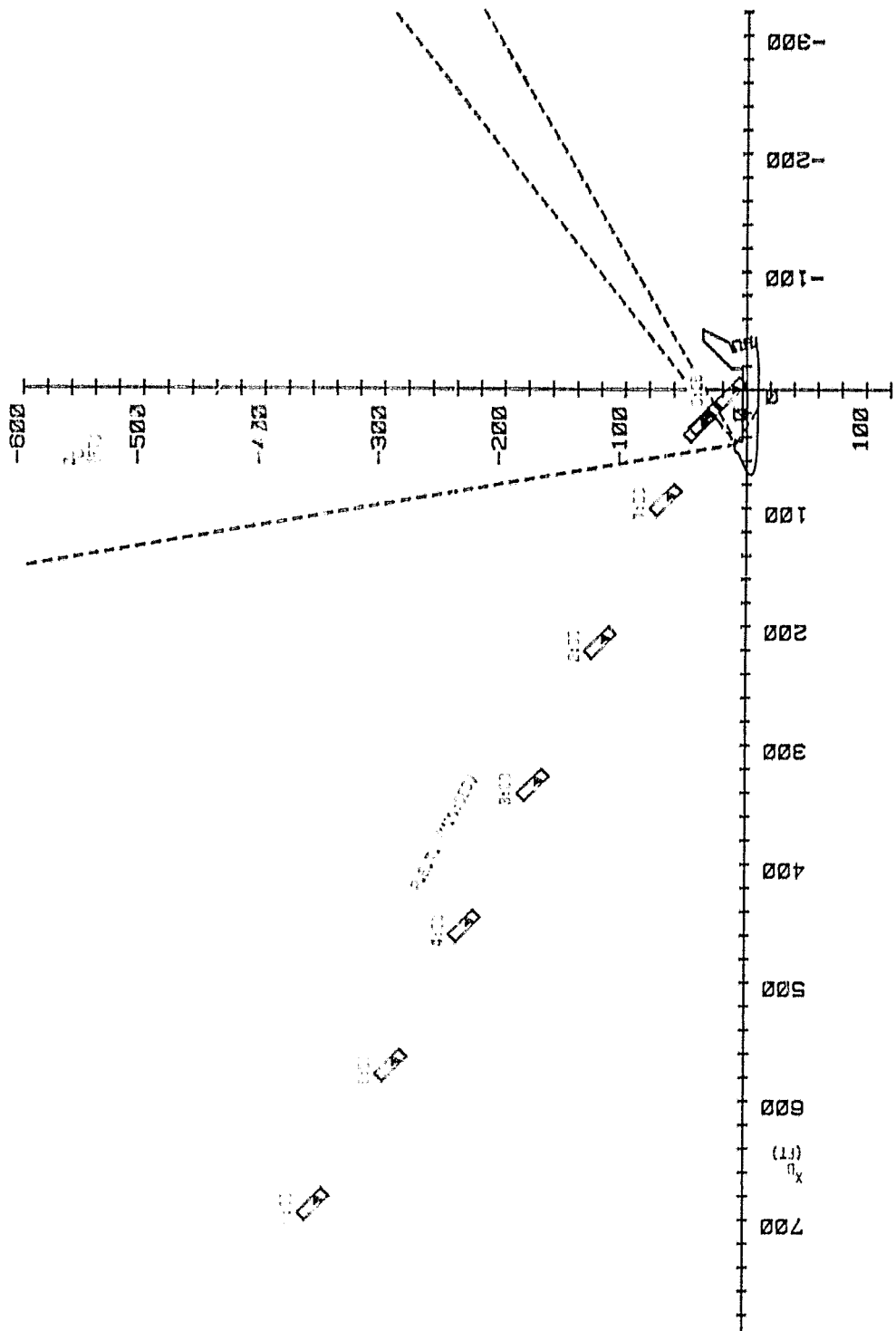


Figure 4-12. Enlarged View of Sequence Plot for SSUS-A
Separation Phase (Pitch +0 DEG)
Between Creation and 0.000000

SSUS-A STD SEP SEQ (PITCH +0 DEG)

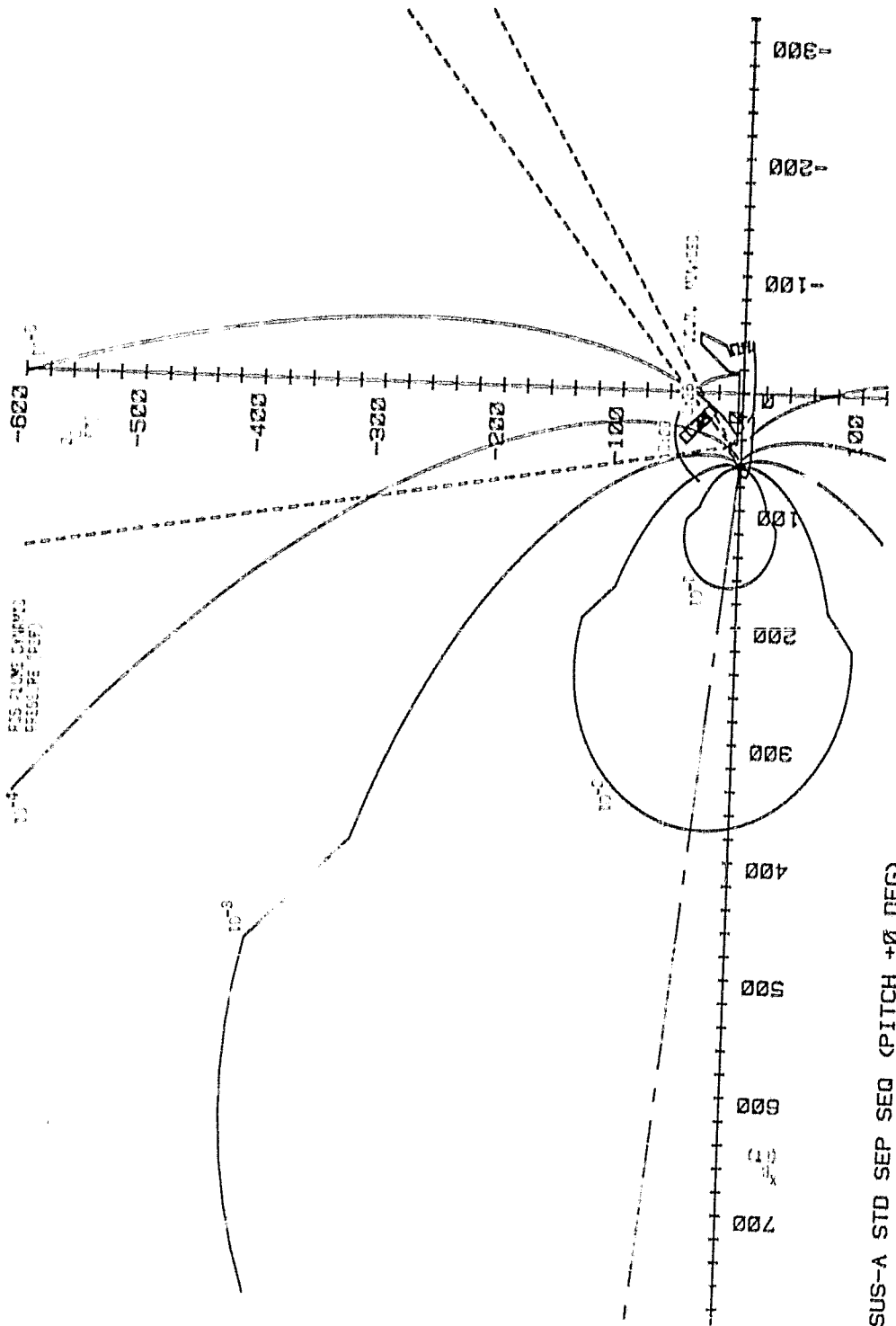


Figure 4-27. SSUS-A STD SEP SEQ (PITCH +0 DEG)

4.2.5 Orbiter Rotation Angle = +45 Degrees

Table 4-5 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-A initial separation sequence and Figure 4-18 presents an enlarged view of the early stage (to 7:00 P.E.T.). Examination of Figure 4-18 will show that the payload is visible through the pilot's overhead window almost continuously for the first 3 minutes of the sequence.

This, and the remaining sequences (positive Orbiter rotation angles), are initiated differently than the previous cases. The Orbiter's motion was uncontrolled from 0:00 P.E.T. until 0:40 P.E.T. at which time a 9-second +X RCS burn was performed. An inertial hold was established immediately after the burn. A 1.0 deg/sec pitch rate was initiated at 2:00 P.E.T. At approximately 2:47 P.E.T., the +45 degree angle of pitch rotation from the nominal ejection attitude has been reached, an inertial hold resumed, and a 10-second -X RCS burn begun. This final RCS translational maneuver places the Orbiter approximately 950 feet behind and 700 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 4-6).

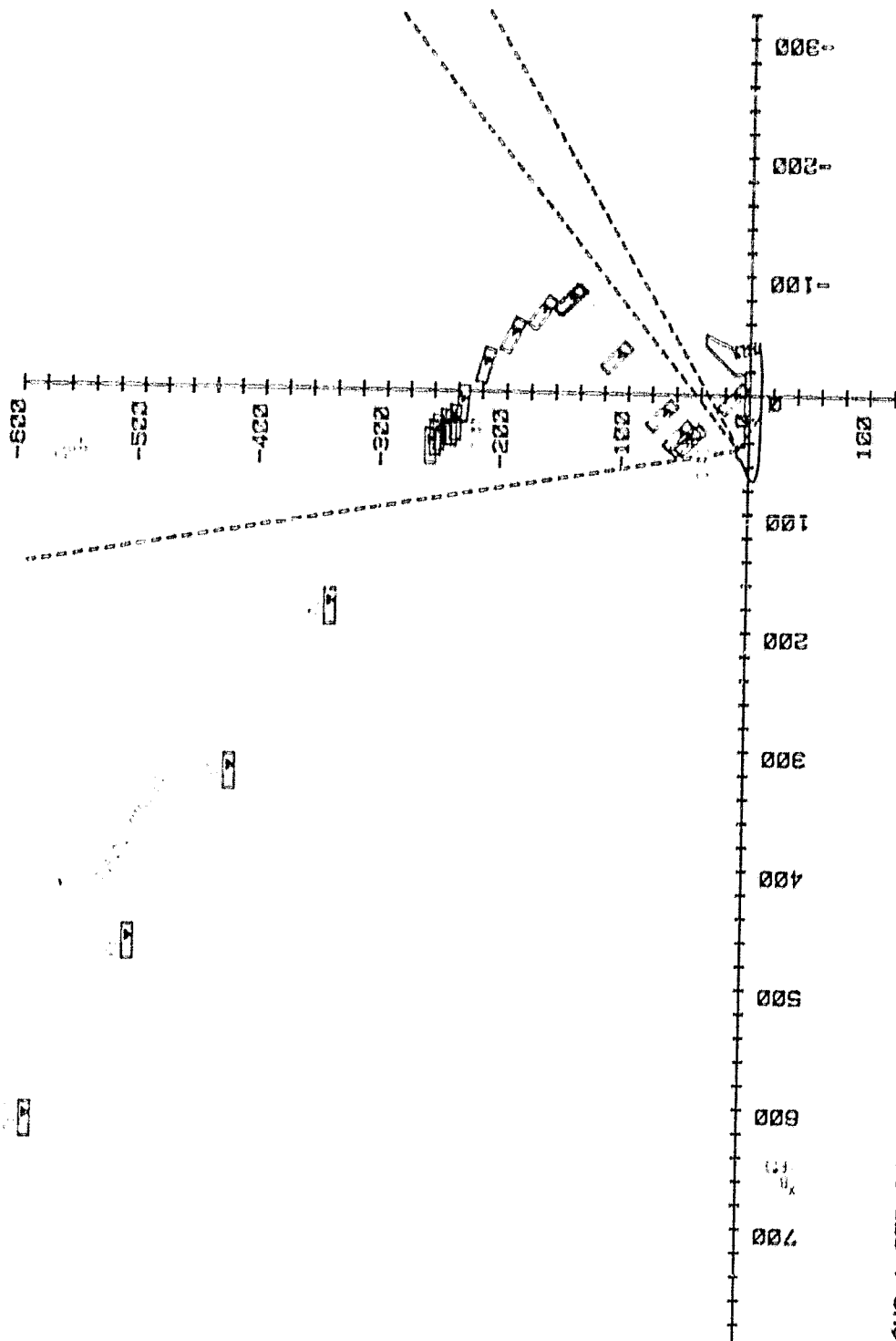
Figures 4-19 and 4-20 illustrate the SSUS-A positions relative to the +X RCS and -X RCS plume, respectively, during the 2 translational maneuvers. The maximum plume dynamic pressure experienced by the payload should be no greater than approximately 8×10^{-5} pounds per square foot during the 19-second total duration of the 2 RCS translational maneuvers.

Table 4-5. Orbiter/SSUS-A Initial Separation Sequence
(+45 Deg Rotation Between Ejection and
OMS Ignition)

1. Initiate 9-second +X RCS burn at 0:40 P.E.T.
2. Begin inertial hold immediately after +X burn.
3. Start +1.0 deg/sec pitch maneuver at 2:00 P.E.T.
4. Resume inertial hold upon reaching +45 deg angle of rotation from nominal ejection attitude (at approximately 2:47 P.E.T.).
5. Execute 10-second -X burn immediately after resuming inertial hold.
6. Trim to nominal OMS burn attitude, if necessary.
7. Begin 28-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	85
Aft Tanks	<u>77</u>
Total	162



SSUS-A STD SEP SEQ (PITCH +45 DEG)

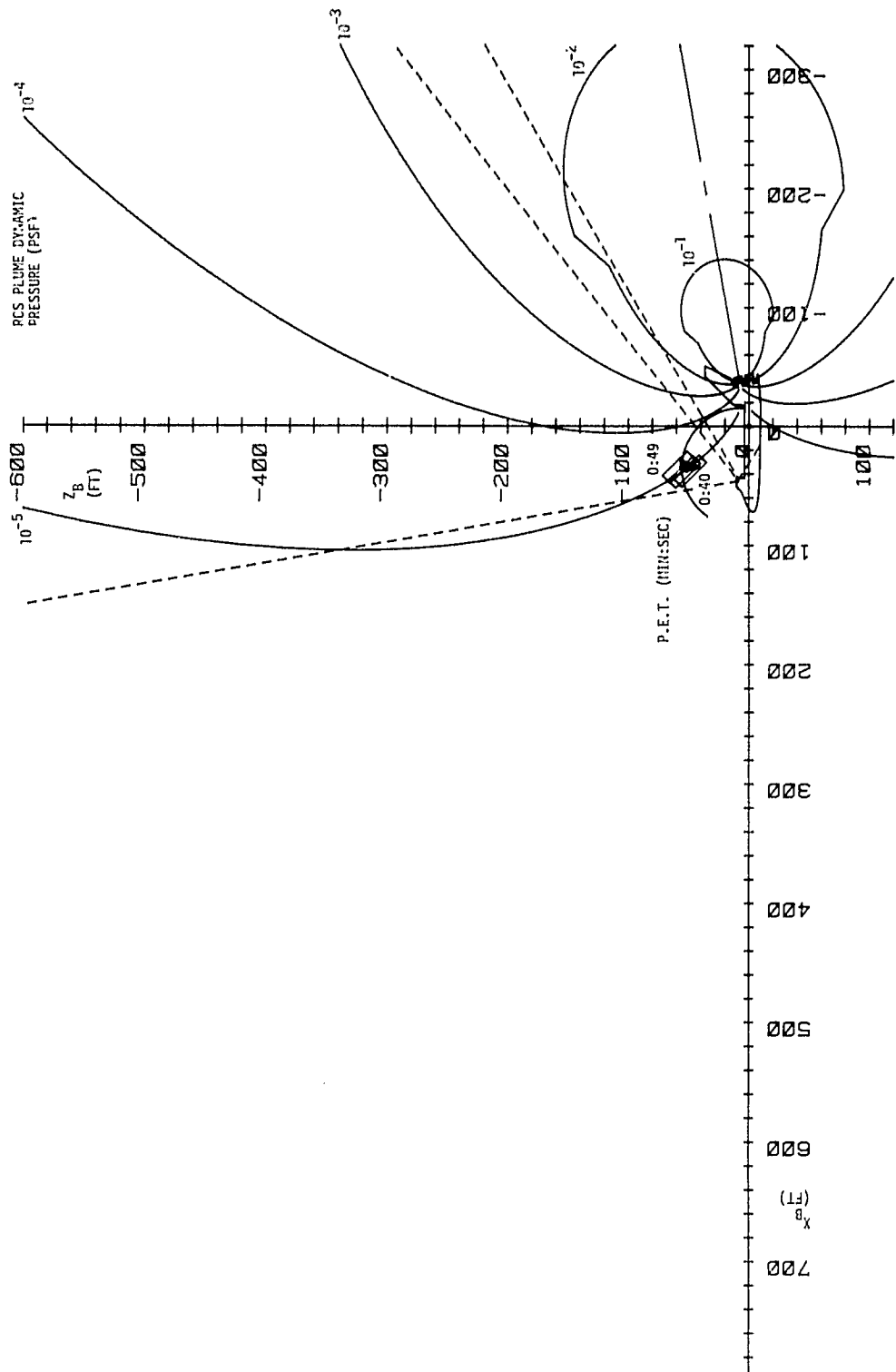


Figure 4-19. SSUS-A Positions Relative to +X RCS Plume (+45 Deg Rotation between Ejection and OMS Ignition)

SSUS-A STD SEP $\sim$ EQ (PITCH +45 DEG)

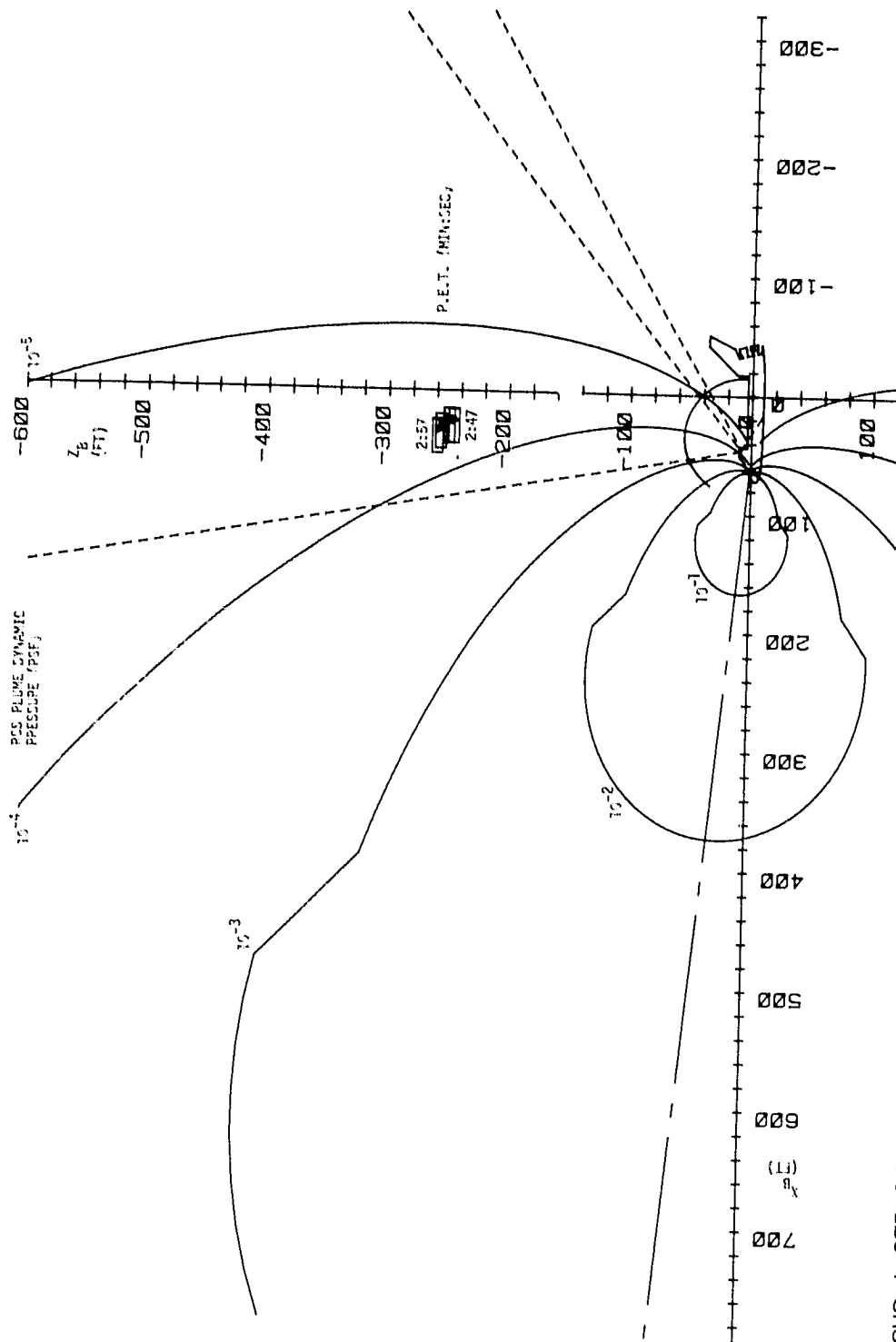


Figure 4-20. SSUS-A Positions Relative to X-Y ZCS Plane (+45 Deg Rotation Between Ejection and ZCS Ignition)

SSUS-A STD SEP SEQ (PITCH +45 DEG)

4.2.6 Orbiter Rotation Angle = +90 Degrees

Table 4-6 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-A initial separation sequence and Figure 4-21 presents an enlarged view of the early stage (to 6:00 P.E.T.). Examination of Figure 4-21 will show that the payload is visible through the pilot's overhead window almost continuously during the first 3 minutes and 5 seconds of the sequence.

The initiation of this sequence is similar to the +45 degree rotation case. The Orbiter's motion was uncontrolled from 0:00 P.E.T. to 0:40 P.E.T. at which time a 12-second +X RCS burn was performed. An inertial hold was established immediately after the burn. A 1.0 deg/sec pitch rate was initiated at 2:00 P.E.T. At 3:00 P.E.T. a 5-second -X RCS burn is performed while maintaining the pitch rate. The burn occurs at this time because less plume impingement would be experienced by the payload than if the burn were performed at the end of the pitch maneuver at approximately 3:32 P.E.T. This final RCS translation places the Orbiter approximately 1100 feet behind and 625 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 4-7).

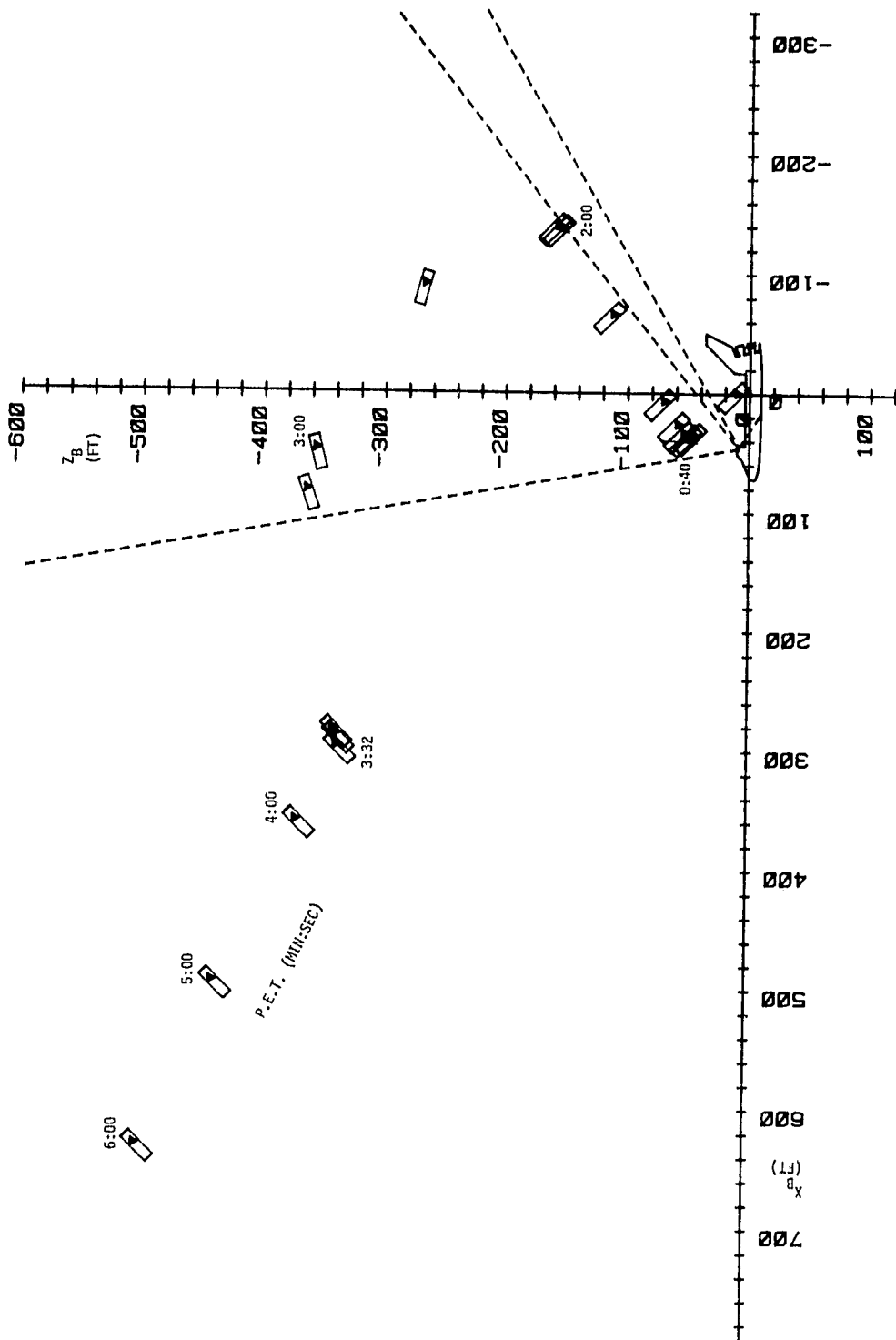
Figures 4-22 and 4-23 illustrate the SSUS-A positions relative to the +X RCS and -X RCS plume, respectively, during the 2 translational maneuvers. The maximum plume dynamic pressure experienced by the payload should be no greater than approximately 9×10^{-5} pounds per square foot during the 17-second total duration of the 2 RCS translational maneuvers.

Table 4-6. Orbiter/SSUS-A Initial Separation Sequence
(+90 Deg Rotation Between Ejection and OMS
Ignition)

1. Initiate 12-second +X RCS burn at 0:40 P.E.T.
2. Begin inertial hold immediately after +X burn.
3. Start +1.0 deg/sec pitch maneuver at 2:00 P.E.T.
4. At 3:00 P.E.T. burn -X for 5 seconds while maintaining +1.0 deg/sec pitch rate.
5. Resume inertial hold upon reaching +90 deg angle of pitch from nominal ejection attitude (at approximately 3:32 P.E.T.).
6. Trim to nominal OMS burn attitude, if necessary.
7. Begin 28-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	50
Aft Tanks	96
Total	146



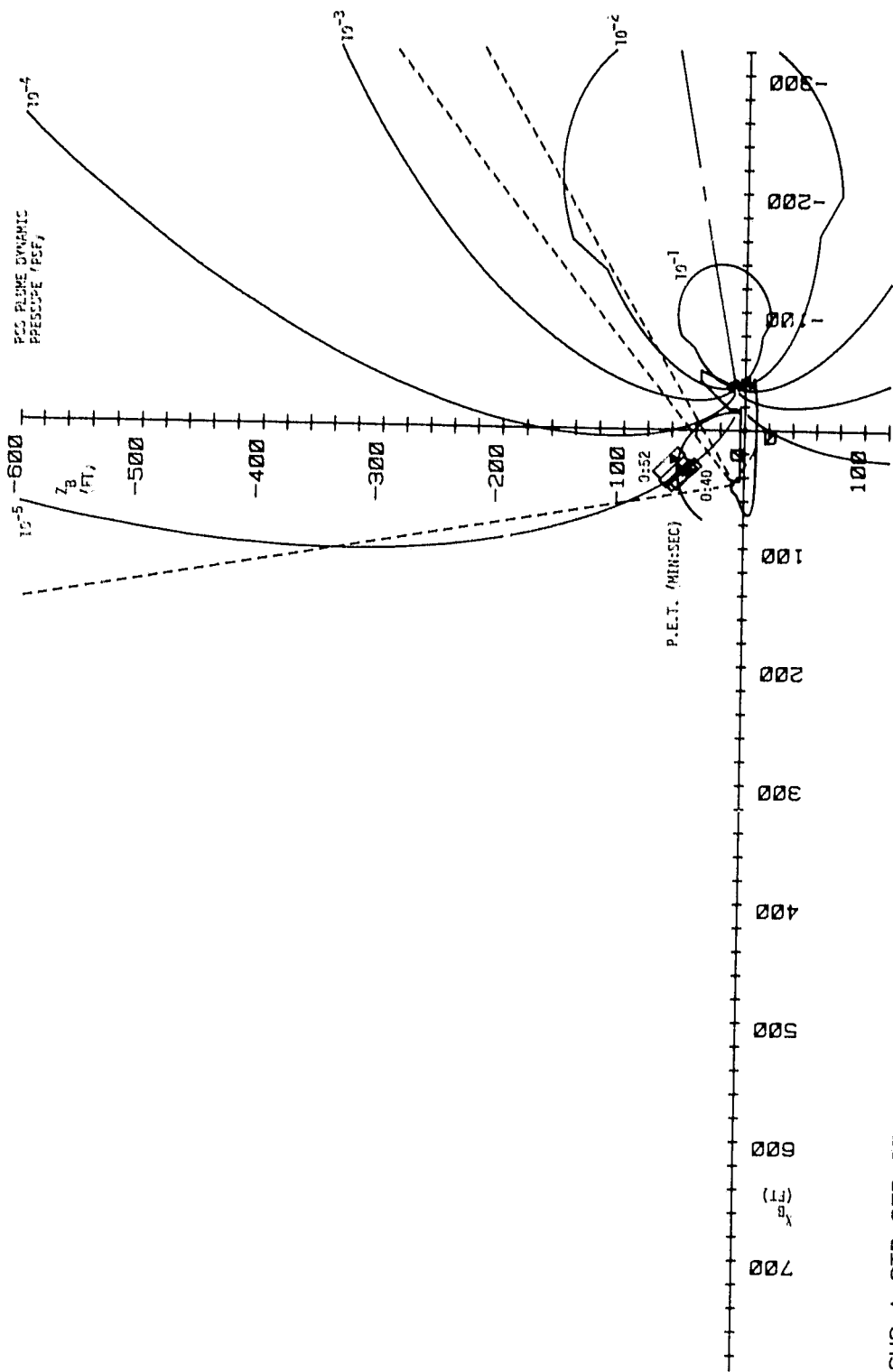


Figure 4-22. SSUS-A Positions Relative to X ECC Plume 1/50 Deg Rotation Between Ejection and JWS Ignition

SSUS-A STD SEP SEC (PITCH +90 DEG)

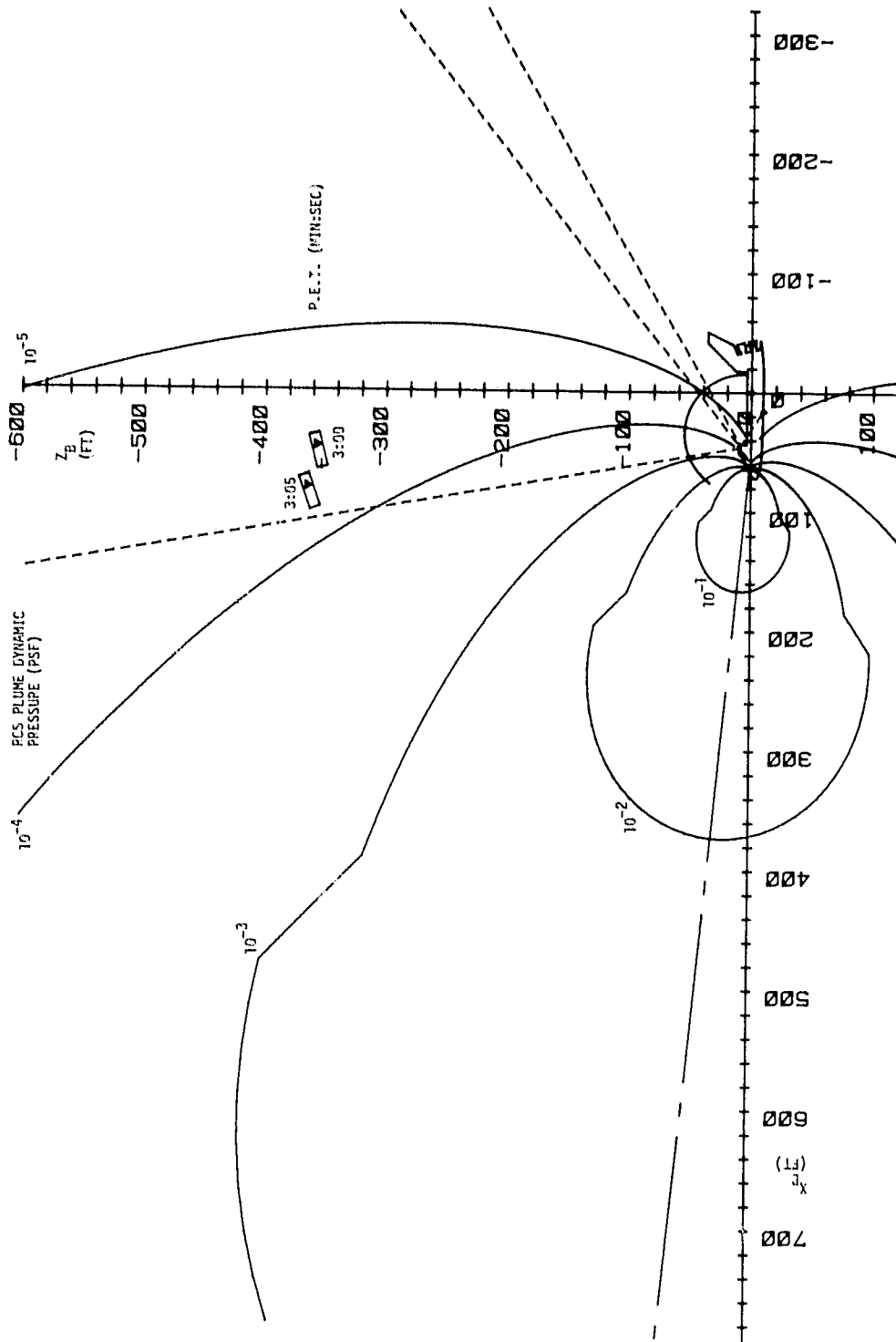


Figure 4-23. SSUS-A Positions Relative to -X RCS Plume (+90 Deg Rotation Between Ejection and OMS Ignition)

SSUS-A STD SEP SEQ (PITCH +90 DEG)

4.2.7 Orbiter Rotation Angle = +135 Degrees

Table 4-7 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-A initial separation sequence and Figure 4-24 presents an enlarged view of the early stage (to 6:00 P.E.T.). Examination of Figure 4-24 will show that the payload is visible through the pilot's overhead window almost continuously for the first 2 minutes of the sequence.

The initiation of this sequence is different than the previous positive pitch (+45 and +90 degree) rotation cases in that the pitch rate is initiated immediately after the first +X RCS translation of 12 seconds at 0:40 P.E.T. At 2:00 P.E.T., a 6-second +X RCS burn was performed to prevent the payload from being below the Orbiter ($Z_B \rightarrow +$) at the completion of the 135-degree pitch rotation (approximately 3:09 P.E.T.). This burn places the Orbiter approximately 900 feet behind and 850 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 4-8).

Figure 4-25 illustrates the SSUS-A positions relative to the +X RCS plume during the 2 translational maneuvers. The maximum plume dynamic pressure experienced by the payload should be no greater than approximately 5×10^{-5} pounds per square foot during the 18-second total duration of the 2 RCS translational maneuvers.

Table 4-7. Orbiter/SSUS-A Initial Separation Sequence
(+135 Deg Rotation Between Ejection and OMS
Ignition)

1. Initiate 12-second +X RCS burn at 0:40 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at +1.0 deg/sec immediately after +X burn.
3. At 2:00 P.E.T., burn +X for 6 seconds while maintaining +1.0 deg/sec pitch rate.
4. Begin inertial hold upon reaching +135 deg angle of pitch from nominal ejection attitude (at approximately 3:09 P.E.T.).
5. Trim to nominal OMS burn attitude, if necessary.
6. Begin 28-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	15
Aft Tanks	134
Total	149

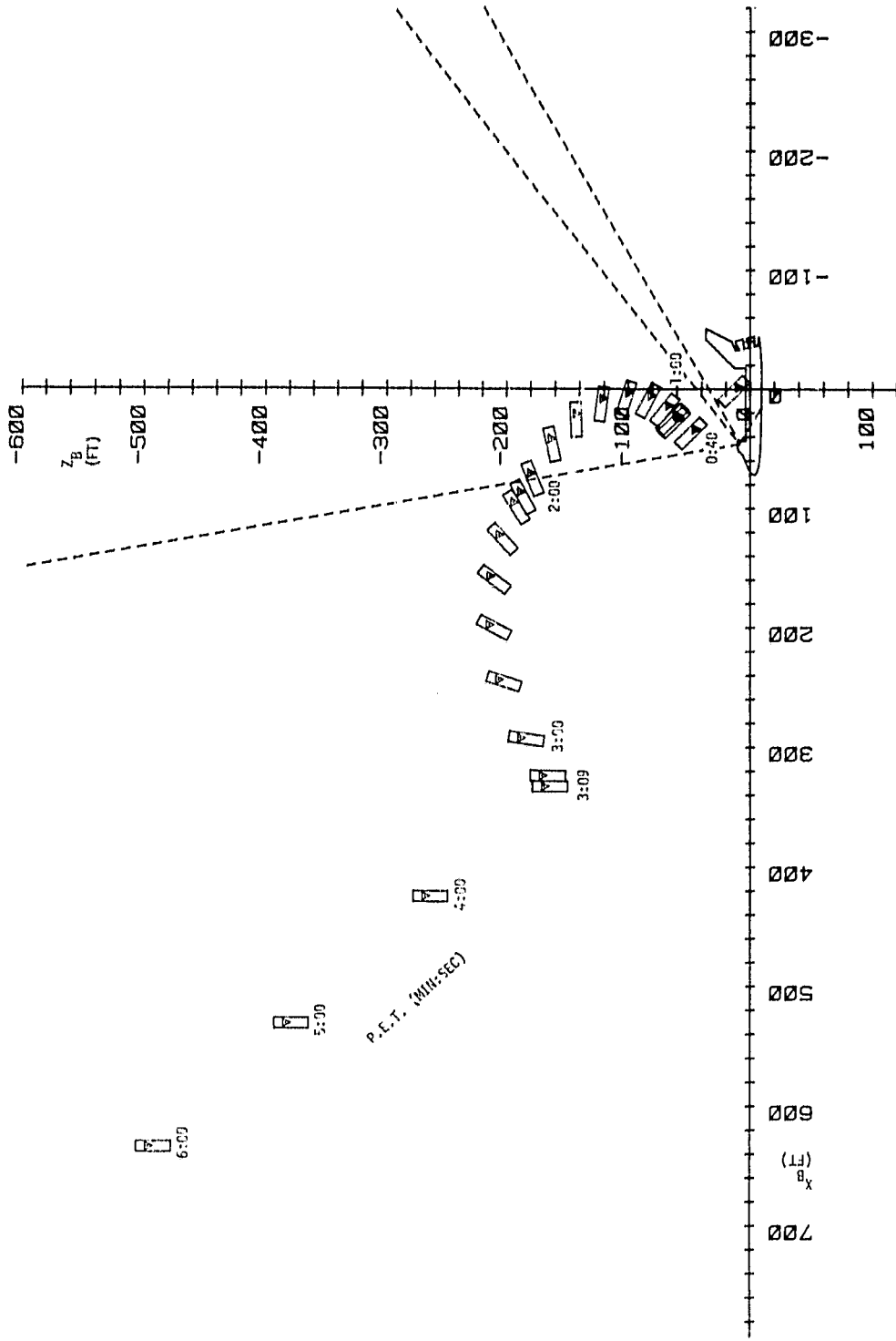


Figure 4-24. Enlarged View of Orbiter/SSUS-A Initial Separation Phase (+135 Deg Rotation Between Ejection and OMS Ignition)

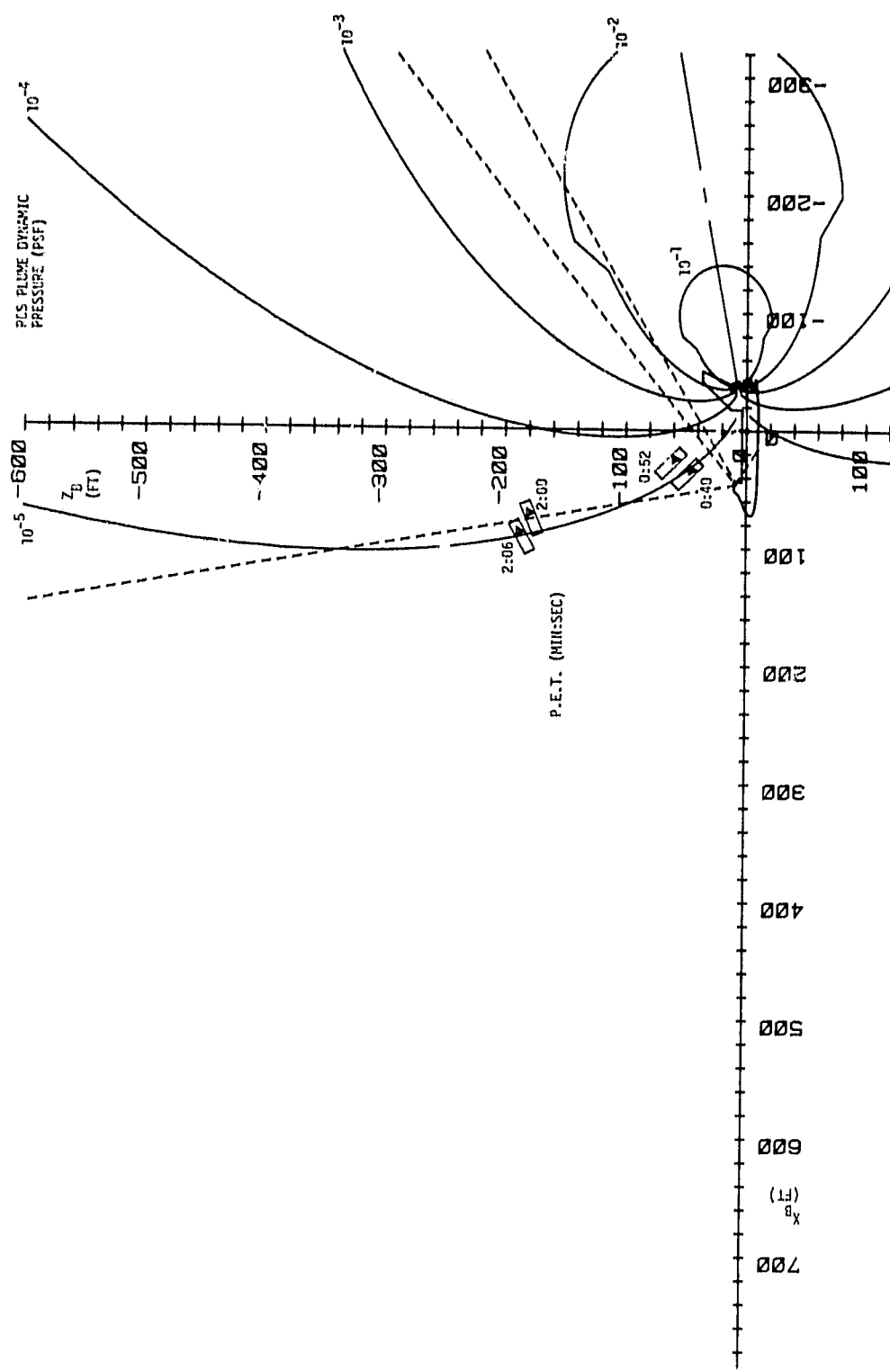


Figure 4-25. SSUS-A Positions Relative to +X RCS Plume (+135 Deg Rotation Between Ejection and OMS Ignition)

SSUS-A STD SEP SEQ (PITCH +135 DEG)

4.2.8 Orbiter Rotation Angle = +180 Degrees

Table 4-8 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-A initial separation sequence and Figure 4-26 presents an enlarged view of the early stage (to 5:00 P.E.T.). Examination of Figure 4-26 will show that payload is visible through the pilot's overhead window almost continuously for the first 1 minute and 50 seconds of the sequence.

The initiation of this sequence is similar to the +135 degree rotation case. The Orbiter's motion was uncontrolled from 0:00 P.E.T. to 0:40 P.E.T. at which time a 10-second +X RCS burn was performed. Immediately after the burn, a pitch rate of 1.0 deg/sec was established. At 2:00 P.E.T., a 12-second +X RCS burn was performed to prevent the payload from being below the Orbiter ($Z_B \rightarrow +$) at the completion of the 180-degree pitch rotation (approximately 3:52 P.E.T.). This burn places the Orbiter approximately 1350 feet behind and 600 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 4-9). At 9:28 P.E.T., the end of the OMS burn, the Orbiter is still approximately 650 feet behind the payload instead of approximately 300 feet as in the other cases. This added displacement is caused by the large magnitude (12-seconds) of the second +X RCS burn required to prevent the payload from going below the Orbiter.

Figure 4-27 illustrates the SSUS-A positions relative to the +X RCS plume during the 2 translational maneuvers. The maximum plume dynamic pressure experienced by the payload should be no greater than approximately 3×10^{-5} pounds per square foot during the 22-second total duration of the 2 RCS translational maneuvers.

Table 4-8. Orbiter/SSUS-A Initial Separation Sequence
(+180 Deg Rotation Between Ejection and
OMS Ignition)

1. Initiate 10-second +X RCS burn at 0:40 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at +1.0 deg/sec immediately after +X burn.
3. At 2:00 P.E.T., burn +X for 12 seconds while maintaining +1.0 deg/sec pitch rate.
4. Begin inertial hold upon reaching +180 deg angle of pitch from nominal ejection attitude (at approximately 3:52 P.E.T.).
5. Trim to nominal OMS burn attitude, if necessary.
6. Begin 28-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	15
Aft Tanks	158
Total	173

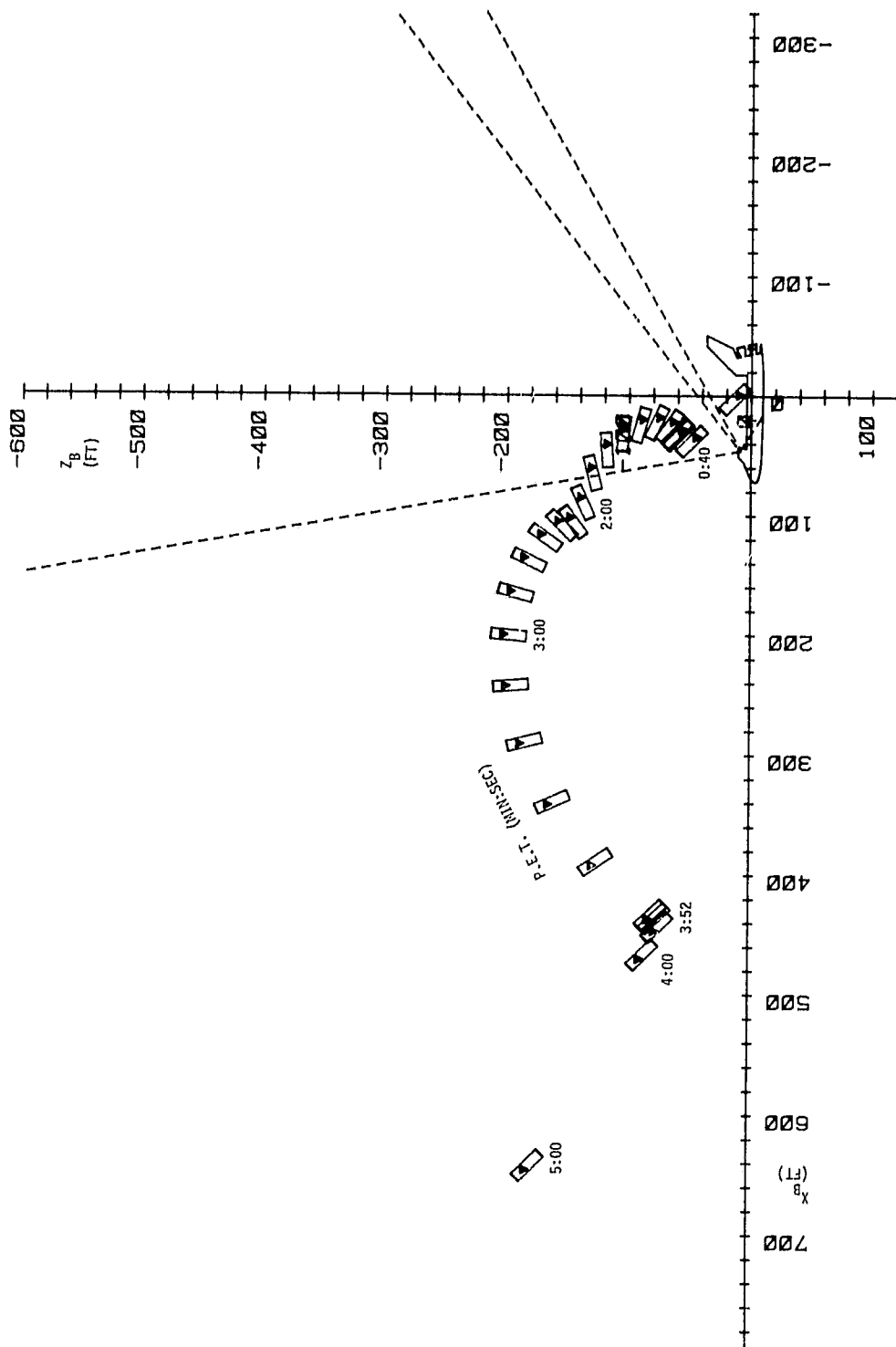
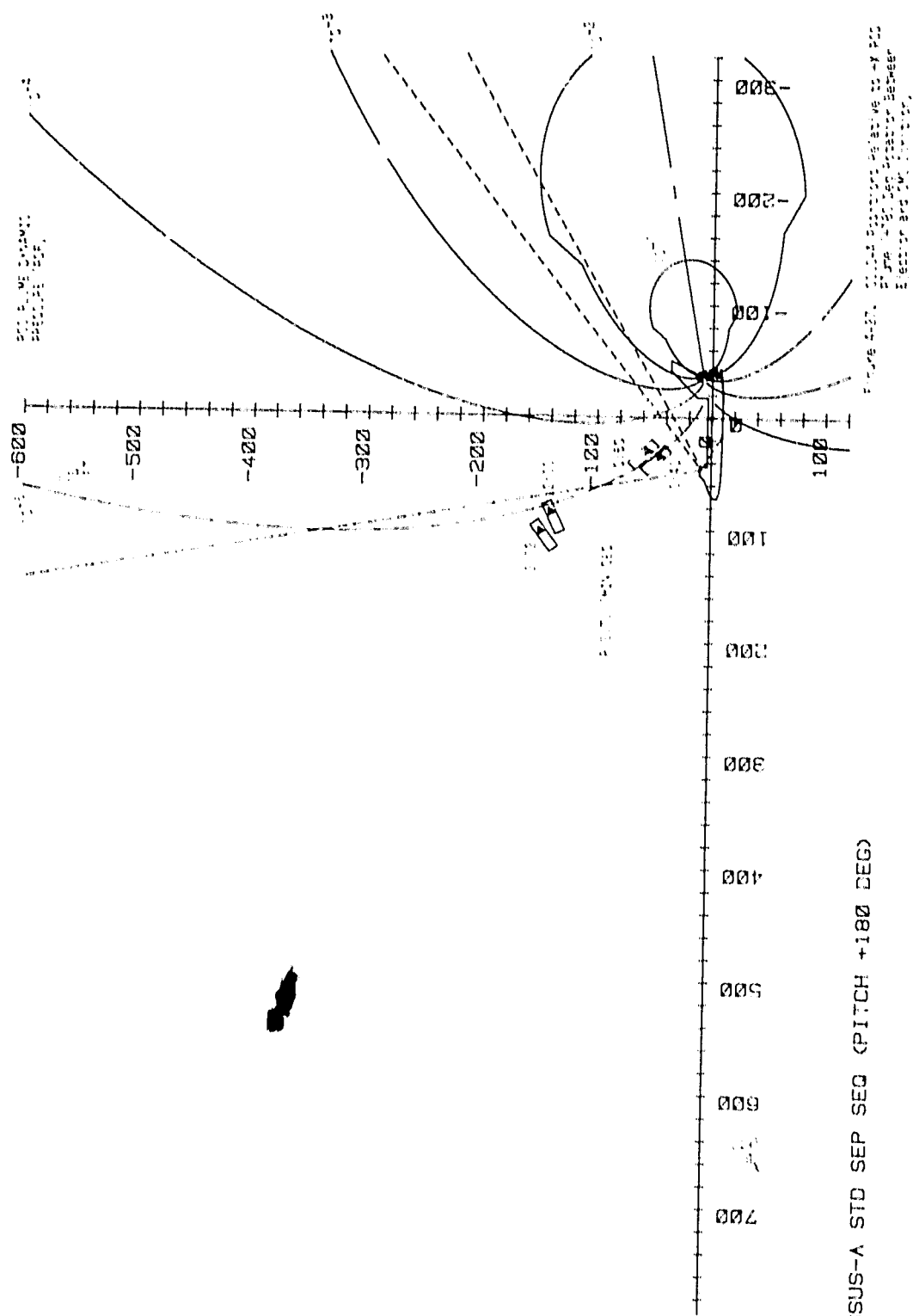


Figure 4-26. Enlarged View of Orbiter/SSUS-A Initial Separation Phase (+180 Deg Rotation Between Ejection and OMS Ignition)

SSUS-A STD SEP SEQ (PITCH +180 DEG)



5.0 DEFINITION OF STANDARD ORBITER/SSUS-D SEPARATION SEQUENCES

5.1 ASSUMPTIONS AND DESIGN REQUIREMENTS

1. For trajectory calculation purposes, the mass properties of the Orbiter were assumed to be as listed in Table 3-1 immediately after ejection of the payload from the cargo bay.
2. The payload (SSUS-D+satellite) was modeled as a right circular cylinder 80 inches in diameter and 120 inches in length.
3. The payload will be ejected by springs from a cradle in the cargo bay, with a relative velocity increment of 2.90 feet per second in the direction of the SSUS-D +X body axis (which is aligned with the SSUS-D SRM thrust vector).
4. At the time of ejection, the payload will be positioned in the cradle so that the SSUS-D +X axis makes an angle of 90 degrees (measured in the Orbiter's plane of symmetry) with the Orbiter +X body axis.
5. The Orbiter attitude is to be inertially stabilized as nearly as possible with all RCS jets inactive at the time of payload ejection. Orbiter RCS jets are to remain inactive until the payload has cleared all Orbiter-fixed structure by at least TBD feet.
6. The line of action of the nominal spring-ejection impulse is assumed to pass through the payload CG and through a point approximately 147 inches behind the Orbiter CG, producing no angular perturbations of the payload, and an Orbiter pitch rate of +0.0611 degrees per second.
7. The payload must be continuously visible to the Orbiter flight crew after ejection until such time as positive separation has been confirmed during or immediately after the Orbiter's final separation maneuver (i.e., the OMS burn).
8. Plume impingement on the payload from Orbiter thrusters is to be minimized.

5.2 MANEUVER SEQUENCE DESCRIPTIONS

Eight discrete Orbiter/SSUS-D initial separation phase maneuver sequences have been designed. In Section 5.2.1 - 5.2.3 each sequence will be described by a table listing the schedule of events and by figures depicting the motion of the payload relative to the Orbiter. Three types of relative motion plots are shown to clarify each sequence: (1) an overall view of the Orbiter/SSUS-D initial separation phase including OMS plume contours, (2) an enlarged view of the early stage of the sequence, and (3) the SSUS-D position relative to the RCS plume contours during RCS translational maneuvers. All of the data presented in this section were obtained by use of the HP-9825A HFRMP, Version 03M (References 1-3).

All rotational and translational maneuvers of the Orbiter were simulated with finite-thrust models. The PRCS was used for Orbiter attitude control. To minimize plume impingement on the payload, the +Z PRCS jets (i.e., those which expel propellant directly upward relative to the body-fixed frame) were inhibited. Attitude-control deadbands were set at 0.5 degrees per axis.

Figure 5-1 presents the nominal RCS propellant consumption and range of applicability for each of the eight standard Orbiter/SSUS-D separation sequences. Figures 5-2 through 5-9 present an overall view of the Orbiter/SSUS-D initial separation phase depicting the motion of the payload relative to the Orbiter for each sequence. In each sequence, a 15-second OMS maneuver is assumed to begin 9 minutes after ejection (P.E.T. = 9:00). The plume contours for the OMS are overlaid on the relative motion plots. Examination of the figures will show that OMS plume impingement on the payload should be completely negligible even allowing for RCS maneuver execution errors that might produce a trajectory dispersion of 300 feet or so at the time of OMS ignition.

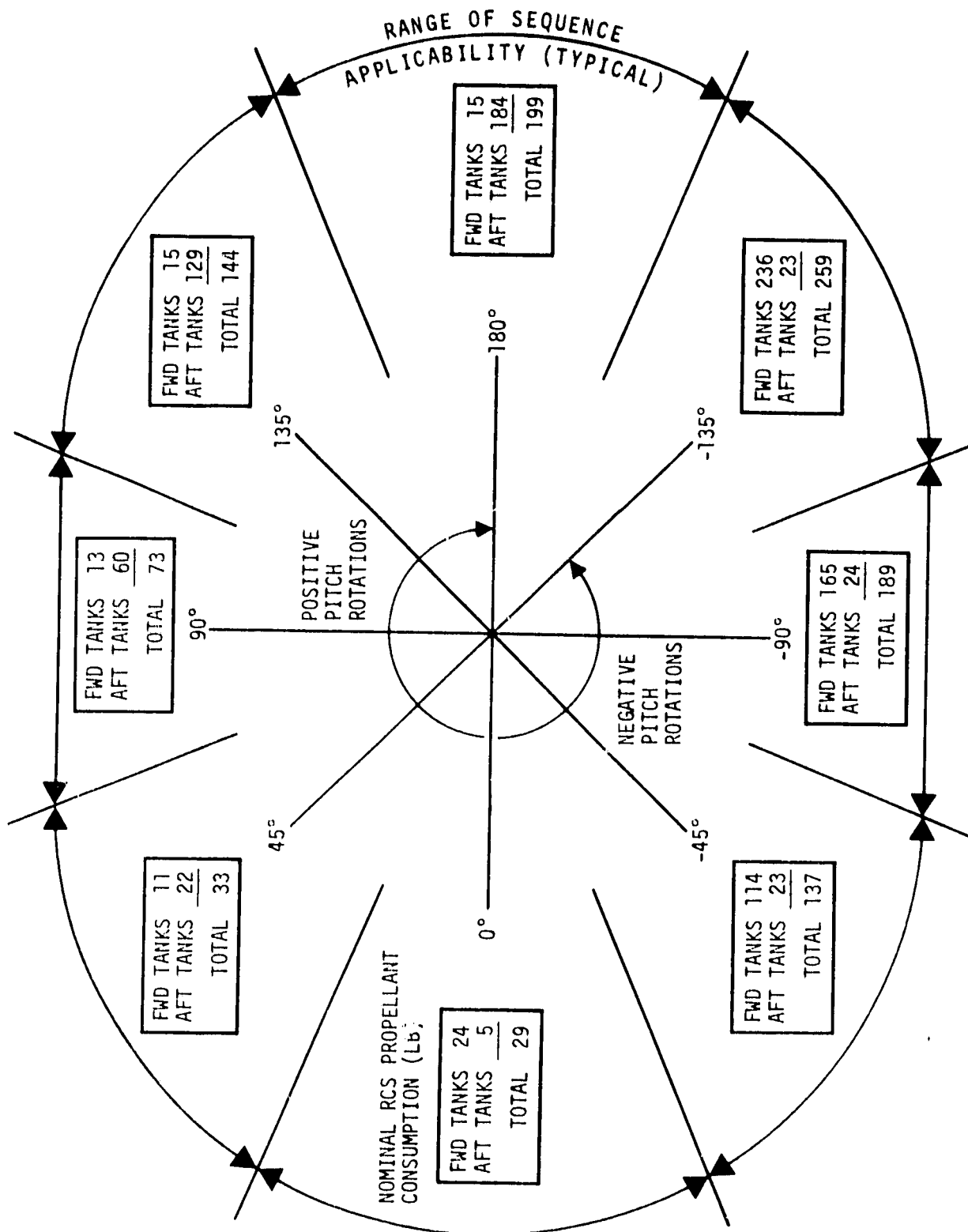


Figure 5-1. Nominal RCS Propellant Consumption for Eight Standard Orbiter/SSUS-D Separation Sequences at Intervals of 45 Degrees in the Angle of Rotation Between Ejection and OMS Ignition

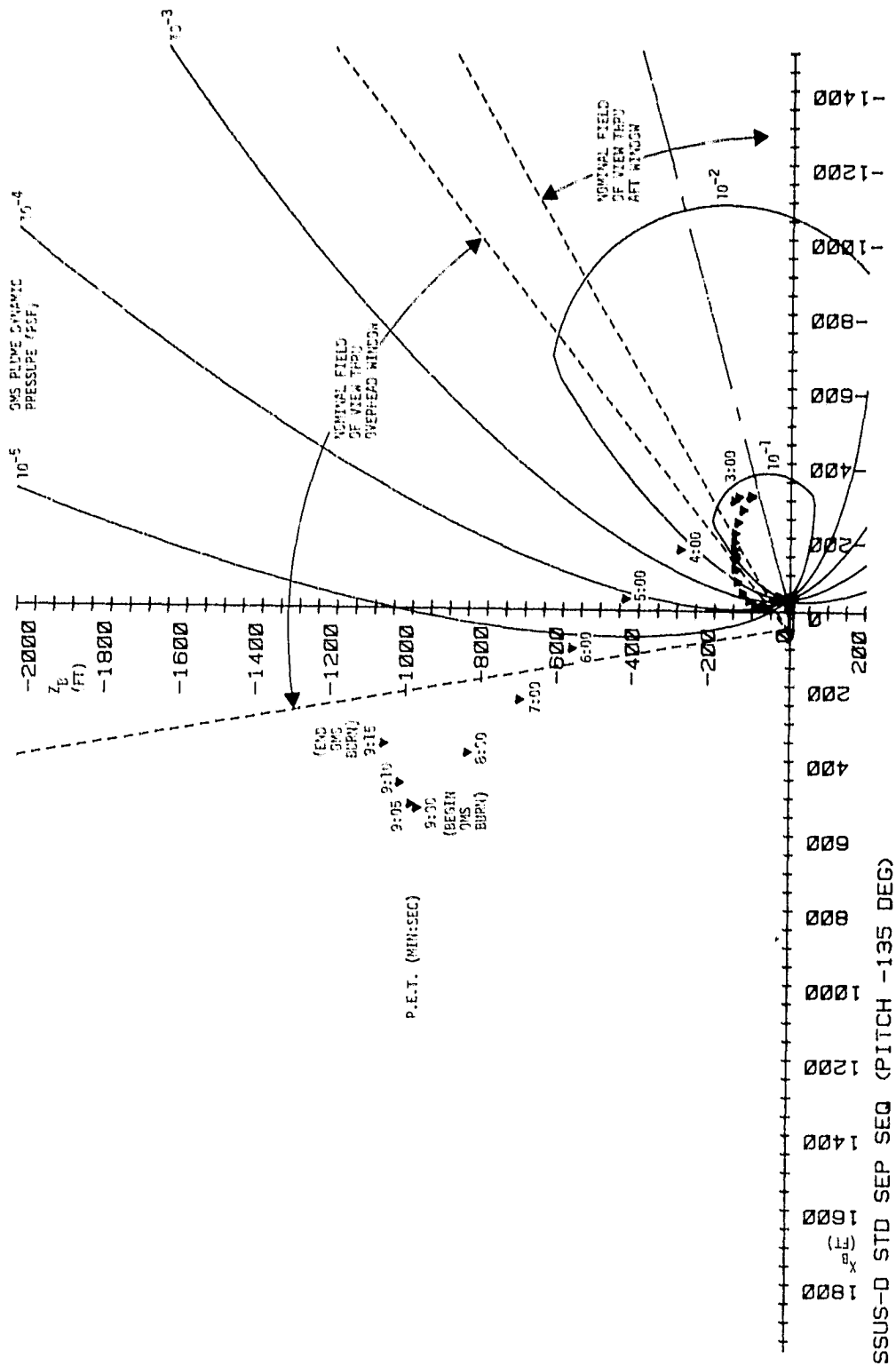


Figure 5-2. Overall View of Orbiter/SSUS-D Initial Separation Phase (-135 Deg Rotation Between Ejection and GMS Ignition)

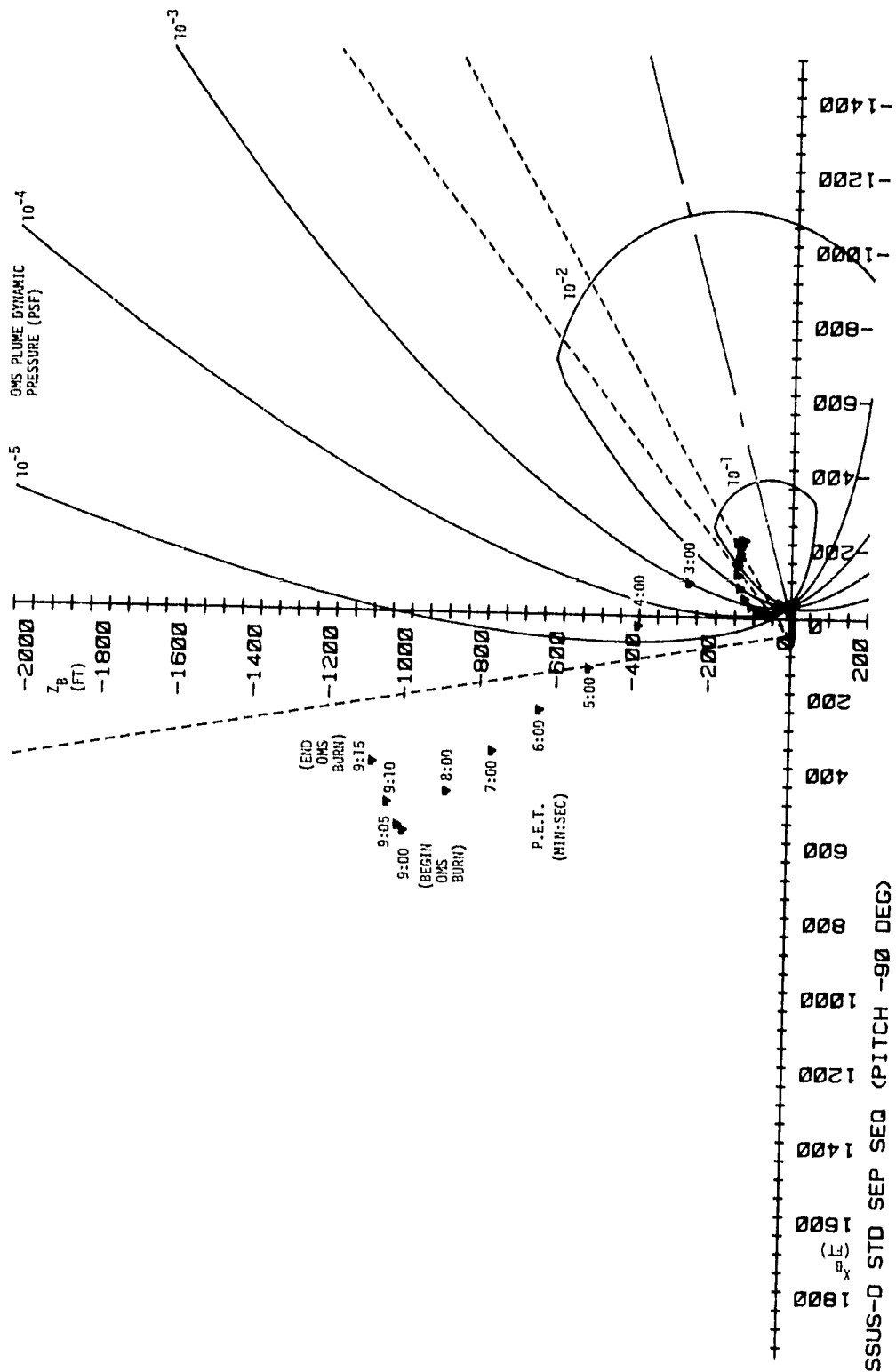


Figure 5-3. Overall View of Orbiter/SSUS-D Initial Separation Phase (-90 Deg Rotation Between Ejection and OMS Ignition)

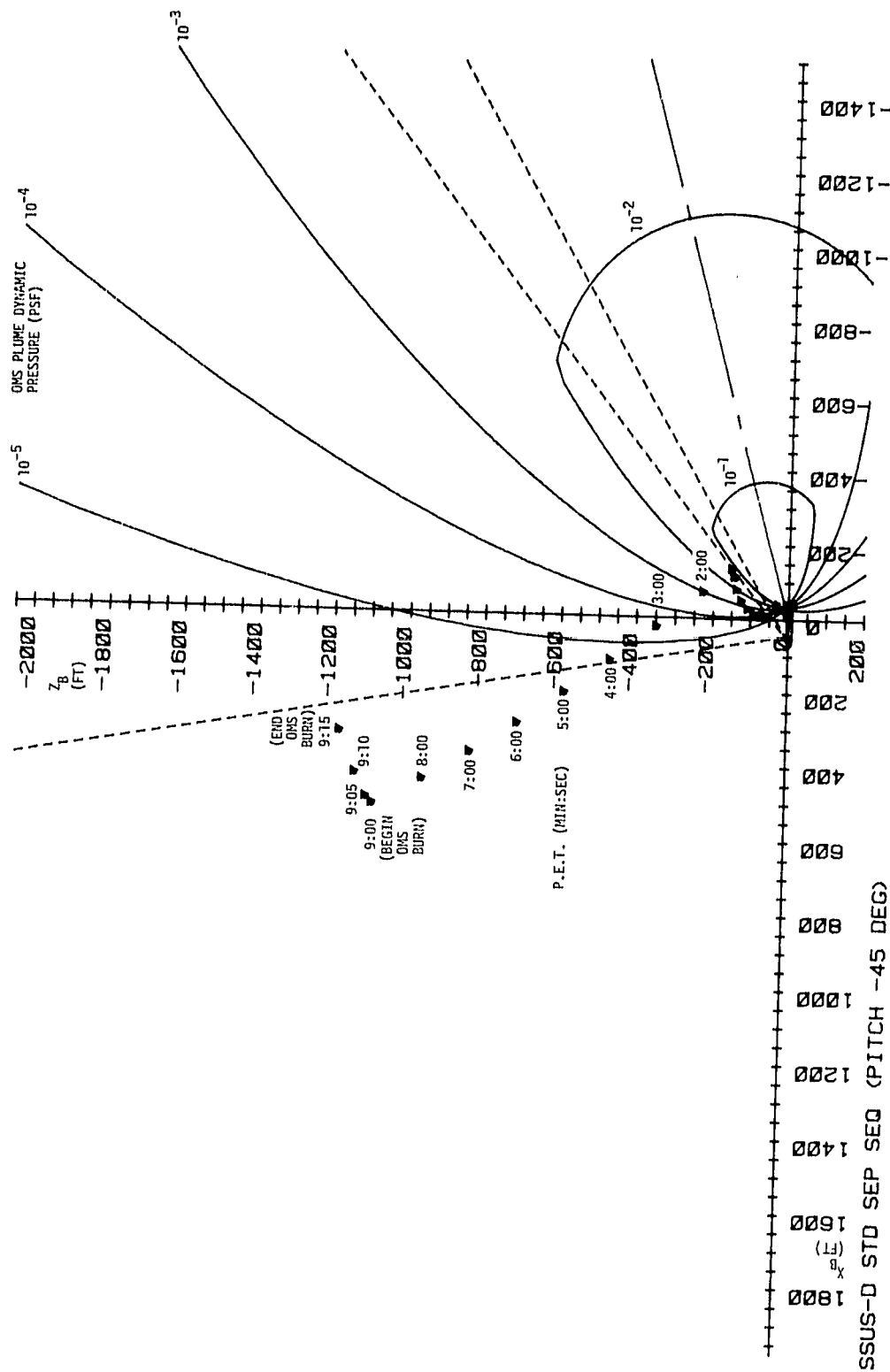


Figure 5-4. Overall View of Orbiter/SSUS-D Initial Separation Phase (-45 Deg Rotation Between Ejection and OMS Ignition)

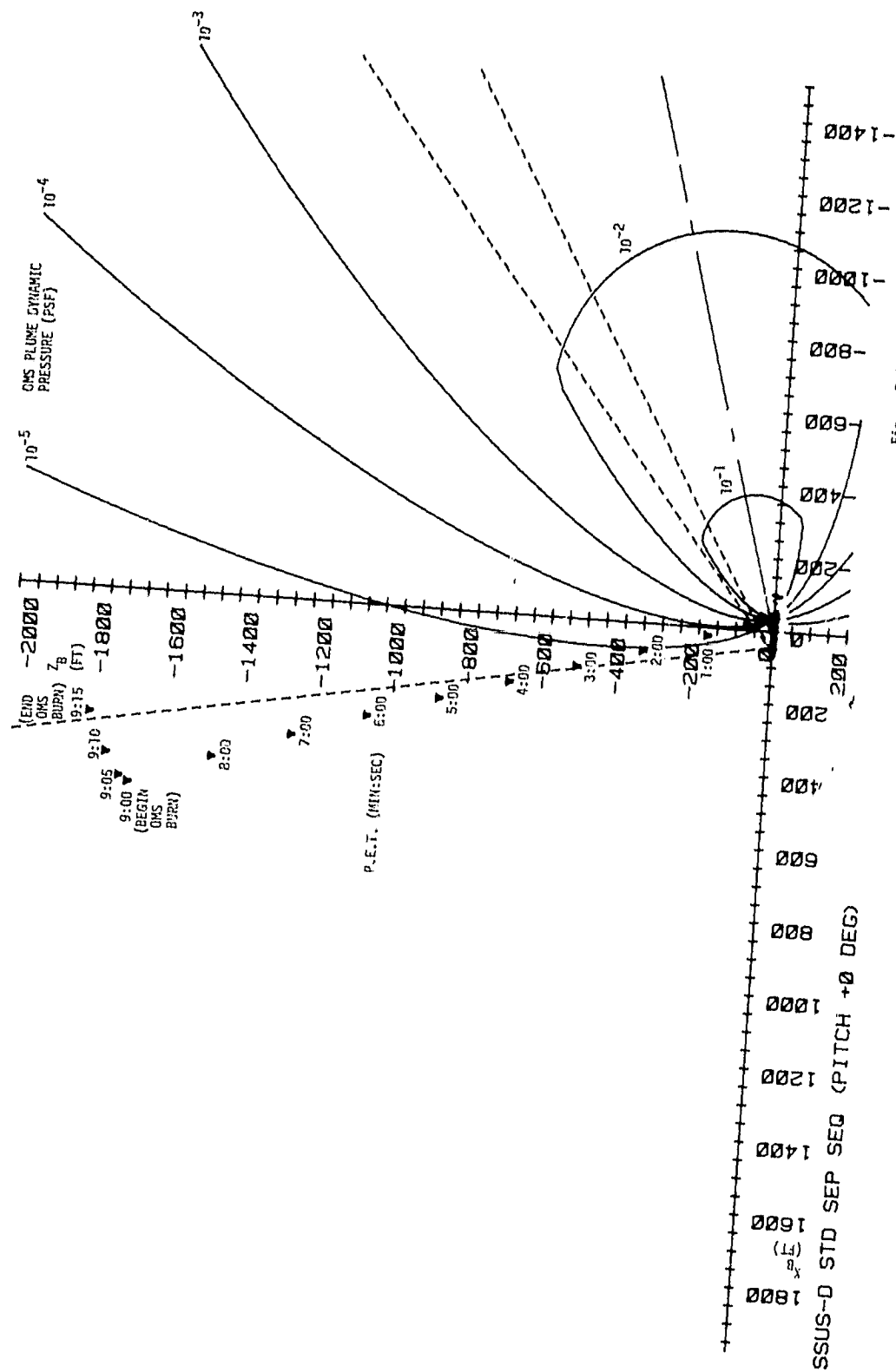


Figure 5-5. Overall View of Orbiter/SSUS-3 Initial Separation Phase (0 Deg Rotation Between Ejection and QMS Ignition)

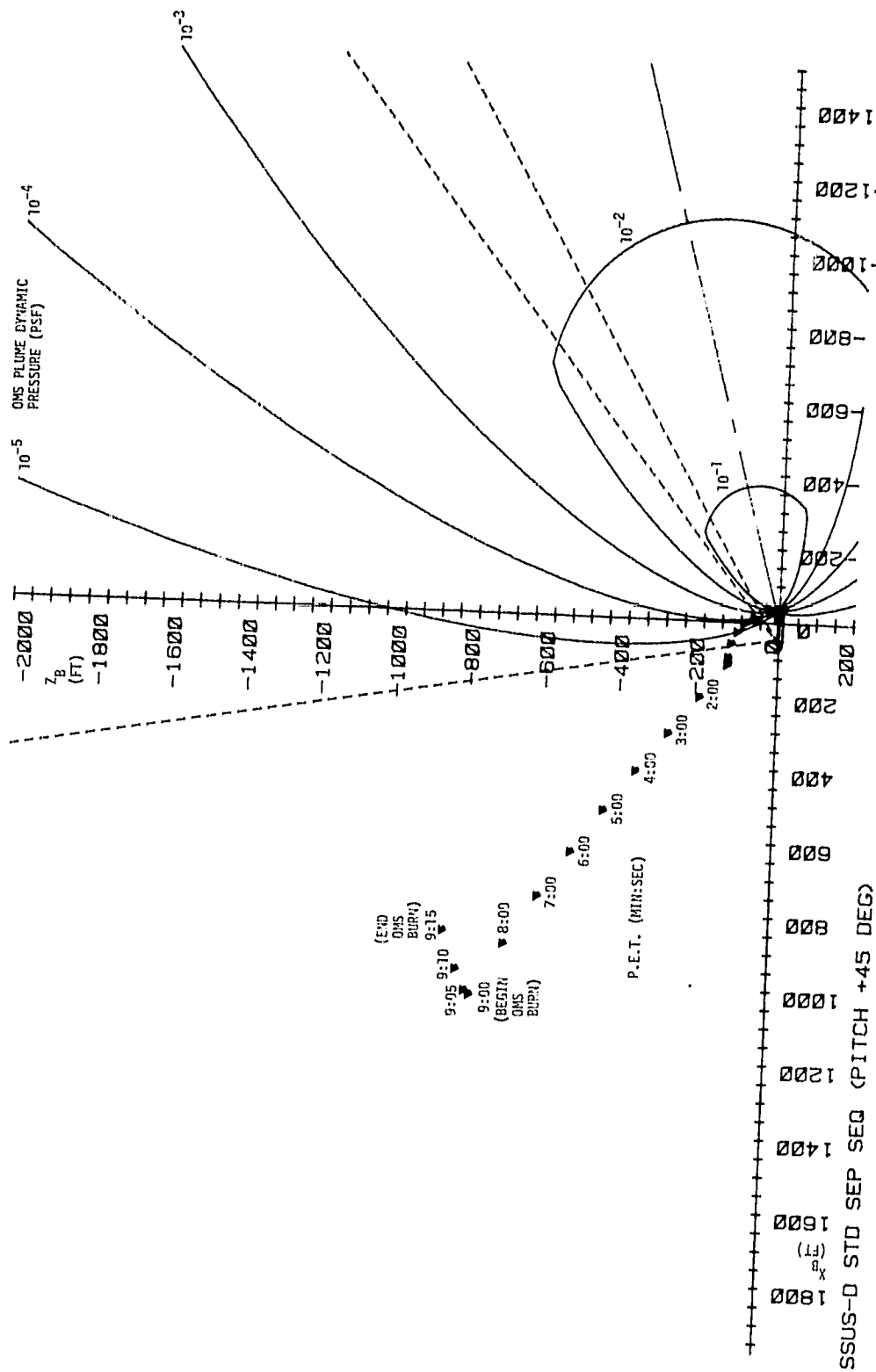


Figure 5-6. Overall View of Orbiter/SSUS-D Initial Separation Phase (+45 Deg Rotation Between Ejection and OMS Ignition)

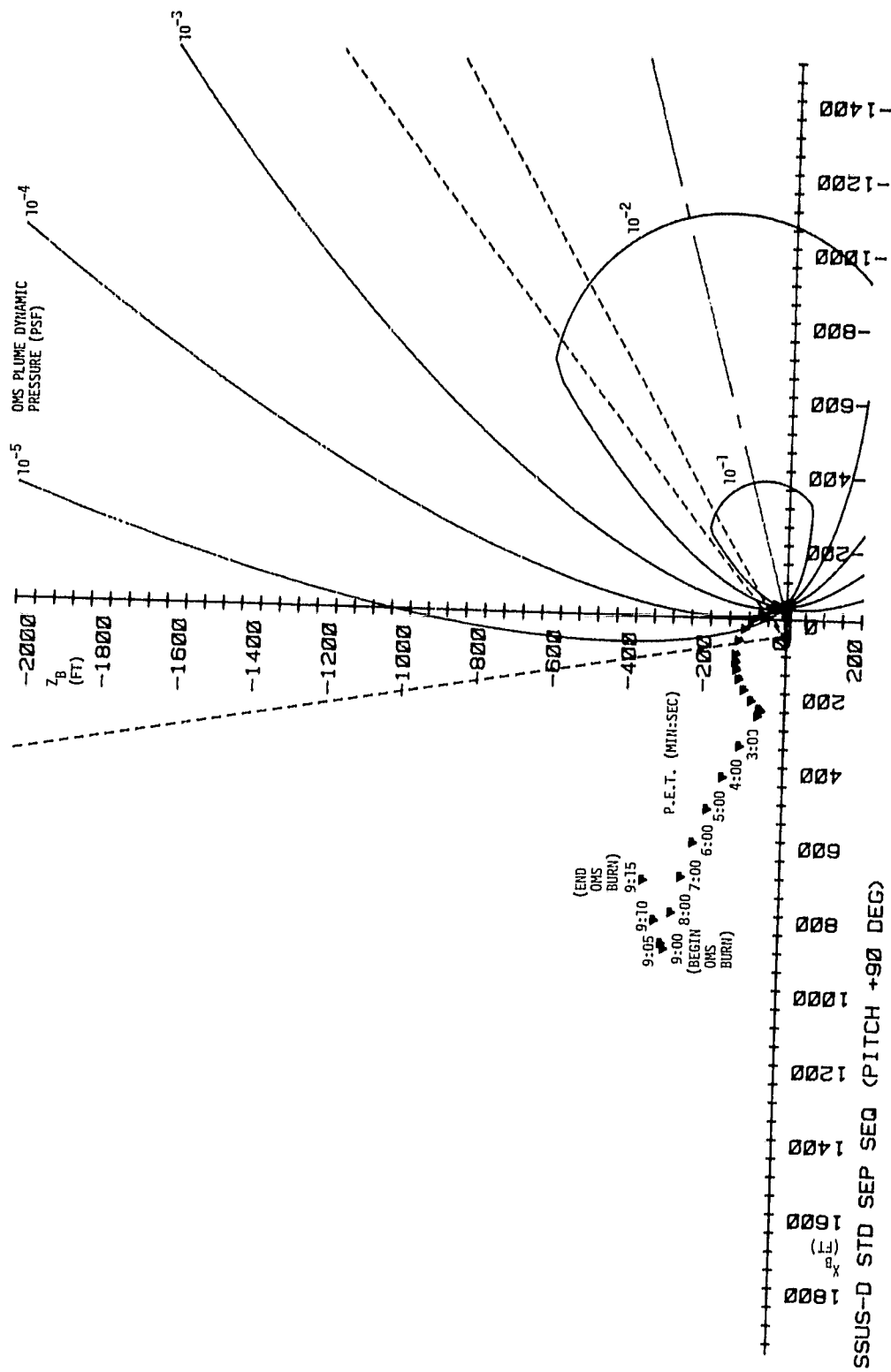


Figure 5-7. Overall View of Orbiter/SSUS-D Initial Separation Phase (+90 Deg Rotation Between Ejection and OMS Ignition)

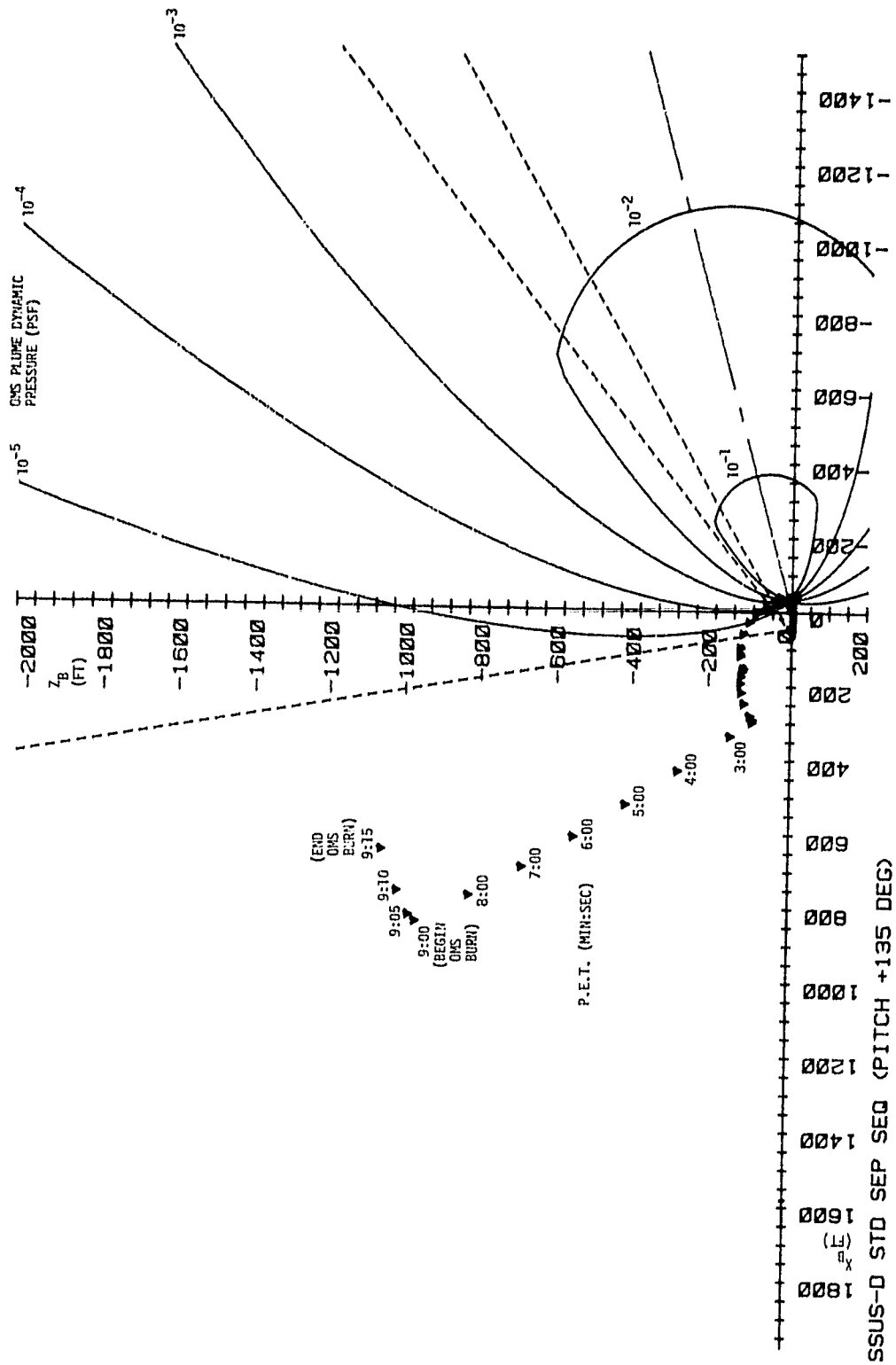


Figure 5-8. Overall View of Orbiter/SSUS-D Initial Separation Phase (+135 Deg Rotation Between Ejection and OMS Ignition)

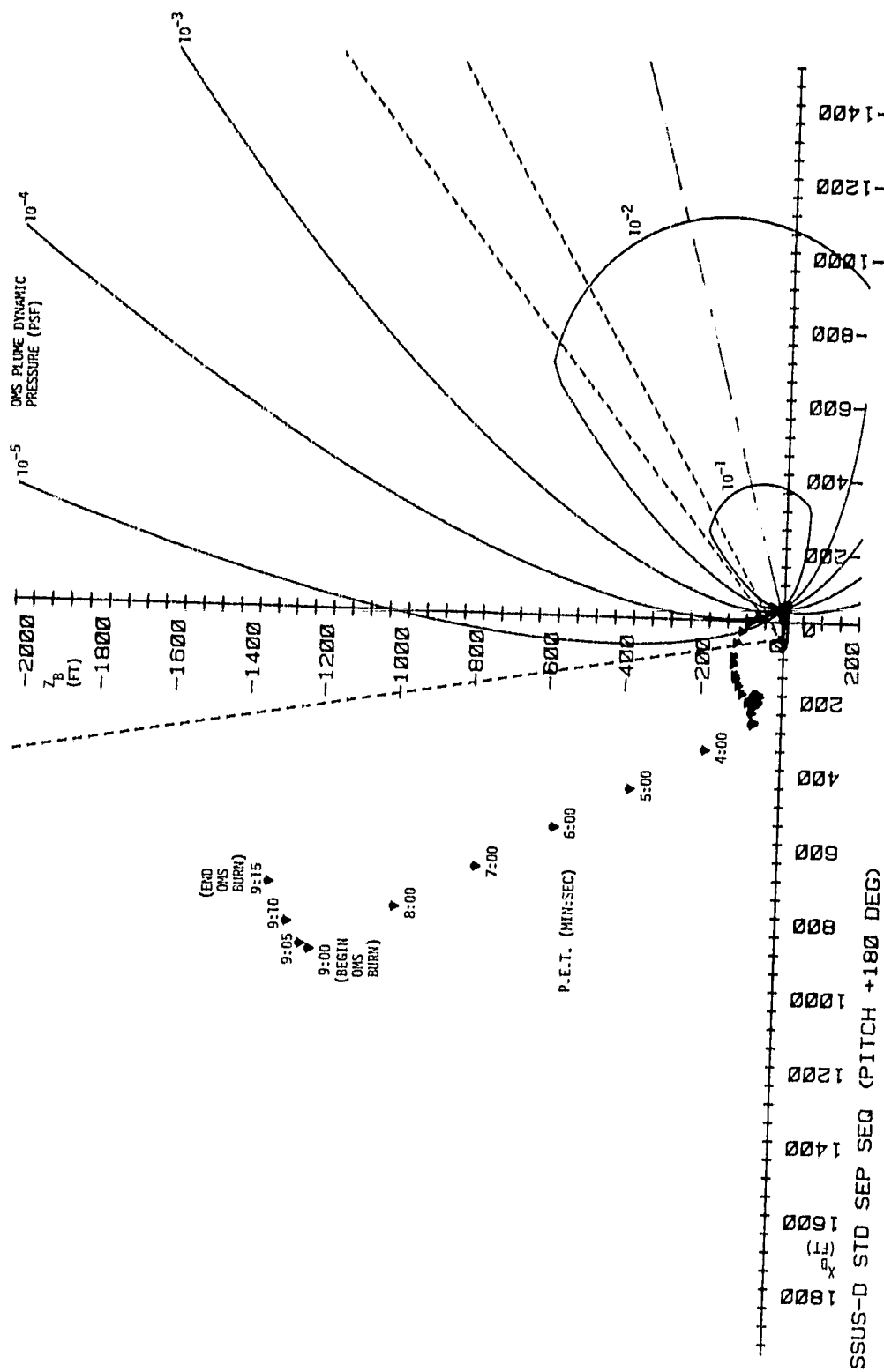


Figure 5-9. Overall View of Orbiter/SSUS-D Initial Separation Phase (+180 Deg Rotation Between Ejection and OMS Ignition)

5.2.1 Orbiter Rotation Angle = -135 Degrees

Table 5-1 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-D initial separation sequence and Figure 5-10 presents an enlarged view of the early stage (to 6:00 P.E.T.). Examination of Figure 5-10 will show that the payload is visible through the pilot's overhead window until 1:10 P.E.T., is visible through the aft window from 2:00 P.E.T. to 3:10 P.E.T., and again through the overhead window 3:30 P.E.T. to 6:00 P.E.T.

The Orbiter's motion was uncontrolled from 0:00 P.E.T. until 0:20 P.E.T., at which time a 3-second -X RCS burn was initiated. Examination of Figure 5-10 will show that the payload is clear of the Orbiter structure at the time of the RCS burn. Immediately after the burn, a pitch rate of -1.0 deg/sec was established. At 1:10 P.E.T. a second -X RCS burn was performed for 14 seconds while maintaining the pitch rate. This burn was required to prevent the payload from going too far behind and too close to the Orbiter ($Z_B \rightarrow 0 \text{ FT}$), blocking the view of the payload behind the OMS pods. At approximately 2:41 P.E.T., the -135 degree angle of pitch from the nominal ejection attitude has been reached, an inertial hold initiated, and a 14-second -X RCS burn begun. This final translation places the Orbiter approximately 550 feet behind and 950 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 5-2).

Figure 5-11 illustrates the SSUS-D positions relative to the -X RCS plume during the 3 translational maneuvers. Because of the high degree of plume expansion, impingement cannot be avoided entirely. However, only the first -X RCS burn will cause plume impingement. The RCS dynamic pressure experienced by the payload should be no greater than approximately 2×10^{-5} pounds per square foot during the 31-second total duration of the 3 RCS translational maneuvers.

Table 5-1. Orbiter/SSUS-D Initial Separation Sequence
 (-135 Deg Rotation Between Ejection and
 OMS Ignition)

1. Initiate 3-second -X RCS burn at 0:20 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at -1.0 deg/sec immediately after -X burn.
3. At 1:10 P.E.T., burn -X for 14 seconds while maintaining -1.0 deg/sec pitch rate.
4. Begin inertial hold upon reaching -135 deg angle of pitch from nominal ejection attitude (at approximately 2:41 P.E.T.).
5. Execute 14-second -X burn immediately after beginning inertial hold.
6. Trim to nominal OMS burn attitude, if necessary.
7. Begin 15-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	236
Aft Tanks	<u>23</u>
Total	259

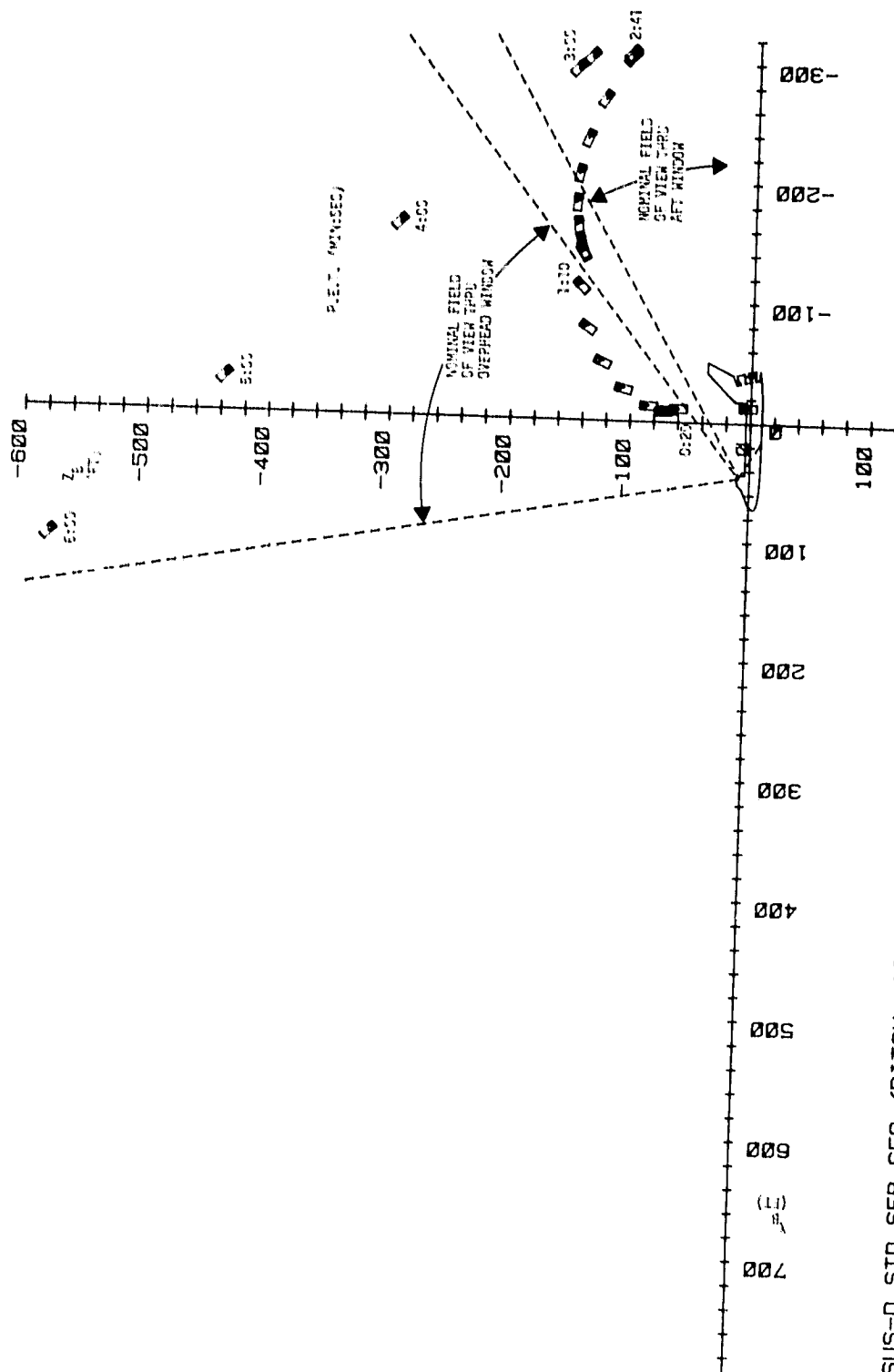


Figure 5-10. Enlarged View of Grbiter/SSUS-D Initial Separation Phase (-135 Deg Rotation Between Ejection and OMS Ignition)

SSUS-D STD SEP SEQ (PITCH -135 DEG)

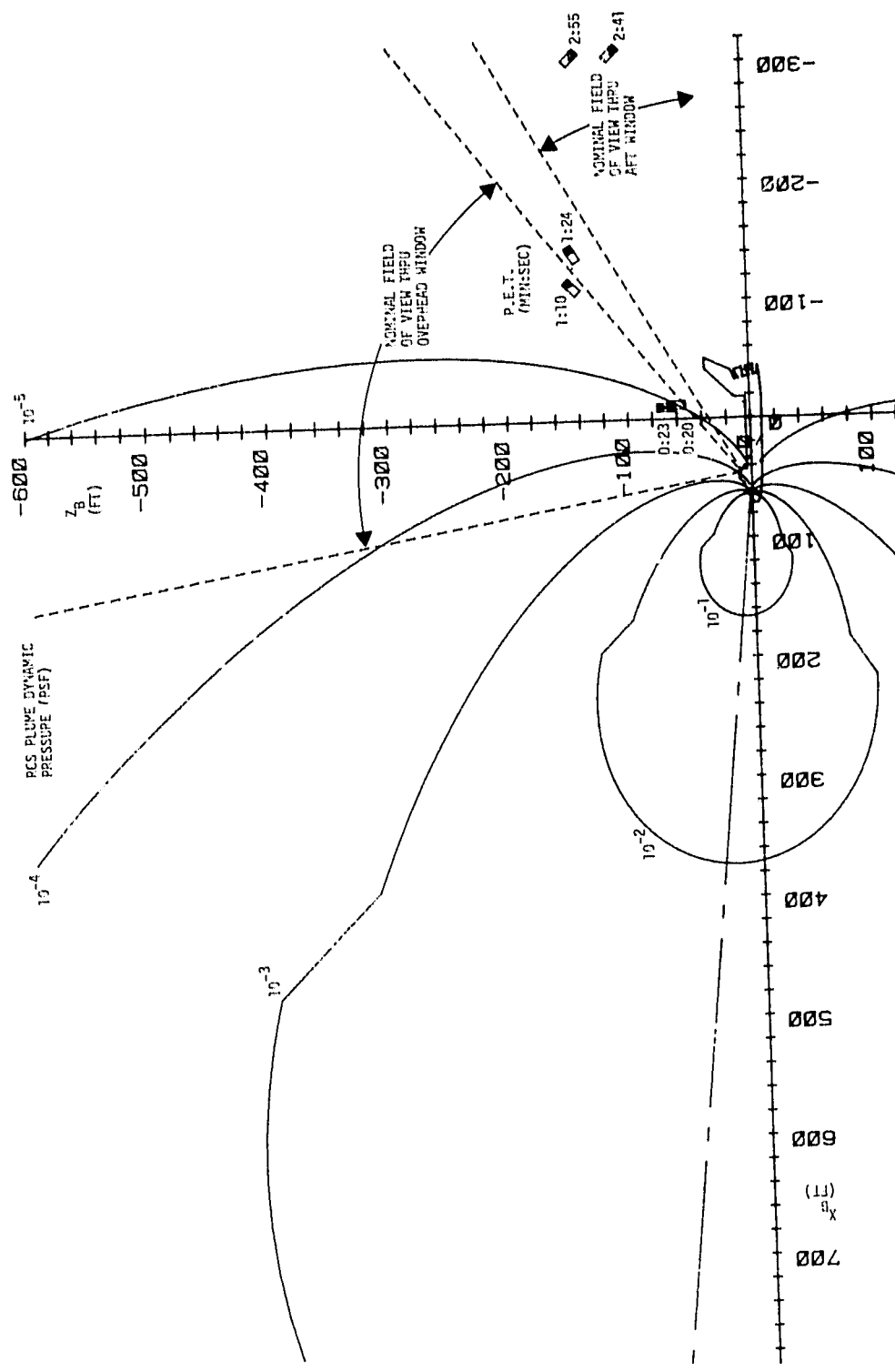


Figure 5-11. SSUS-D Positions Relative to -X RCS Plume (-135 Deg Rotation Between Ejection and OMS Ignition)

SSUS-D STD SEP SEQ (PITCH -135 DEG)

5.2.2 Orbiter Rotation Angle = -90 Degrees

Table 5-2 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-D initial separation sequence and Figure 5-12 presents an enlarged view of the early stage (to 5:00 P.E.T.). Examination of Figure 5-12 will show that the payload is visible through the pilot's overhead window until 1:10 P.E.T., and again from 2:10 P.E.T. until 5:00 P.E.T.

The initiation of this sequence is similar to the -135 degree rotation case with a -1.0 deg/sec pitch rate established immediately upon completion of the 3-second -X RCS burn at 0:20 P.E.T. At 1:10 P.E.T. a second -X RCS burn was performed for 10 seconds while maintaining the pitch rate. At approximately 1:56 P.E.T., the -90 degree angle of pitch rotation from the nominal ejection attitude has been reached, an inertial hold initiated, and an 8-second -X RCS burn begun. This burn is less than the final translation maneuver required in the -135 degree rotation case because the payload is not as far behind the Orbiter at the completion of the pitch maneuver. This burn places the Orbiter approximately 600 feet behind and 1000 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 5-3).

Figure 5-13 illustrates the SSUS-D positions relative to the -X RCS plume during the 3 translational maneuvers. The impingement situation is the same as in the -135 degree rotation case. The payload should experience plume dynamic pressure not greater than approximately 2×10^{-5} pounds per square foot during the 21-second total duration of the 3 RCS translational maneuvers.

Table 5-2. Orbiter/SSUS-D Initial Separation Sequence
 (-90 Deg Rotation Between Ejection and OMS
 Ignition)

1. Initiate 3-second -X RCS burn at 0:20 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at -1.0 deg/sec immediately after -X burn.
3. At 1:10 P.E.T., burn -X for 10 seconds while maintaining -1.0 deg/sec pitch rate.
4. Begin inertial hold upon reaching -90 deg angle of pitch from nominal ejection attitude (at approximately 1:56 P.E.T.).
5. Execute 8-second -X burn immediately after beginning inertial hold.
6. Trim to nominal OMS burn attitude, if necessary.
7. Begin 15-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	165
Aft Tanks	<u>24</u>
Total	189

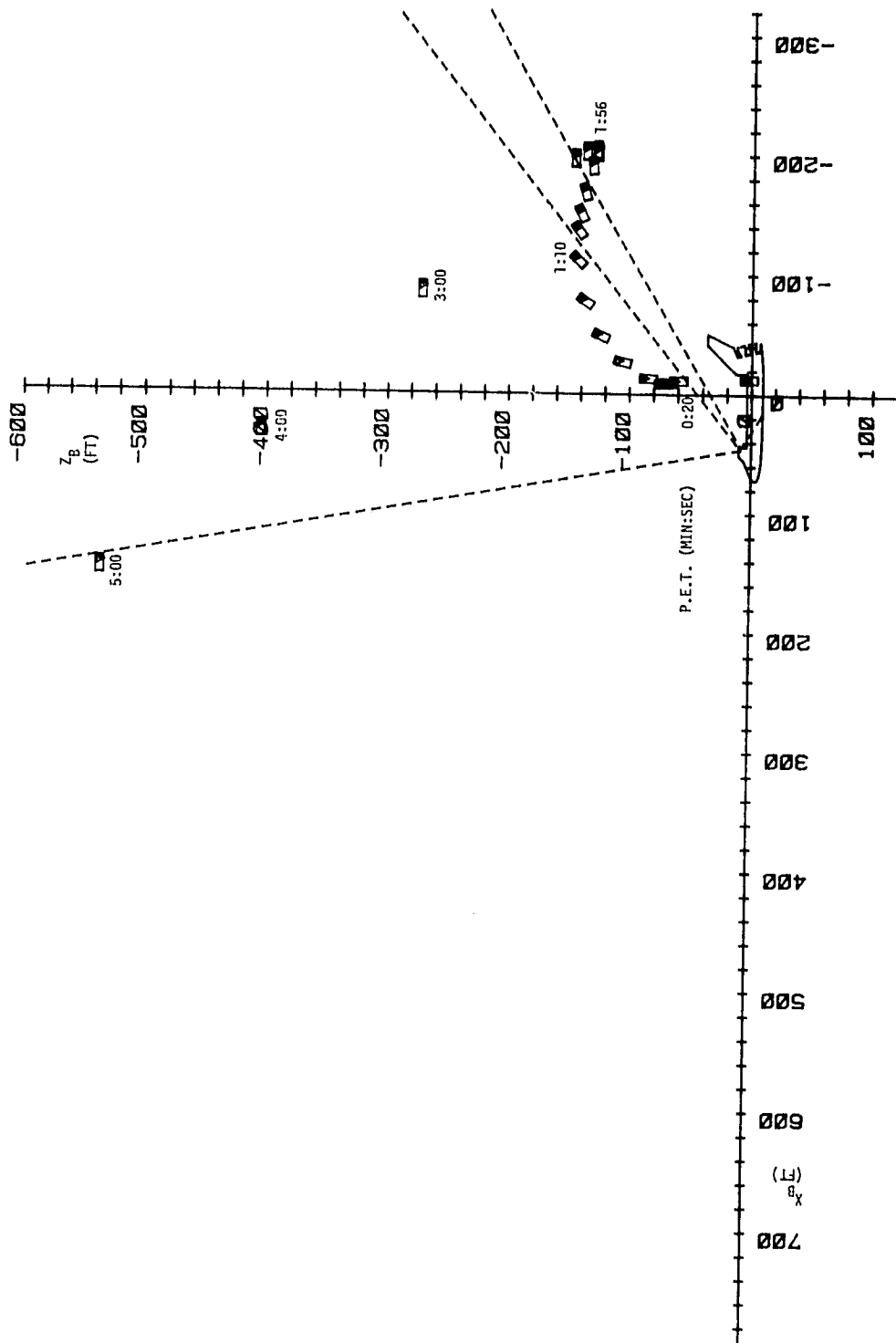


Figure 5-12. Enlarged View of Orbiter/SSUS-D Initial Separation Phase (-90 Deg Rotation Between Ejection and OMS Ignition)

SSUS-D STD SEP SEQ (PITCH -90 DEG)

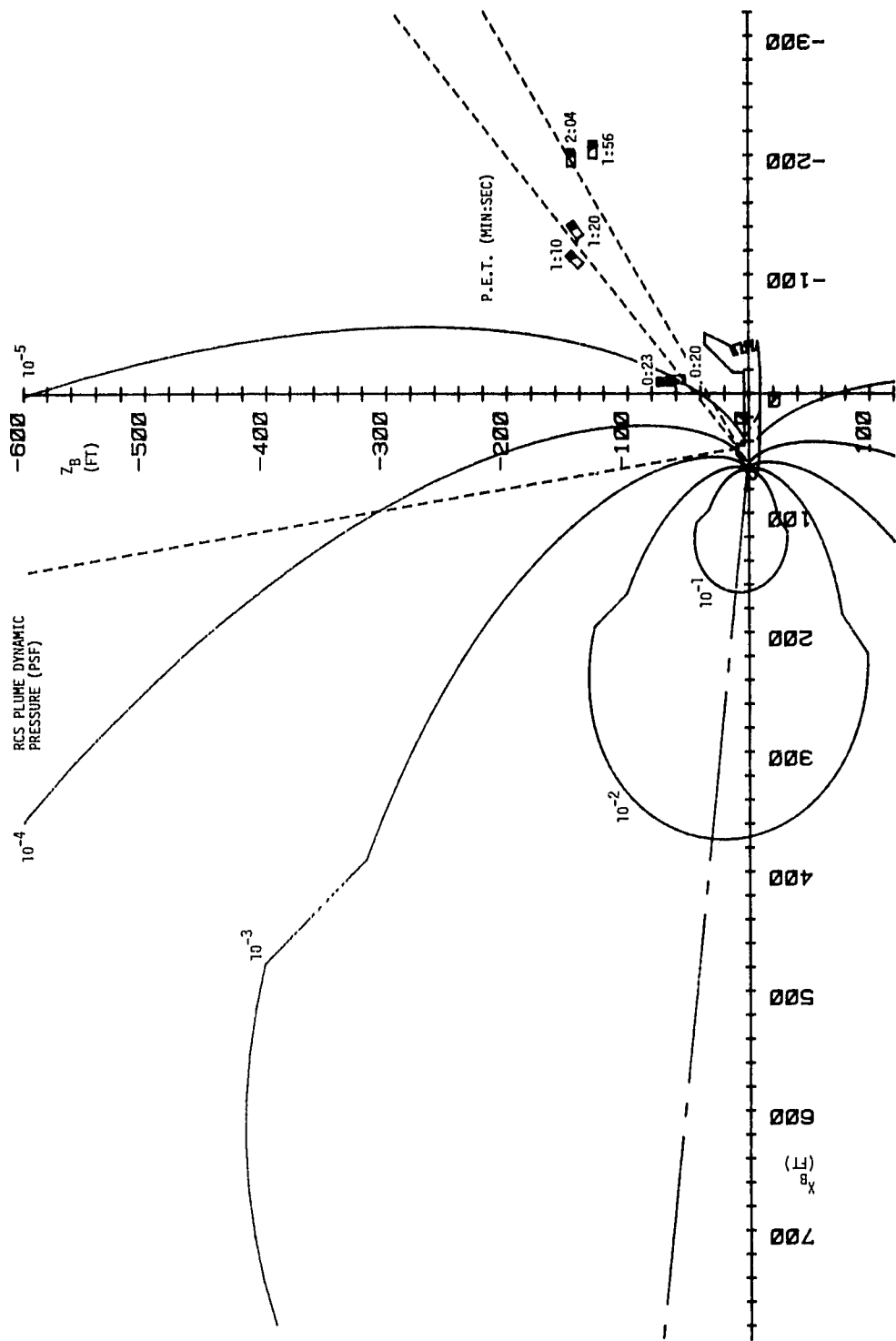


Figure 5-13. SSUS-D Positions Relative to -X RCS Plume (-90 Deg Rotation Between Ejection and OMS Ignition)

SSUS-D STD SEP SEQ (PITCH -90 DEG)

5.2.3 Orbiter Rotation Angle = -45 Degrees

Table 5-3 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-D initial separation sequence and Figure 5-14 presents an enlarged view of the early stage (to 5:00 P.E.T.). Examination of Figure 5-14 will show that the payload is visible through the pilot's overhead window almost continuously for the first 4 minutes of the sequence.

The initiation of this sequence is similar to the -135 and -90 degree rotation cases with a -1.0 deg/sec pitch rate established immediately upon completion of the 3-second -X RCS burn at 0:20 P.E.T. At approximately 1:11 P.E.T., the -45 degree angle of pitch rotation from the nominal ejection attitude has been reached, an inertial hold initiated, and an 11-second -X RCS burn begun. This final RCS translation places the Orbiter approximately 500 feet behind and 1100 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 5-4).

Figure 5-15 illustrates the SSUS-D positions relative to the -X RCS plume during the 2 translational maneuvers. The impingement situation is the same as in the -135 and -90 degree rotation cases. The payload should experience plume dynamic pressure not greater than approximately 2×10^{-5} pounds per square foot during the 14-second total duration of the 2 RCS translational maneuvers.

Table 5-3. Orbiter/SSUS-D Initial Separation Sequence
(-45 Deg Rotation Between Ejection and OMS
Ignition)

1. Initiate 3-second -X RCS burn at 0:20 P.E.T.
2. Null yaw and roll rates and stabilize pitch rate at -1.0 deg/sec immediately after -X burn.
3. Begin inertial hold upon reaching -45 deg angle of pitch from nominal ejection attitude (at approximately 1:11 P.E.T.).
4. Execute 11-second -X burn immediately after beginning inertial hold.
5. Trim to nominal OMS burn attitude, if necessary.
6. Begin 15-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	114
Aft Tanks	<u>23</u>
Total	137

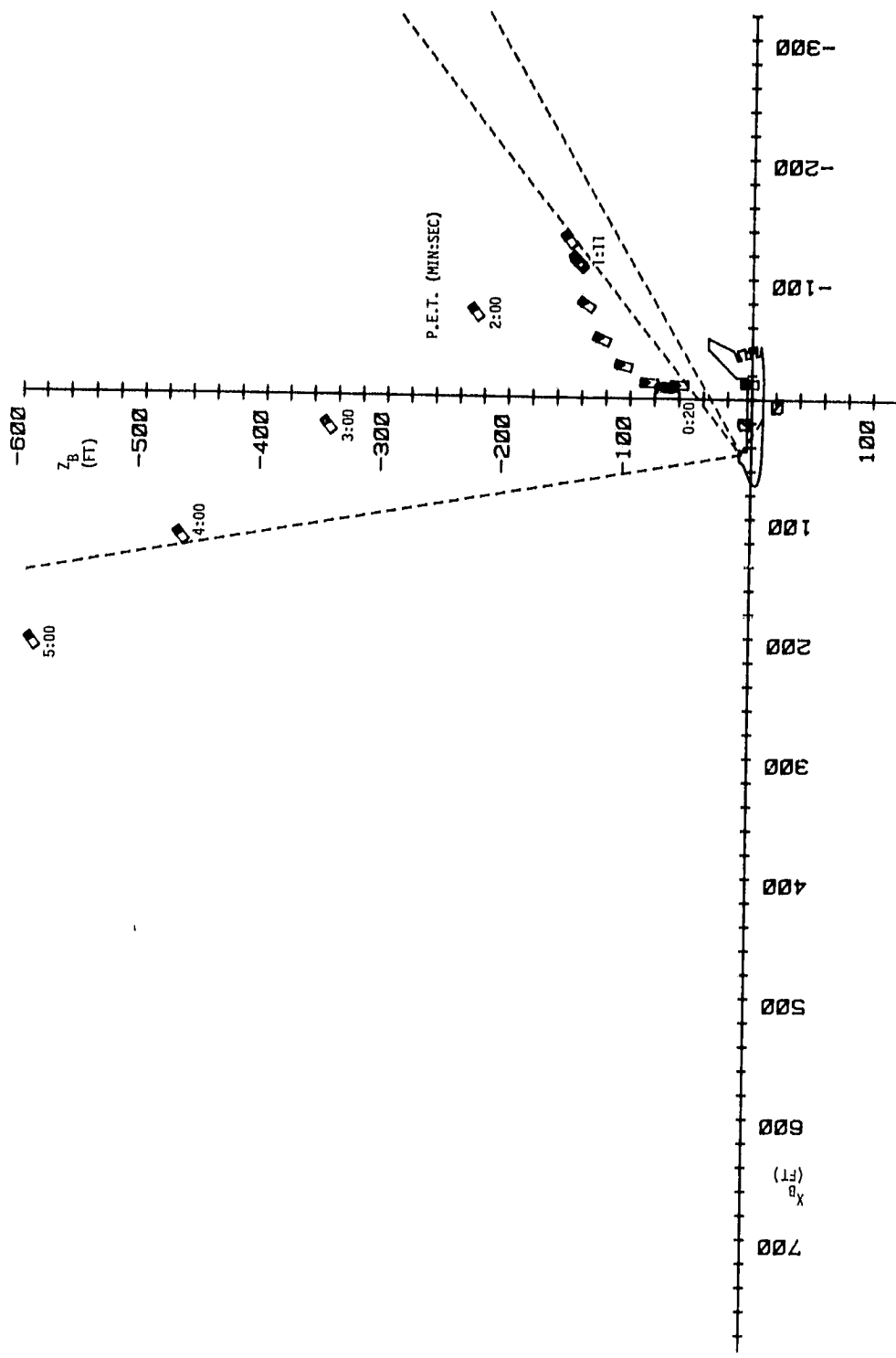
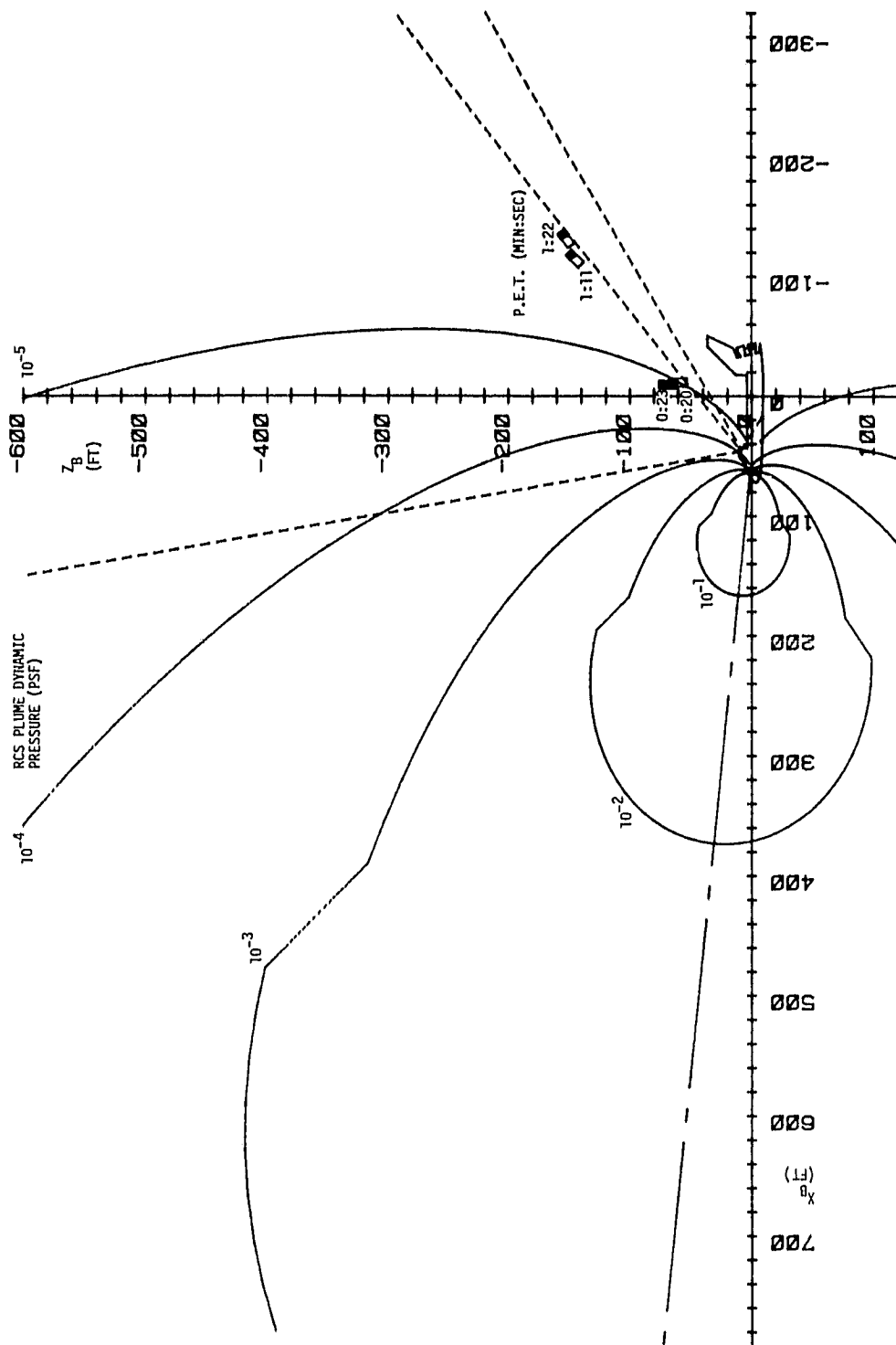


Figure 5-14. Enlarged View of Orbiter/SSUS-D Initial Separation Phase (-45 Deg Rotation Between Ejection and OWS Ignition)

SSUS-D STS SEP SEQ (PITCH -45 DEG)



SSUS-D STD SEP SEQ (PITCH - 45 DEG) Figure 5-15. SSUS-D Positions Relative to -X RCS Plume (-45 Deg Rotation Between Ejection and DMS Ignition)

5.2.4 Orbiter Rotation Angle = 0 Degrees

Table 5-4 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-D Initial separation sequence and Figure 5-16 presents an enlarged view of the early stage (to 3:00 P.E.T.). Examination of Figure 5-16 will show that the payload is visible through the pilot's overhead window almost continuously for the first 3 minutes of the sequence.

This sequence is unique in that the initial 3-second -X RCS burn at 0:20 P.E.T. provides sufficient translation to place the Orbiter approximately 550 feet behind and 1700 feet below the payload at 9:00 P.E.T., the time of OMS ignition (see Figure 5-5).

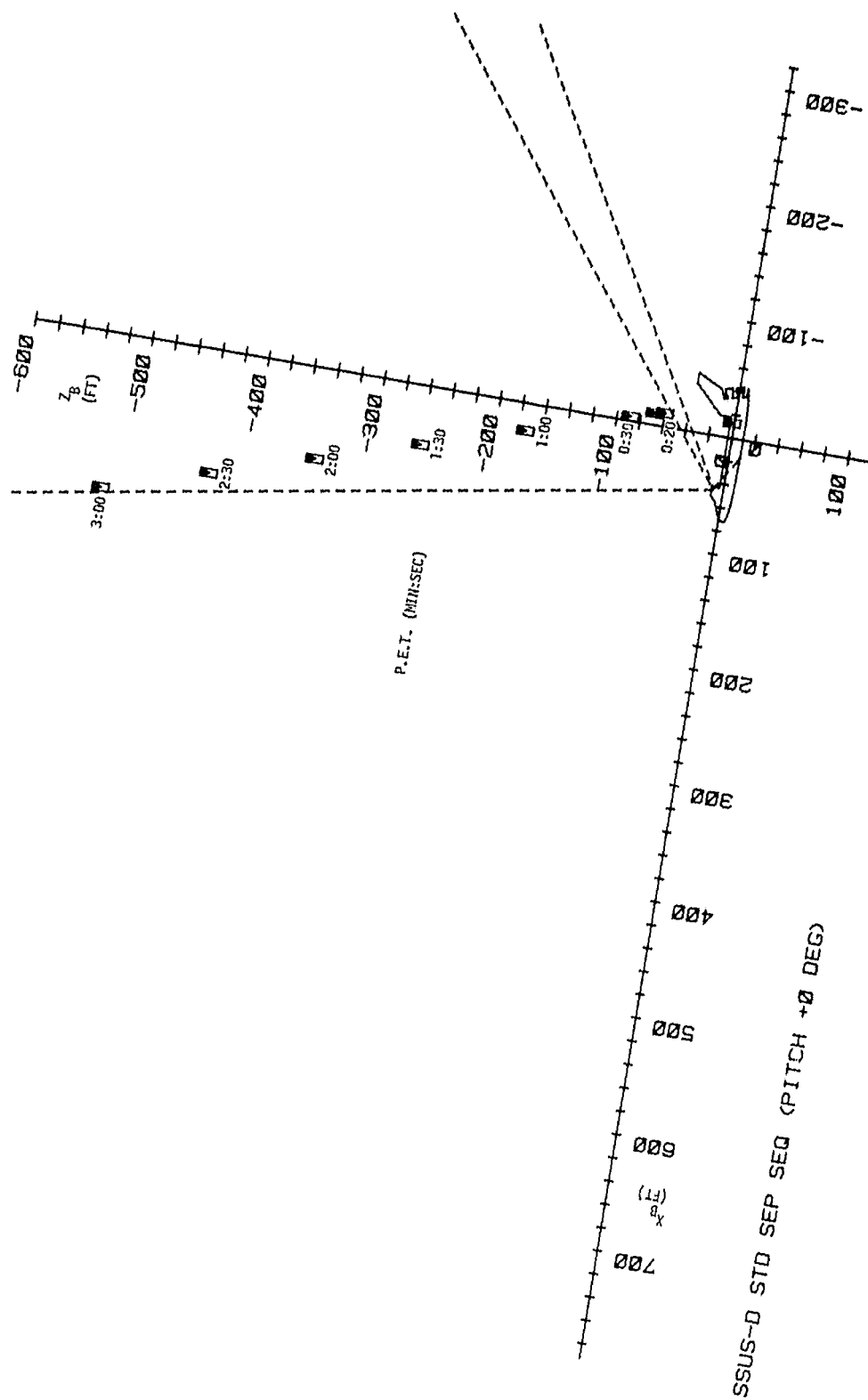
Figure 5-17 illustrates the SSUS-D positions relative to the -X RCS plume during the translational maneuver. The impingement situation is the same as for all three negative rotation angle cases. The payload should experience plume dynamic pressure not greater than approximately 2×10^{-5} pounds per square foot during the 3-second total duration of the RCS translation maneuver.

Table 5-4. Orbiter/SSUS-D Initial Separation Sequence
(0 Deg Rotation Between Ejection and OMS
Ignition)

1. Initiate 3-second -X RCS burn at 0:20 P.E.T.
2. Begin inertial hold immediately after -X burn.
3. Trim to nominal OMS burn attitude, if necessary.
4. Begin 15-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	24
Aft Tanks	<u>5</u>
Total	29



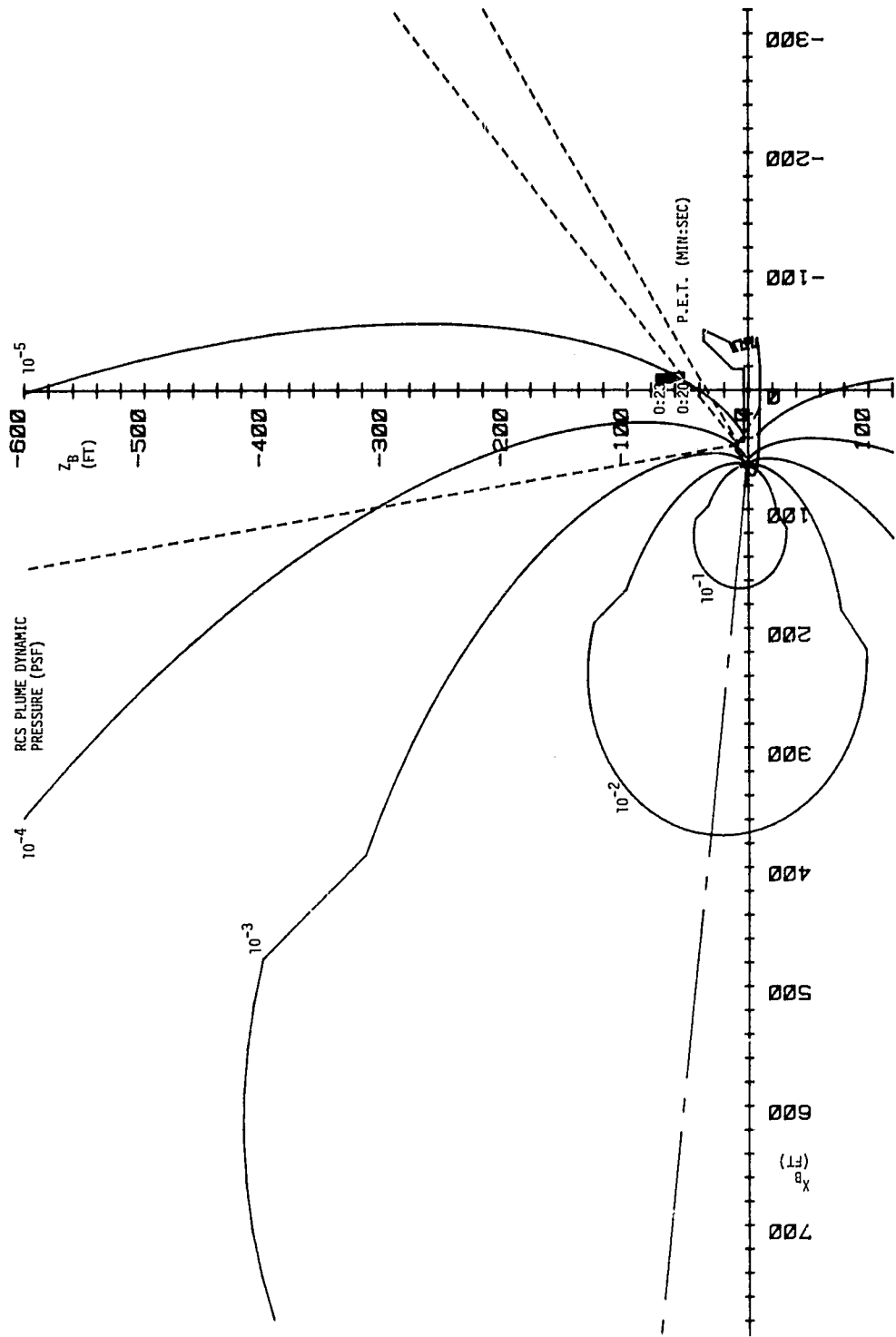


Figure 5-17. SSUS-D Positions Relative to -X RCS Plume (0 Deg Rotation Between Ejection and OMS Ignition)

SSUS-D STD SEP SEQ (PITCH +0 DEG)

5.2.5 Orbiter Rotation Angle = +45 Degrees

Table 5-5 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-D initial separation sequence and Figure 5-18 presents an enlarged view of the early stage (to 6:00 P.E.T.). Examination of Figure 5-18 will show that the payload is visible through the pilot's overhead window for only 50 seconds of the sequence.

This, and the remaining sequences (positive Orbiter rotation angles), are initiated differently than the previous cases. The Orbiter's motion was uncontrolled from 0:00 P.E.T. until 0:20 P.E.T., at which time a 1.0 deg/sec pitch rate was initiated. At approximately 1:06 P.E.T., the +45 degree angle of pitch rotation from the nominal ejection attitude has been reached and an inertial hold initiated. No RCS translational maneuver is required for this case. The initial ejection impulse is sufficient to place the Orbiter approximately 1050 feet behind and 750 feet below the payload at 9:00 P.E.T., the time of OMS ignition; which provides at least 850 feet of clearance between the payload and the OMS plume during the 15-second OMS burn (see Figure 5-6).

Table 5-5. Orbiter/SSUS-D Initial Separation Sequence
(+45 Deg Rotation Between Ejection and OMS
Ignition)

1. Start +1.0 deg/sec pitch maneuver at 0:20 P.E.T.
2. Begin inertial hold upon reaching +45 deg angle of pitch from nominal ejection attitude (at approximately 1:06 P.E.T.).
3. Trim to nominal OMS burn attitude, if necessary.
4. Begin 15-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	11
Aft Tanks	22
Total	33

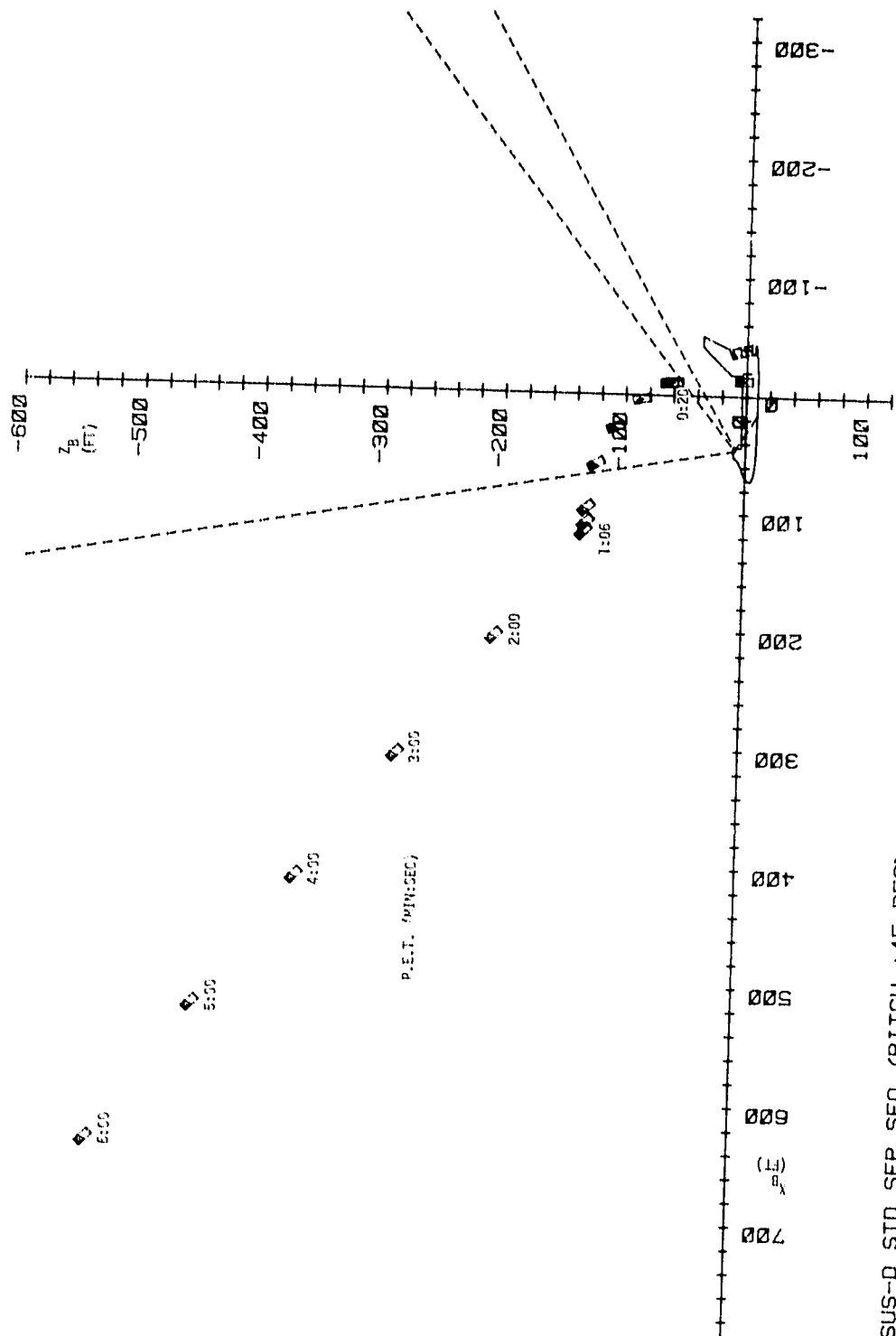


Figure 5-10. Enlarged View of Orbiter/SSUS-D Initial Separation Phase (+45 Deg Rotation Between Ejection and OMS Ignition)

SSUS-D STD SEP SEQ (PITCH +45 DEG)

5.2.6 Orbiter Rotation Angle = +90 Degrees

Table 5-6 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-D initial separation sequence and Figure 5-19 presents an enlarged view of the early stage (to 8:00 P.E.T.). Examination of Figure 5-19 will show that the payload is visible through the pilot's overhead window for only 50 seconds of the sequence.

The initiation of this sequence is similar to the +45 degree rotation case with a 1.0 deg/sec pitch rate established at 0:20 P.E.T. At 1:05 P.E.T., a 6-second +X RCS burn was performed while maintaining the pitch rate. This burn was performed to prevent the payload from being below the Orbiter ($Z_B \rightarrow +$) at the completion of the pitch rotation and out of the line-of-sight of the crew through the forward windows. At approximately 1:51 P.E.T., the +90 degree angle of pitch from the nominal ejection attitude has been reached and an inertial hold initiated. No additional RCS translational maneuvers are required to insure OMS plume clearance. The Orbiter is approximately 900 feet behind and 300 feet below the payload at 9:00 P.E.T., the time of OMS ignition, which provides at least 700 feet of clearance between the payload and the OMS plume during the 15-second OMS burn (see Figure 5-7).

Figure 5-20 illustrates the SSUS-D positions relative to the +X RCS plume during the 6-second translational maneuver. Nominally, no plume dynamic pressure should be experienced by the payload.

Table 5-6. Orbiter/SSUS-D Initial Separation Sequence
(+90 Deg Rotation Between Ejection and OMS
Ignition)

1. Start +1.0 deg/sec pitch maneuver at 0:20 P.E.T.
2. At 1:05 P.E.T., burn +X RCS for 6 seconds while maintaining +1.0 deg/sec pitch rate.
3. Begin inertial hold upon reaching +90 deg angle of pitch from nominal ejection attitude (at approximately 1:51 P.E.T.).
4. Trim to nominal OMS burn attitude, if necessary.
5. Begin 15-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	13
Aft Tanks	<u>60</u>
Total	73

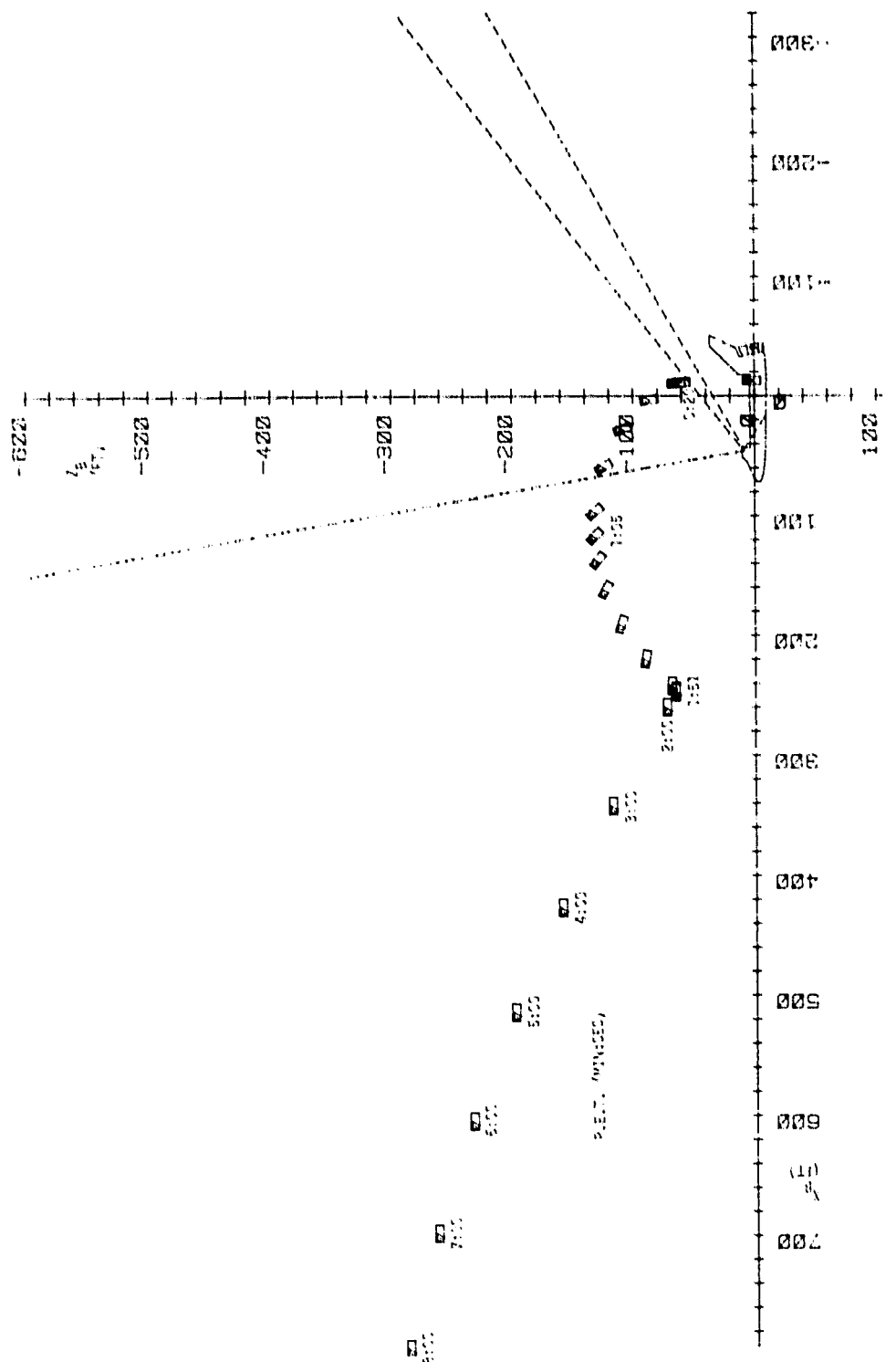


Figure 5-13. Enlarged view of scatter plot area at separation phase +90 deg based on separation phase and yaw velocity.

SSUS-0 STD SEP SEQ (PITCH +90 DEG)

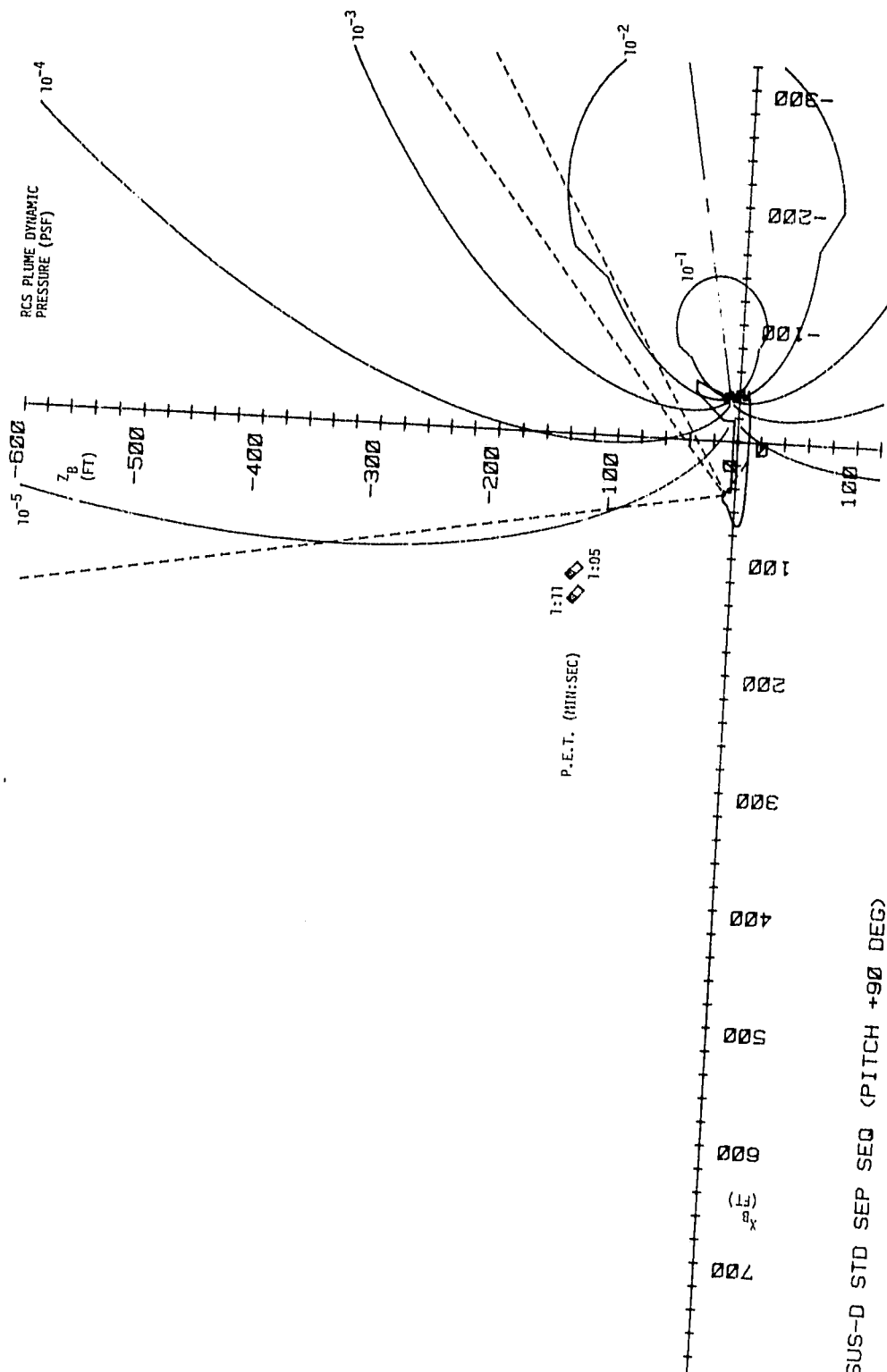


Figure 5-20. SSUS-D Positions Relative to +X RCS Plume (+90 Deg Rotation Between Ejection and 0.95 Ignition)

SSUS-D STD SEP SEQ (PITCH +90 DEG)

5.2.7 Orbiter Rotation Angle = +135 Degrees

Table 5-7 presents the schedule of events and the nominal RCS propellant consumption for this Orbiter/SSUS-D initial separation sequence and Figure 5-21 presents an enlarged view of the early stage (to 6:00 P.E.T.). Examination of Figure 5-21 will show that the payload is visible through the pilot's overhead window for only 50 seconds of the sequence.

The initiation of this sequence is similar to the +45 and +90 degree rotation cases with a 1.0 deg/sec pitch rate established at 0:20 P.E.T. At 1:05 P.E.T., a 17-second +X RCS burn is performed while maintaining the pitch rate. This burn was performed to prevent the payload from being below the Orbiter ($Z_B \rightarrow +$) at the completion of the pitch rotation and out of the line-of-sight of the crew through the forward windows. At approximately 2:36 P.E.T., the +135 degree angle of pitch from the nominal ejection attitude is reached and an inertial hold initiated. No additional RCS translational maneuvers are required to insure OMS plume clearance. The Orbiter is approximately 850 feet behind and 950 feet below the payload at 9:00 P.E.T., the time of OMS ignition, which provides at least 650 feet of clearance between the payload and the OMS plume during the 15-second OMS burn (see Figure 5-8).

Figure 5-22 illustrates the SSUS-D positions relative to the +X RCS plume during the 17-second translational maneuver. Nominally, no plume dynamic pressure should be experienced by the payload.

Table 5-7. Orbiter/SSUS/D Initial Separation Sequence
(+135 Deg Rotation Between Ejection and OMS
Ignition)

1. Start +1.0 deg/sec pitch maneuver at 0:20 P.E.T.
2. At 1:05 P.E.T., burn +X RCS for 17 seconds while maintaining +1.0 deg/sec pitch rate.
3. Begin inertial hold upon reaching +135 deg angle of pitch from nominal ejection attitude (at approximately 2:36 P.E.T.).
4. Trim to nominal OMS burn attitude, if necessary.
5. Begin 15-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	15
Aft Tanks	<u>129</u>
Total	144

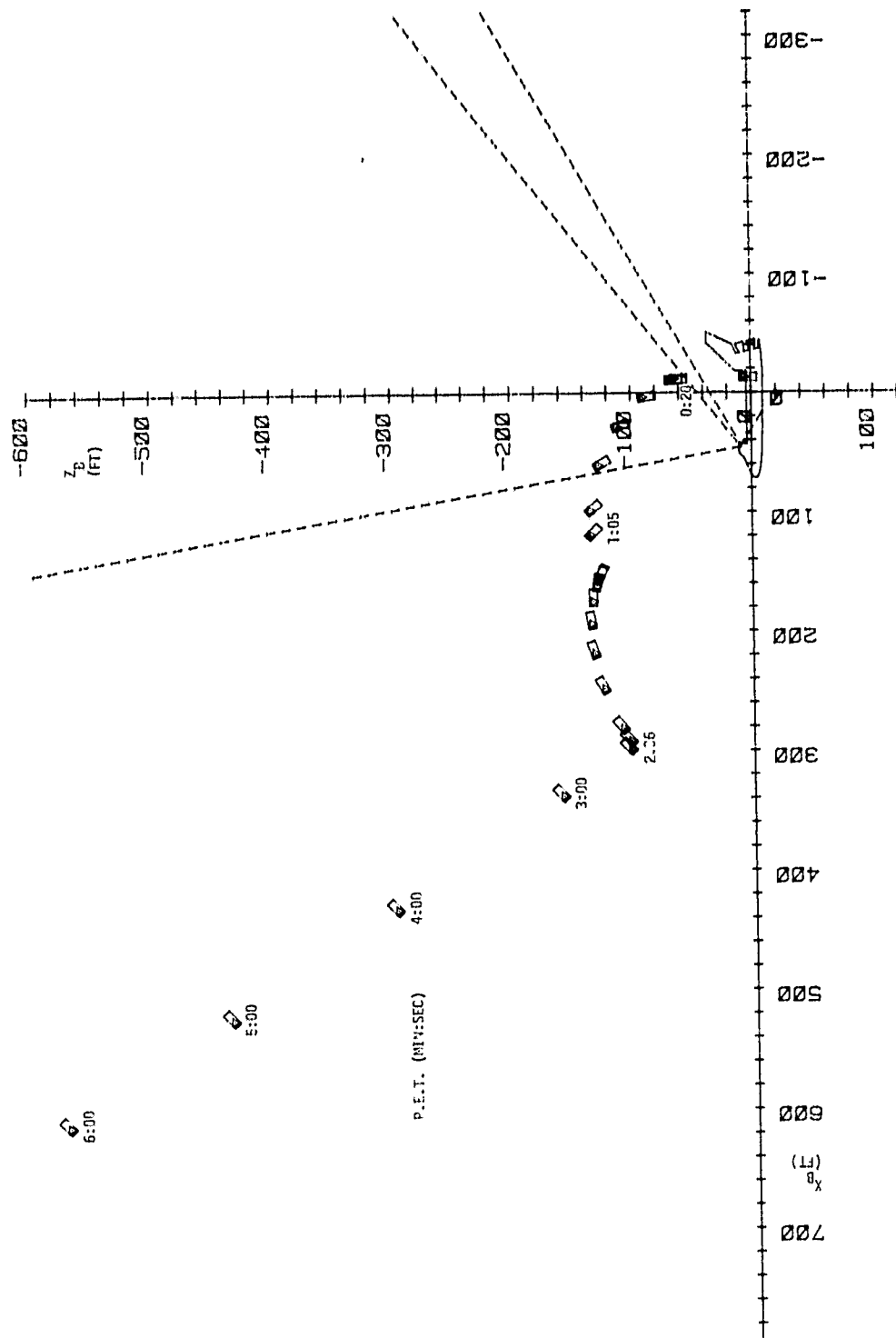


Figure 5-21. Enlarged View of Orbiter/SSUS-D Initial Separation Phase (+135 Deg Rotation Between Ejection and QMS Ignition)

SSUS-D STD SEP SEQ (PITCH +135 DEG)

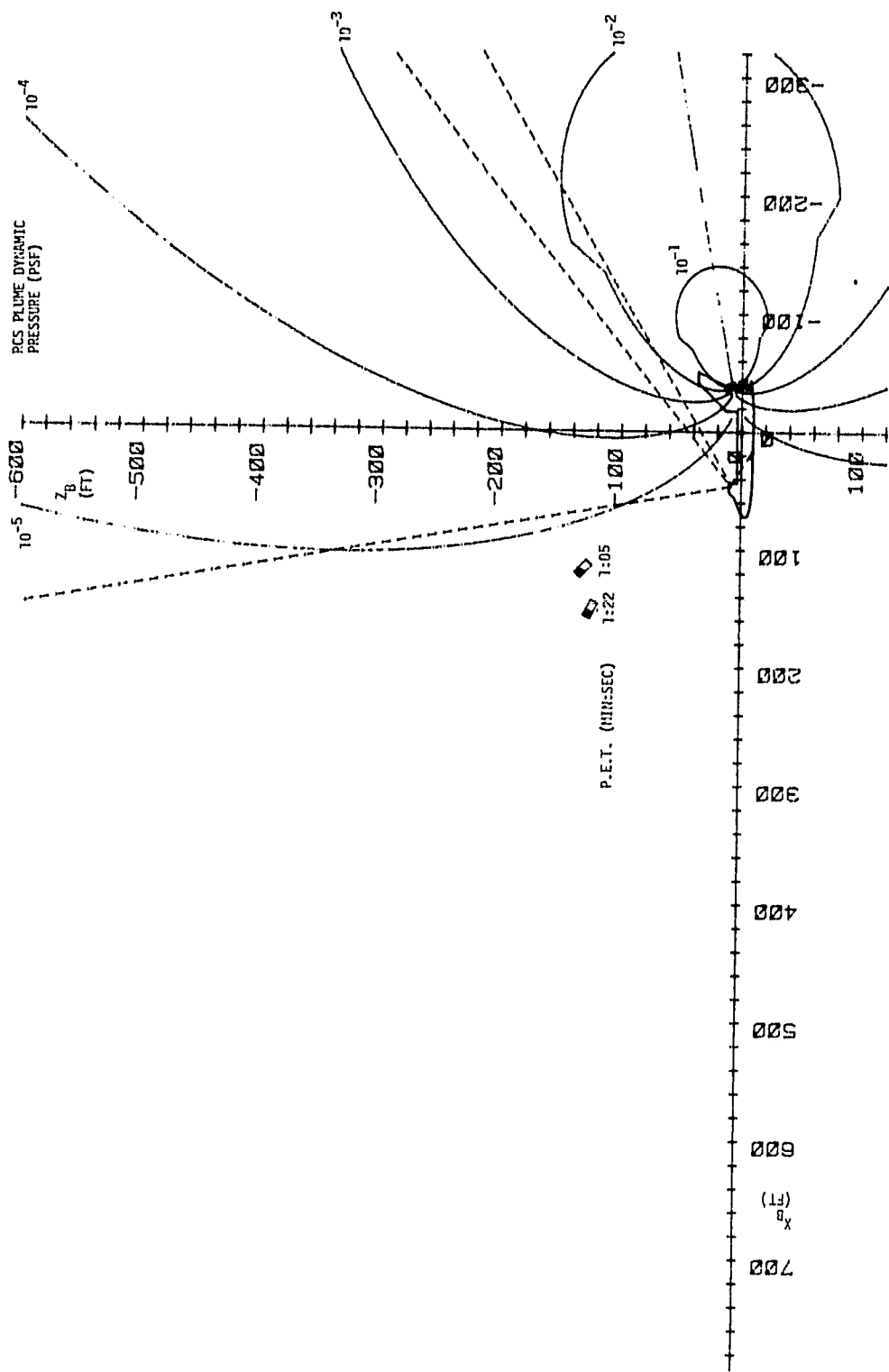


Figure 5-22. SSUS-D Positions Relative to +X RCS Plume (+135 Deg Rotation Between Ejection and OMS Ignition)

SSUS-D STD SEP SEQ (PITCH +135 DEG)

5.2.8 Orbiter Rotation Angle = +180 Degrees

Table 5-8 presents the schedule of events and the final RCS propellant consumption for this Orbiter/SSUS-D initial separation sequence and Figure 5-23 presents an enlarged view of the early stage (to 6:00 P.E.T.). Examination of Figure 5-23 will show that the payload is visible through the pilot's overhead window for only 50 seconds of the sequence.

The initiation of this sequence is similar to the other positive pitch rotation cases with a 1.0 deg/sec pitch rate established at 0:20 P.E.T. At 1:05 P.E.T., a 10-second +X RCS burn is performed while maintaining the pitch rate. Another +X burn occurs at 1:50 P.E.T. for 16 seconds while maintaining the pitch rate. These burns were performed to prevent the payload from being below the Orbiter ($Z_B \rightarrow +$) at the completion of the pitch rotation and to insure OMS plume clearance. The Orbiter is approximately 900 feet behind and 1250 feet below the payload at 9:00 P.E.T., the time of OMS ignition, which provides at least 800 feet of clearance between the payload and the OMS plume during the 15-second OMS burn (see Figure 5-9).

Figure 5-24 illustrates the SSUS-D positions relative to the +X RCS plume during the two translational maneuvers. Nominally, no plume dynamics pressure should be experienced by the payload.

Table 5-8. Orbiter/SSUS-D Initial Separation Sequence
(+180 Deg Rotation Between Ejection and OMS
Ignition)

1. Start +1.0 deg/sec pitch maneuver at 0:20 P.E.T.
2. At 1:05 P.E.T., burn +X RCS for 10 seconds while maintaining +1.0 deg/sec pitch rate.
3. At 1:50 P.E.T., burn +X for 16 seconds while maintaining +1.0 deg/sec pitch rate.
4. Begin inertial hold upon reaching +180 degree angle of pitch from nominal ejection attitude (at approximately 3:21 P.E.T.).
5. Trim to nominal OMS burn attitude, if necessary.
6. Begin 15-second OMS burn (2 engines) at 9:00 P.E.T.

NOMINAL RCS PROPELLANT CONSUMPTION (LB)

Fwd Tanks	15
Aft Tanks	<u>184</u>
Total	199

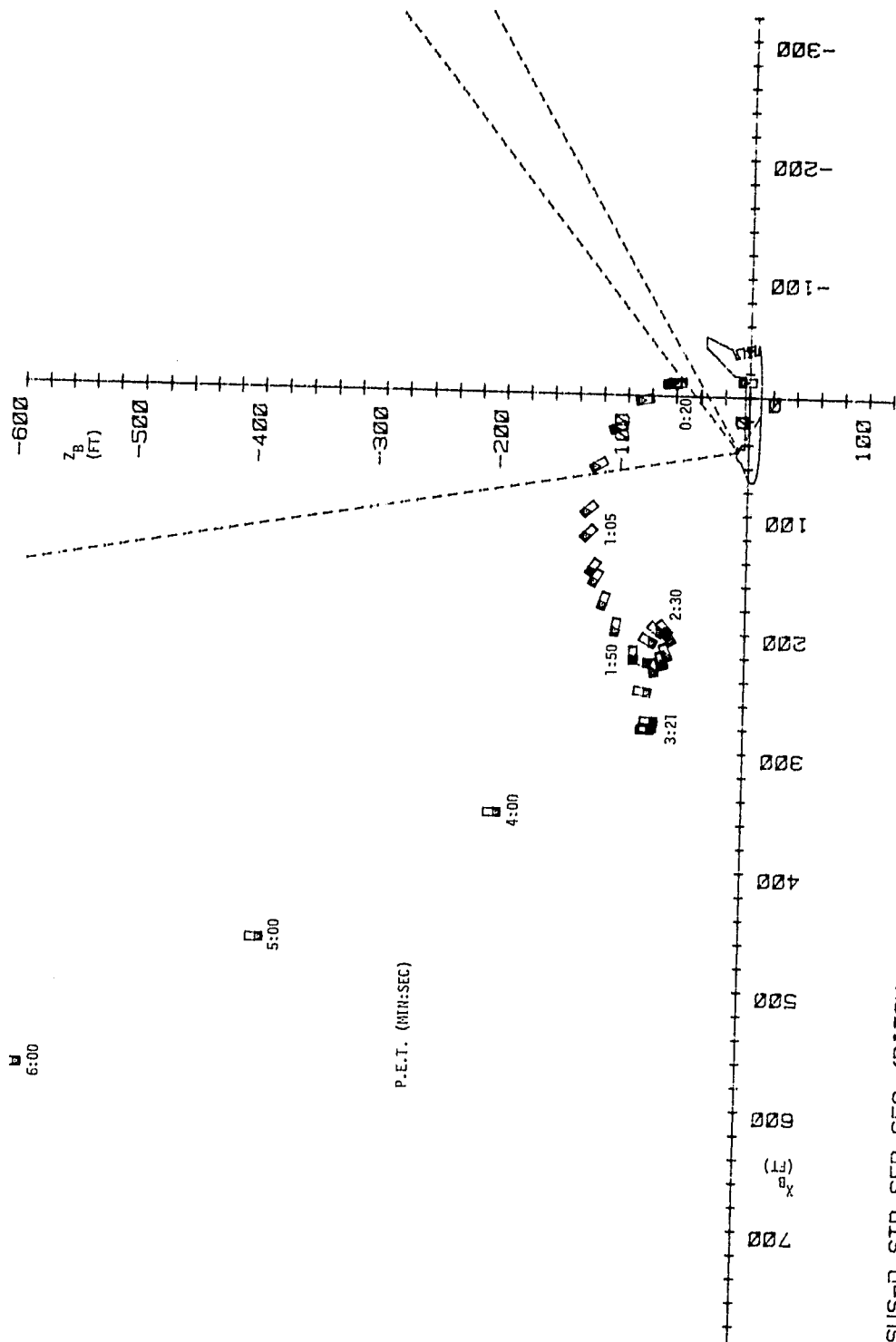
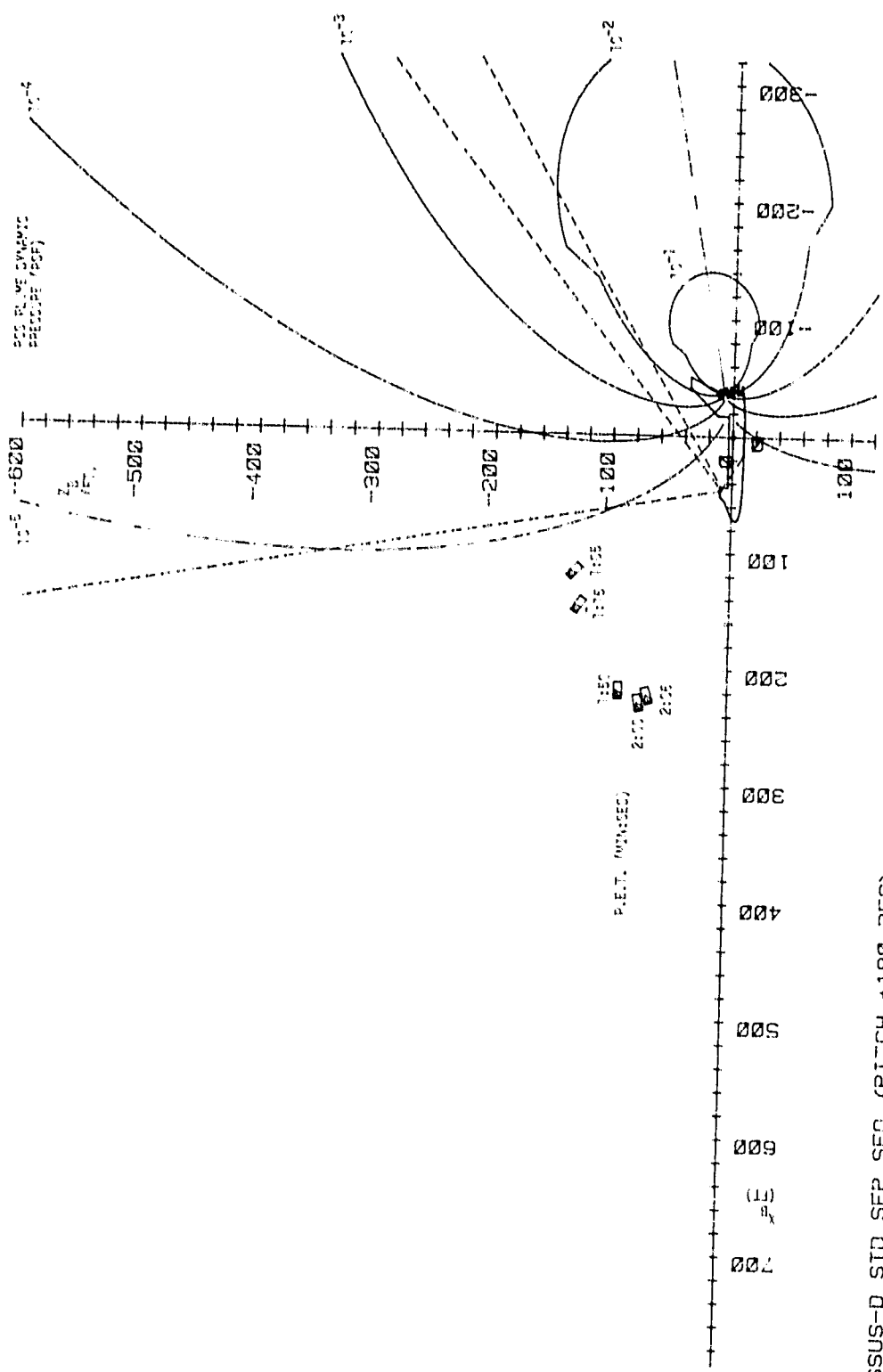


Figure 5-23. Enlarged View of Orbiter/SSUS-D Initial Separation Phase (+180 Deg Rotation Between Ejection and OMS Ignition)



SSUS-D STD SEP SEG (PITCH +180 DEG)

Figure 5-24. SSUS Positions Relative to $\pm 180^\circ$ Pitch
+180° Sep Rotation Between Erection and
1st Iteration.

6.0 REFERENCES

1. S. M. Kindall, "User's Guide for the HP-9825A High Fidelity Relative Motion Program (HFRMP)," TRW IOC No. 78:2511.4-33, 16 October 1978.
2. S. M. Kindall and S. W. Wilson, "Detailed Description of the HP-9825A HFRMP Trajectory Processor (#TRAJ)," TRW Report No. 28415-H011-R0-00, 20 November 1979.
3. S. W. Wilson, "Delivery of HP-9825A High Fidelity Relative Motion Program (HFRMP), Version 03T," TRW Letter No. 79:2552.1-130, 21 November 1979.

APPENDIX

LIMITATION OF SRM EXHAUST PARTICLE DAMAGE TO ORBITER SURFACES

The STS standard upper-stage types (IUS, SSUS-A, SSUS-D) utilize solid-propellant rocket motors (SRMs) as their major propulsion units. The SRM effluent contains solid particles which are capable of inflicting damage on the Orbiter's windows and thermal protection tiles. In the interest of flight safety and avoiding excessive maintenance between flights, Orbiter/upper-stage separation sequences must be designed in such a manner as to limit the damage from SRM exhaust particles. Separation sequences currently are being designed on the basis of the SRM plume models, surface damage models, and damage-limitation criteria which are described in Reference A-1. These models and criteria will be discussed in following sections of this appendix.

A.1. SOURCE AND DISTRIBUTION OF PARTICLES

Aluminum is a major component of the solid propellant used in the IUS and SSUS stages, and aluminum oxide particles account for approximately 34% of the total mass expelled by these stages. According to the SRM exhaust plume models currently being used for damage analysis, the aluminum oxide particle diameters range between 0.018 and 10 μm . Their velocity magnitudes with respect to the upper stage at ejection time range from approximately 2.5 km/s for the largest particles to 3.5 km/s for the smallest ones.

Particles are ejected from the SRM in an axially symmetric pattern, with the mass flux being represented as a function of particle diameter and ejection angle. The ejection angle is measured with respect to the centerline of the exhaust plume (i.e., the SRM thrust line). The range of ejection angles for the largest aluminum oxide particles is from 0 to approximately 11 degrees. For the smallest particles, the ejection angles range from 0 to approximately 49 degrees.

In addition to the aluminum oxide particles just described, current models of the SRM exhaust plume include larger particles with diameters ranging between 18 and 1000 μm . These are rather loosely characterized as "carbon". The relative velocity vectors of these particles (with respect to the upper stage at ejection time) have an annular distribution about the plume centerline, and their magnitudes vary from approximately 0.5 km/s for the 1000 μm particles to approximately 2.4 km/s for the 18 μm particles. The ejection angles range between the approximate limits of 15 and 25 degrees with respect to the plume centerline.

The particle size/velocity distribution for the IUS first-stage SRM plume model is shown in Figure A-1. The shaded regions represent the loci of particle velocity vector coordinates relative to a stationary IUS, in a plane section through the centerline of the plume.

The larger particles (18 - 1000 μm) were included in the plume models originally to account for carbon erosion from the SRM nozzle (see Figure A-2), and that is the basis for the annular distribution of their relative velocity vectors. The original estimates of the mass which would be ejected in the form of (relatively) large particles was later increased to account for erosion of the pressure chamber lining. There now seems to be reason to suspect also that some of the aluminum oxide may agglomerate on the nozzle walls, and then enter the exhaust stream in the form of particles much larger than the nominal 10 μm maximum diameter (Reference A-2).

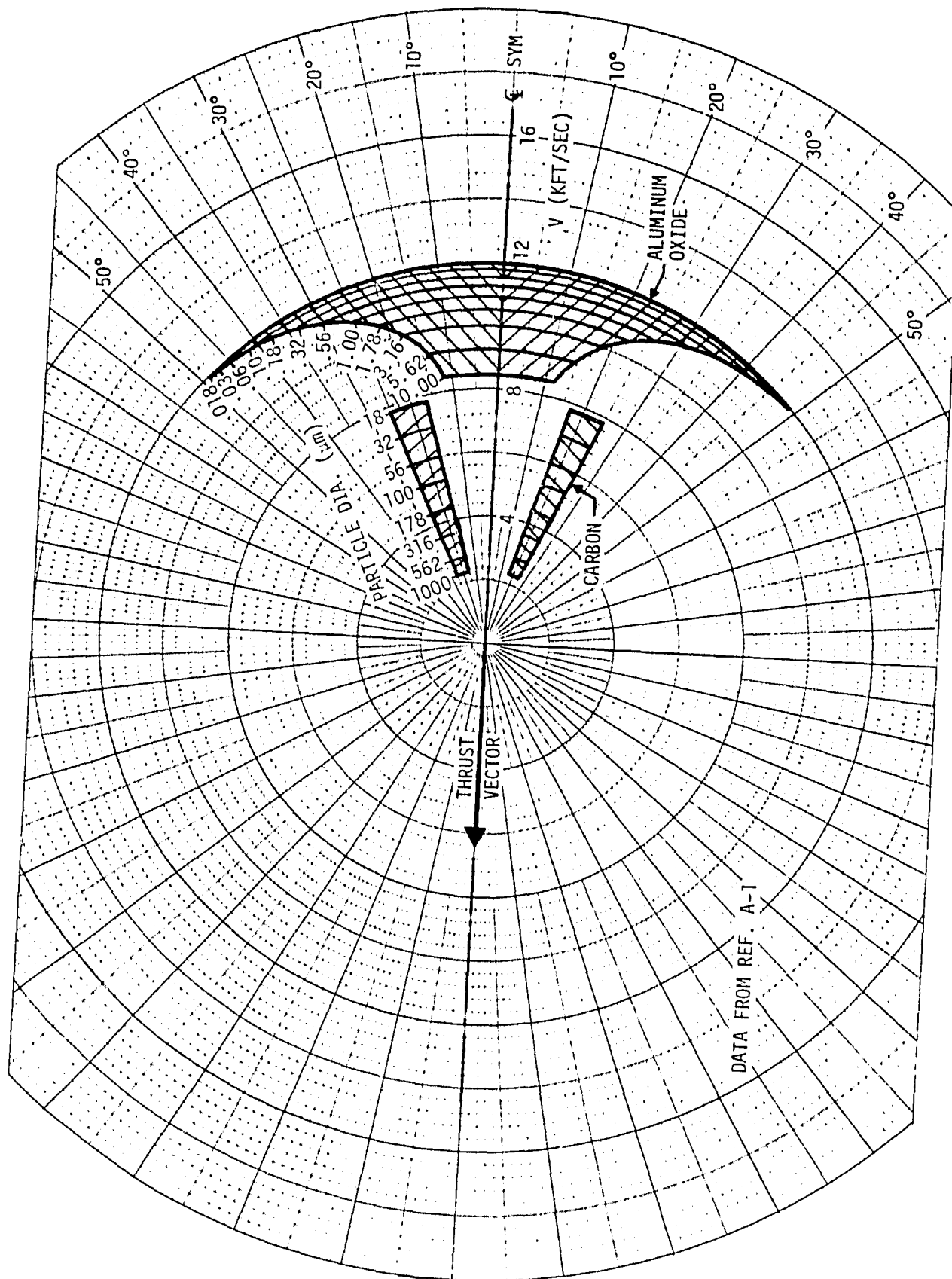


Figure A-1. IUS SRM Exhaust Particle Size/Velocity Distribution with Respect to Stationary IUS

STATE-OF-THE-ART IN PLUME ANALYSIS: SOLID

Engineering Analysis Division

B. Roberts - EX32

5/19/79

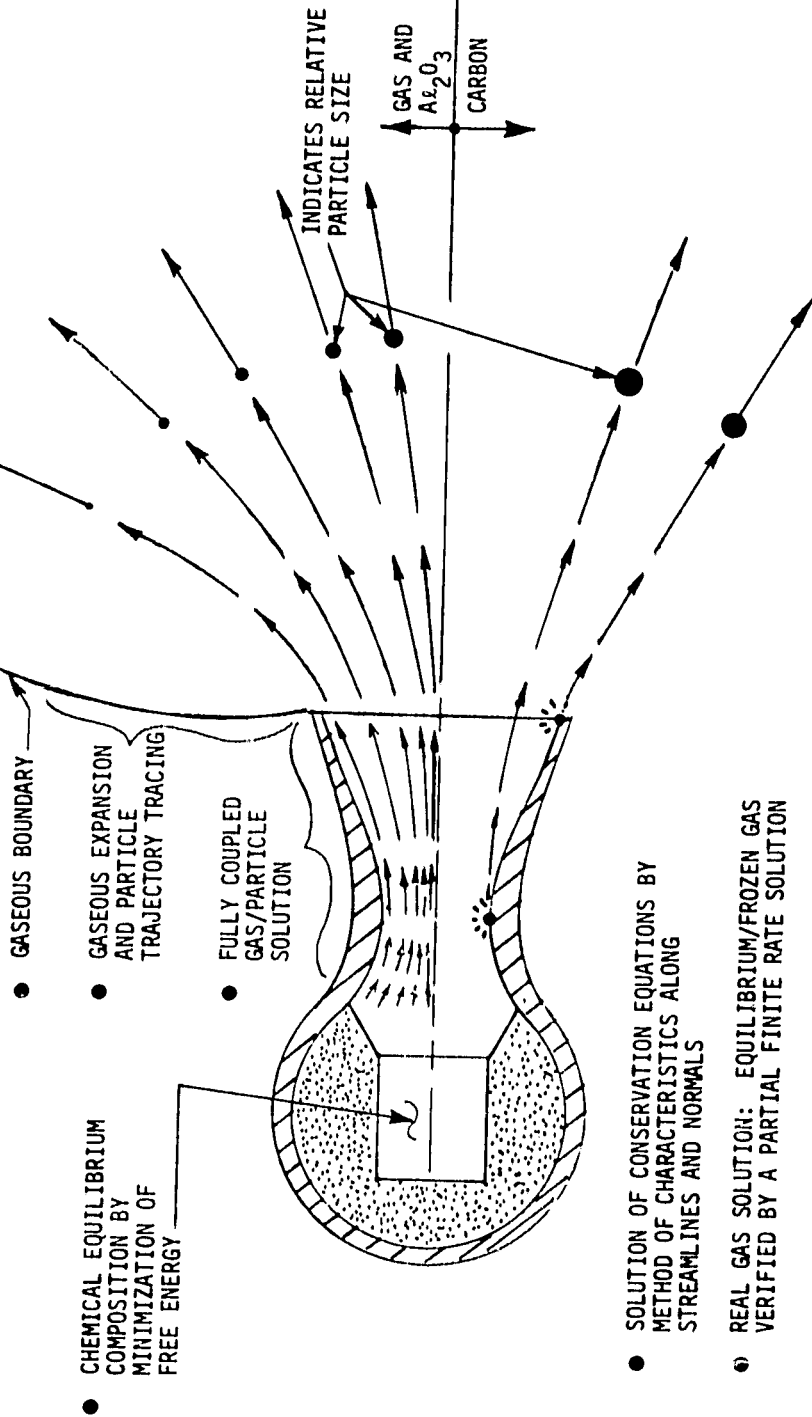


Figure A-2. Schematic Illustration of SRM Exhaust Flowfield

In any event, there is considerable uncertainty about the actual abundance and size distribution of "carbon" particles (i.e., those having diameters greater than 10 μm) in the SRM exhaust plumes. Experiments are planned which should remove some of this uncertainty. Meanwhile, until the test results become available, separation sequences are being designed on the basis of the aluminum oxide and "carbon" particle models of the SRM exhaust plume which are defined in Reference A-1. According to those models, the carbon (large particle) content of the exhaust amounts to somewhere between 0.6% and 1.0% of the total propellant mass (depending on stage type). As indicated in Figure A-3, this amount of mass is distributed into each of eight distinct particle families, every family being composed of particles having a certain diameter in the range of 18 to 1000 μm . Since this results in an eight-fold overabundance of carbon mass in the exhaust model, it might seem that it would produce an eight-fold increase in the calculated damage to Orbiter surfaces and therefore be unrealistic. However, this is not the case. Computer simulations indicate that almost all of the damage inflicted by "carbon" particles ejected during any particular SRM firing is caused by only one particle size, all other sizes being relatively ineffectual for the Orbiter/upper-stage trajectory under consideration. Modeling the large-particle mass distribution in the manner described has the effect of assuming that all of the "carbon" mass is ejected in particles of the one size (which is unknown, a priori) that inflicts maximum damage. Such a degree of conservatism seems reasonable in view of the uncertainties that currently exist with regard to the actual abundance and size-distribution of the larger particles.

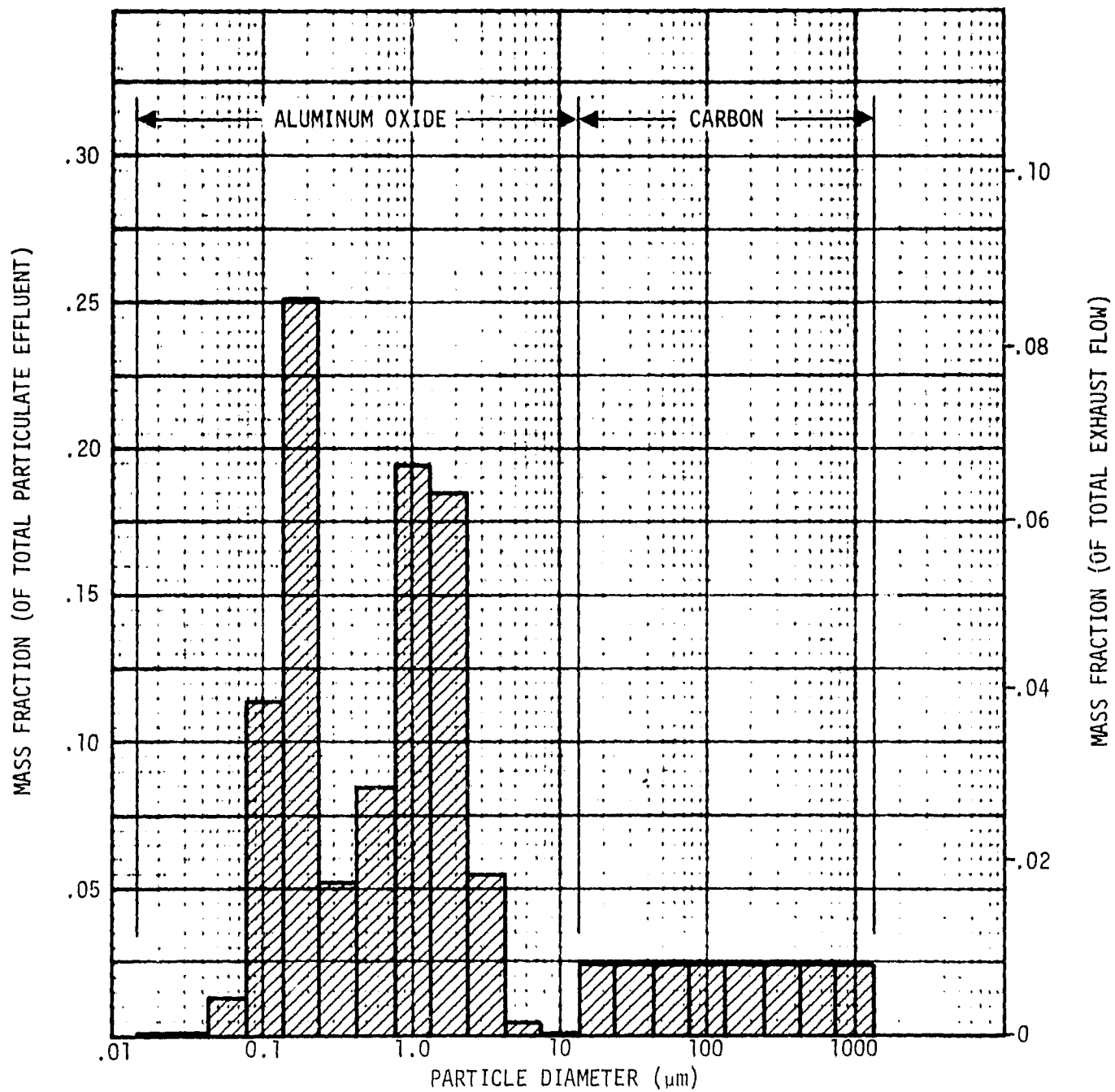


Figure A-3. IUS SRM Exhaust Particle Size/Mass Distribution

A.2 DAMAGE DEFINITIONS

SRM exhaust particle damage to Orbiter surfaces is classified either as erosion or as breakage. Breakage is measured in terms of "breaks" per unit of surface area. With reference to a window, a break represents the formulation of an impact crater of sufficient depth (46 μm or more) to possibly result in crack propagation under the stress of atmospheric entry and/or ascent of the Orbiter on a subsequent flight. With reference to a tile, a break represents a penetration of its fused silica coating. The thickness of the coating on a low-temperature tile is 305 μm , while for a high-temperature tile it is 381 μm . Thus it can be seen that, of the three surface types under consideration, high-temperature tiles are the least susceptible to breakage and windows are the most susceptible.

Erosion refers to the formation of craters too shallow to be classified as breaks. It is measured in terms of the percentage of the total surface area (not depth) that is chipped off by the aggregate of particle impacts.

The "carbon" particles are the primary agents of breakage, the individual masses of aluminum oxide particles generally being too small to form deep craters. On the other hand, the aluminum oxide particles are the primary agents of erosion, simply because they are so very much more numerous than the larger "carbon" particles.

A.3. DAMAGE-LIMITATION CRITERIA

Table A-1 contains a summary of the damage-limitation criteria currently being used for flight design. With regard to window and tile breakage, the basic criterion is that the lifetime accumulation of breakage due to SRM particle impingement should not exceed that which is expected to result from meteoroid impacts. Assuming the Orbiter surfaces experience 135 SRM firings and 421 days on orbit during a design lifetime of 100 flights, it follows that the breakage due to a single SRM firing should not exceed the meteoroid breakage that is expected to accumulate during 3.1 days on orbit.

The lifetime limit on window erosion is currently set at 2%. Any greater amount might require window replacement because of degraded visibility. The per-firing limit comes to 0.015%, based on 135 SRM firings during the design lifetime.

Thermal tests indicate that the Orbiter tiles can sustain 10% erosion without a measurable degradation of their thermal radiation properties, and this value is currently being used as a lifetime limit for tile erosion. Based on 135 SRM firings during the design lifetime, the per-firing limit on tile erosion is 0.074%.

Table A-1. Current SRM Particle Damage Limits for Flight Design

ORBITER SURFACE TYPE	EROSION (% OF SURFACE AREA)		BREAKAGE ("BREAKS" PER SQUARE FOOT)	
	LIFETIME LIMIT	PER-FIRING LIMIT†	LIFETIME LIMIT*	PER-FIRING LIMIT†
WINDOW	2	0.015	9.7	0.072
LO-TEMP THERMAL PROTECTION TILE	10	0.074	1.6	0.012
HIGH-TEMP THERMAL PROTECTION TILE	10	0.074	0.9	0.007

*EQUAL TO EXPECTED BREAKAGE FROM MICRO-METEORIDS DURING 421 DAYS ON ORBIT
(SEE REFERENCE A-1)

†BASED ON 135 SRM FIRINGS DURING LIFETIME

A.4. ON-ORBIT DAMAGE POTENTIALS

For a given particle type and size, the instantaneous rates of erosion and breakage on an element of the Orbiter's surface can be expressed in terms of the local flowfield mass density of the particles, their speed relative to the surface under consideration, and their incidence angle ϕ (i.e., the angle between the surface normal and the relative velocity vector of the particles).

With regard to incidence angle, the breakage rate varies as $\cos \phi$, and the erosion rate varies as $\cos^3 \phi$ (see Table 6-1 of Reference A-1). Obviously, at any instant the value of ϕ will be different at different points on the Orbiter's surface, and the value at a given point is a function of the Orbiter's attitude. It simplifies the problem of estimating surface damage to consider, instead of the whole surface of the Orbiter or any particular point on it, a hypothetical surface element which is always normal to the relative velocity vectors of the impinging particles (of whatever size and at whatever time). When the breakage or erosion rate for such a hypothetical surface is integrated with respect to time and with respect to particle size, the result can be thought of as a "potential" for the type of damage and the type of surface under consideration. The potential represents the maximum real damage that a surface of the given type would incur if the Orbiter attitude were such as to expose it continuously to particle impingement during the period (sometimes lasting several minutes beyond SRM burnout) when the local flux of exhaust particles is significant.

Even with the simplification that results from assuming $\cos \phi = 1$, the problem of calculating and integrating the damage rates is not an easy one. Basically, the difficulty arises because (1) the damage rate equations are necessarily referenced to a coordinate system in which the Orbiter is stationary, (2) the equations which define the exhaust plume (particle velocity and flowfield mass density) are necessarily referenced to a coordinate system in which the SRM is instantaneously stationary, and (3) the two systems are in a state of accelerated motion relative to each other during the SRM burn. The problem is further complicated by the facts that (a) the particle damage rates must be integrated with respect to impact time, (b) the SRM exhaust flow rate is a function of ejection time, and (c) the time interval between ejection and impact is significant and variable. A detailed discussion of this problem, and its resolution, is given in Reference A-3.

In the computer programs that are used to calculate damage potentials, the trajectories of the Orbiter and the upper stage are obtained by integrating equations of motion which include the effects of gravity as well as SRM thrust. However, the motions of individual exhaust particles (from ejection to impact time) are assumed to be linear at constant velocity in a coordinate system wherein the Orbiter is stationary. Such a simplification is practically necessary because of the great number of particle trajectories that have to be considered. If the frame of reference is not rotating with respect to inertial space, it is reasonably accurate for particle flight times up to 5 or 10 minutes between ejection and impact, provided aerodynamic drag can be ignored (see Table A-2).*

Particle velocities relative to the Orbiter are obtained by adding, vectorially at the time of ejection, their SRM-relative velocities (see Figure A-1) to that of the SRM itself. As illustrated in Figure A-4, the effect of SRM thrust acceleration is to continuously modify the directions and magnitudes of the Orbiter-relative particle velocity vectors as the burn progresses. The Orbiter is to be considered stationary in Figure A-4, conceivably at any position in the frame of reference. It is obvious that only those particles whose velocity vectors point toward the Orbiter can damage it, and therefore that the integrated damage potential is highly dependent on the Orbiter's position relative to the SRM line of flight. The diagram shown in Figure A-4 applies specifically to $0.18 \mu\text{m}$ aluminum oxide particles, which are particularly effective in producing erosion. Considering the fact that the number flux for these particles is greatest on the plume centerline and falls off significantly as the ejection angle (relative to the SRM) approaches its limit, it is easy to predict that the erosion potential should be greater if the Orbiter were behind the SRM at ignition time than if it were ahead and off to one side of the powered flight path.

*The accelerations shown in Table A-2 indicate that the trajectories of the smallest particles can be affected significantly by drag. However, the net effect on the damage potentials is not very great because (1) as shown in Figure A-3, the mass fraction of particles smaller than $0.1 \mu\text{m}$ is essentially negligible, (2) most of the smaller particles have airspeeds considerably lower than orbital velocity; and, (3) the ejection-to-impact flight time for the smaller particles is generally less than two minutes. In addition, the atmospheric density assumed for the construction of Table A-2 is near the upper limit for the indicated altitude. Depending on solar activity and the position of the SRM burn with respect to the subsolar point, the actual density could be smaller by as much as an order of magnitude.

Table A-2. Particle Acceleration Due to Aerodynamic Drag

	PARTICLE DIAMETER (μm)	AERODYNAMIC DRAG ACCELERATION* (m/s^2)
ALUMINUM OXIDE	.0178 .0316 .0562 .100 .178 .316 .562 1.00 1.78 3.16 5.62 10.0	49.1 27.7 15.6 8.74 4.91 2.77 1.56 .874 .491 .277 .156 .0874
CARBON	17.8 31.8 56.2 100 178 316 562 1000	.0928 .0523 .0294 .0165 .00928 .00523 .00294 .00165

*ASSUMPTIONS:

$$C_D = 2.0$$

particle airspeed = 7732 m/s
(circular orbit velocity at 296 km)
atmospheric density = $3.87 \times 10^{-14} \text{ g/cm}^3$
(1962 standard atmosphere)

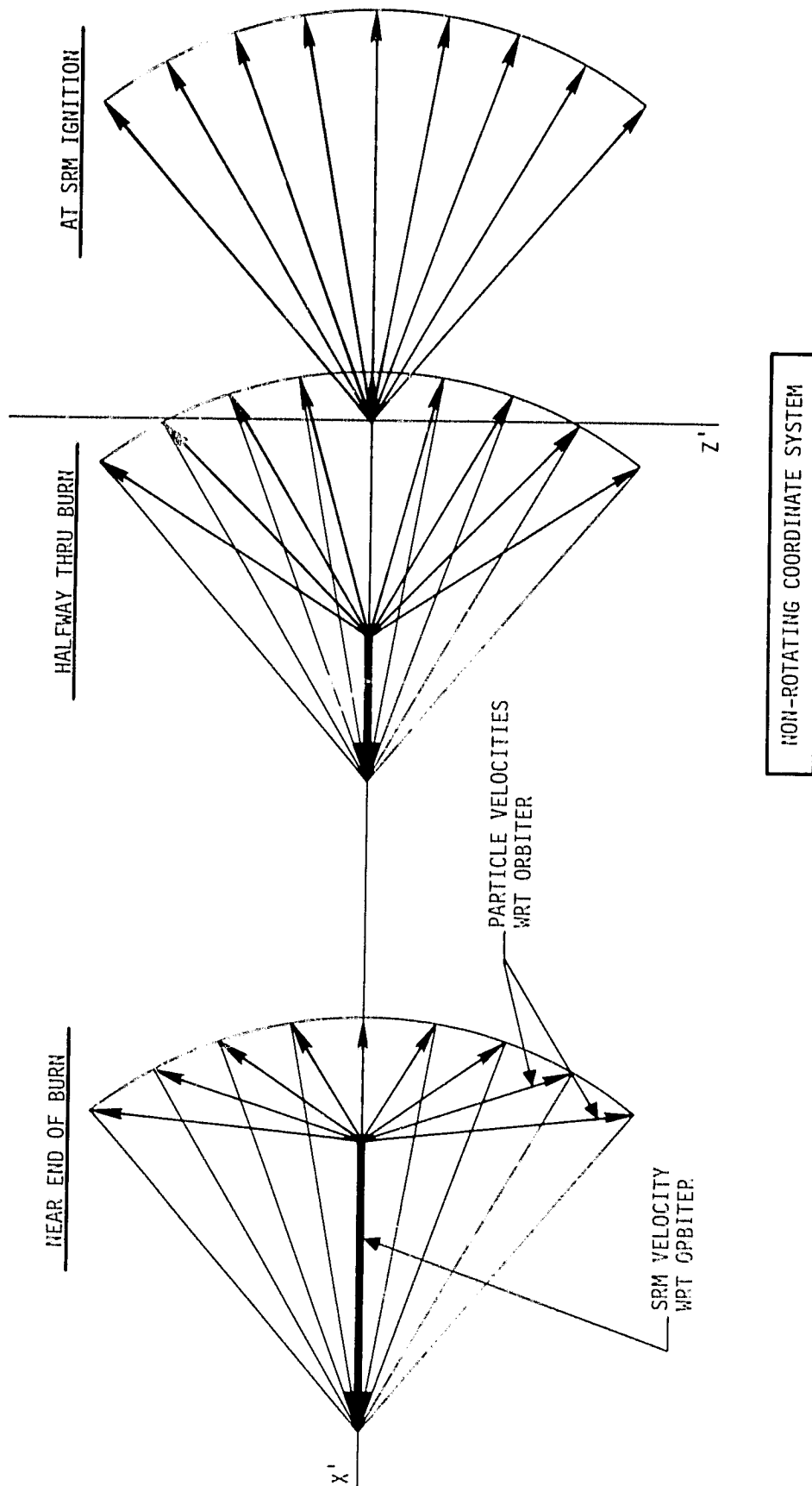


Figure A-4. Effect of IUS Motion on 0.18 μm Aluminum Oxide Particle Velocities with Respect to Orbiter

Figure A-5 is analogous to A-4, except that it applies to the "carbon" particles, whose SRM-relative velocities (again, see Figure A-1) are smaller and distributed annularly about the plume centerline. As can be seen by comparing Figures A-4 and A-5, the effect of SRM thrust acceleration is quite different in this case. Long before SRM burnout, the upper stage has gained enough velocity to give significant forward components to the Orbiter-relative velocity vectors of the larger particles, and velocity magnitudes far exceeding what they were at ignition time. As previously pointed out, these particles are the primary agents of breakage. Therefore it is easy to predict that (in contrast to erosion) the breakage potential would be rather severe if the Orbiter were ahead and off to one side of the SRM powered flight path.

The qualitative inferences drawn from Figures A-4 and A-5 are borne out by the quantitative results of damage-potential surveys in the upper stage deployment orbit plane, which are contained in Reference A-1. The results of the survey, as they apply to the erosion potential of a SSUS-D firing, are depicted graphically in Figure A-6.

It should be noted that the coordinate axes shown in Figure A-6 are different from those shown in Figures A-4 and A-5. The origin of the local vertical/local horizontal (LVLH) axes depicted in Figure A-6 (and other illustrations to follow) is centered on a phantom satellite in circular earth orbit, whose position coincides with the upper stage prior to SRM ignition, but which does not experience SRM thrust acceleration. The Z axis of the LVLH system points continuously at the center of the earth (which means that the system is rotating), the Y axis (not shown) is normal to the geocentric orbit plane, and the X axis completes a right-handed orthogonal triad. Following SRM ignition, the upper stage accelerates away from the origin. Assuming a posigrade burn (i.e., with the thrust vector pointed in the direction of orbital motion) the SRM powered flight path coincides initially with the LVLH X axis, curving upward as the burn progresses. Exhaust particle trajectories in the LVLH system are also curved. However, over the time intervals of interest here, the curvature is small enough to permit a mental superposition of the velocity diagrams of Figures A-4 and A-5 onto the LVLH frame without introducing gross qualitative errors.

Conceivably, the Orbiter could be located anywhere in the LVLH coordinate system of Figure A-6 at SRM ignition time. The laws of orbital mechanics forbid its being stationary anywhere except possibly on the X axis. Again, however, for

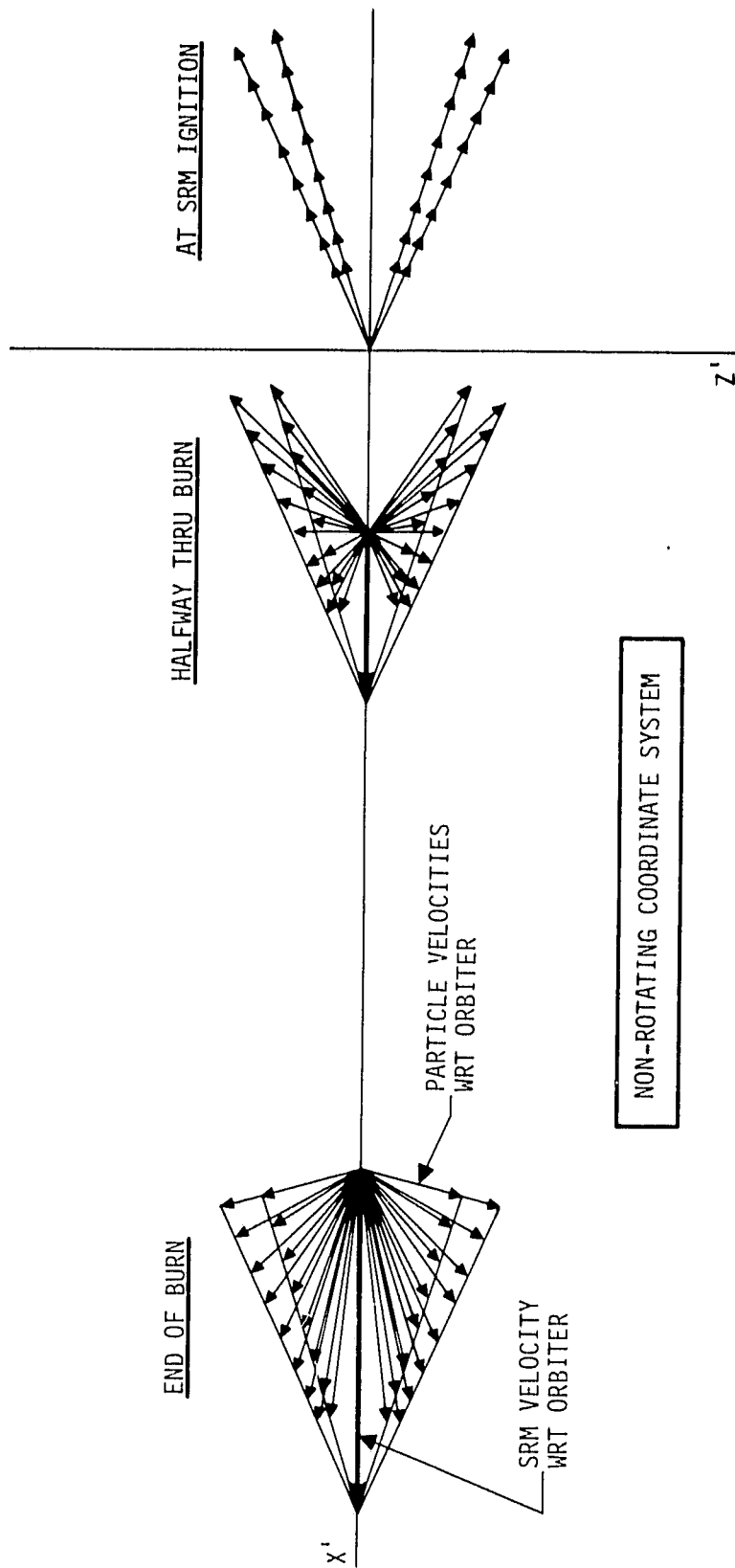


Figure A-5. Effect of IUS Motion on Carbon Particle Velocities with Respect to Orbiter

SSUS-D IGNITION CONDITIONS

PROPELLANT LOAD = 3700 LB

GROSS WEIGHT = 6188 LB

PITCH = -4.2 DEG, YAW = 0

ORBITER ASSUMED TO BE
IN CIRCULAR ORBIT

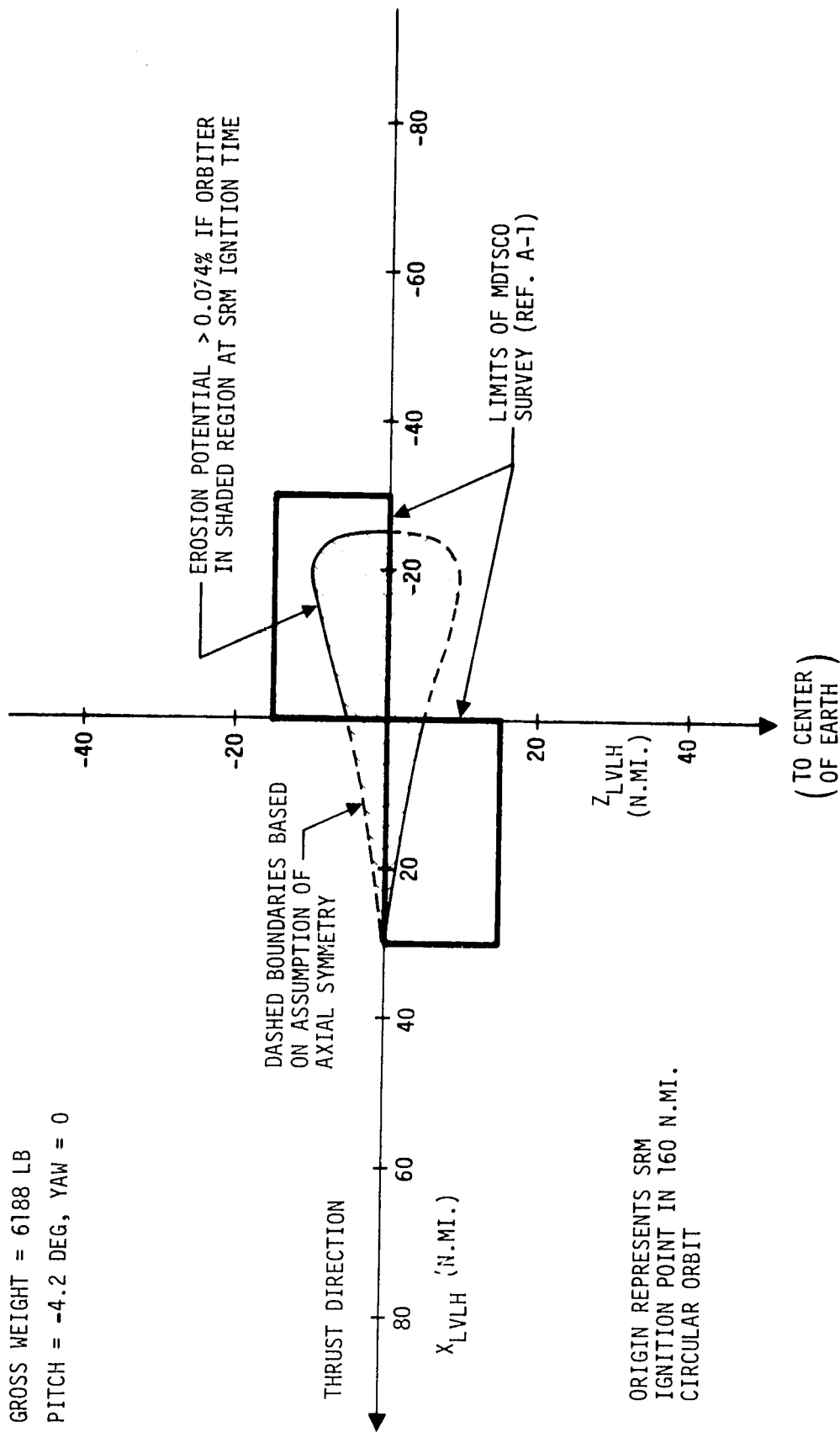


Figure A-6. Region of Excessive Tile Erosion Potential in SSUS-D Deployment Orbit Plane

the purpose of transferring the qualitative inferences drawn from Figures A-4 and A-5 to the rotating coordinate system, it is not grossly erroneous to think of the Orbiter as being stationary in the LVLH frame.

According to the survey results contained in Reference A-1, the surface erosion potential would exceed the flight design limit for thermal protection tiles (see Table A-1) if the Orbiter were to be located anywhere inside the shaded region of Figure A-6 at SSUS-D SRM ignition time. Similarly, the breakage potential for high-temperature tiles would exceed flight design limits if the Orbiter were located anywhere inside the shaded region of Figure A-7 at ignition time. The shapes of these two excessive-damage regions are consistent with the vector diagrams shown in Figures A-4 and A-5.

Figure A-8 contains the same type of information as that shown in Figures A-6 and A-7, except that the upper stage under consideration here is the SSUS-A, and the regions of excessive tile erosion and excessive high-temperature tile breakage are superimposed on each other in a single display. Figure A-9 is similar to A-8, except that it applies to the IUS first-stage (large) SRM. Comparing Figures A-6 through A-9, the progressive increase in the sizes of the excessive-damage regions is due mainly to the increase in mass of propellant expelled by the different SRMs.

The damage potentials represented in Figures A-6 through A-9 were obtained by systematically varying the Orbiter's position relative to the upper stage at SRM ignition time. For each initial position, damage potentials were obtained by integrating the appropriate damage rates with respect to time and with respect to particle size. The results were then mapped as functions of the Orbiter's initial position.

The reason for showing only the breakage potentials that apply to the Orbiter's high-temperature tiles can be explained as follows. According to the survey results given in Reference A-1, the potentials for window breakage are so great that it seems clear there is only one practicable way to satisfy the damage-limitation criteria given in Section A.3: the Orbiter must be oriented such that its windows will be shielded from SRM exhaust particles. Any orientation that shields the windows effectively shields the low-temperature tiles as well. It follows then that the basic trajectory design requirement, insofar as the limitation of damage to the Orbiter is concerned, is to maneuver the Orbiter to a

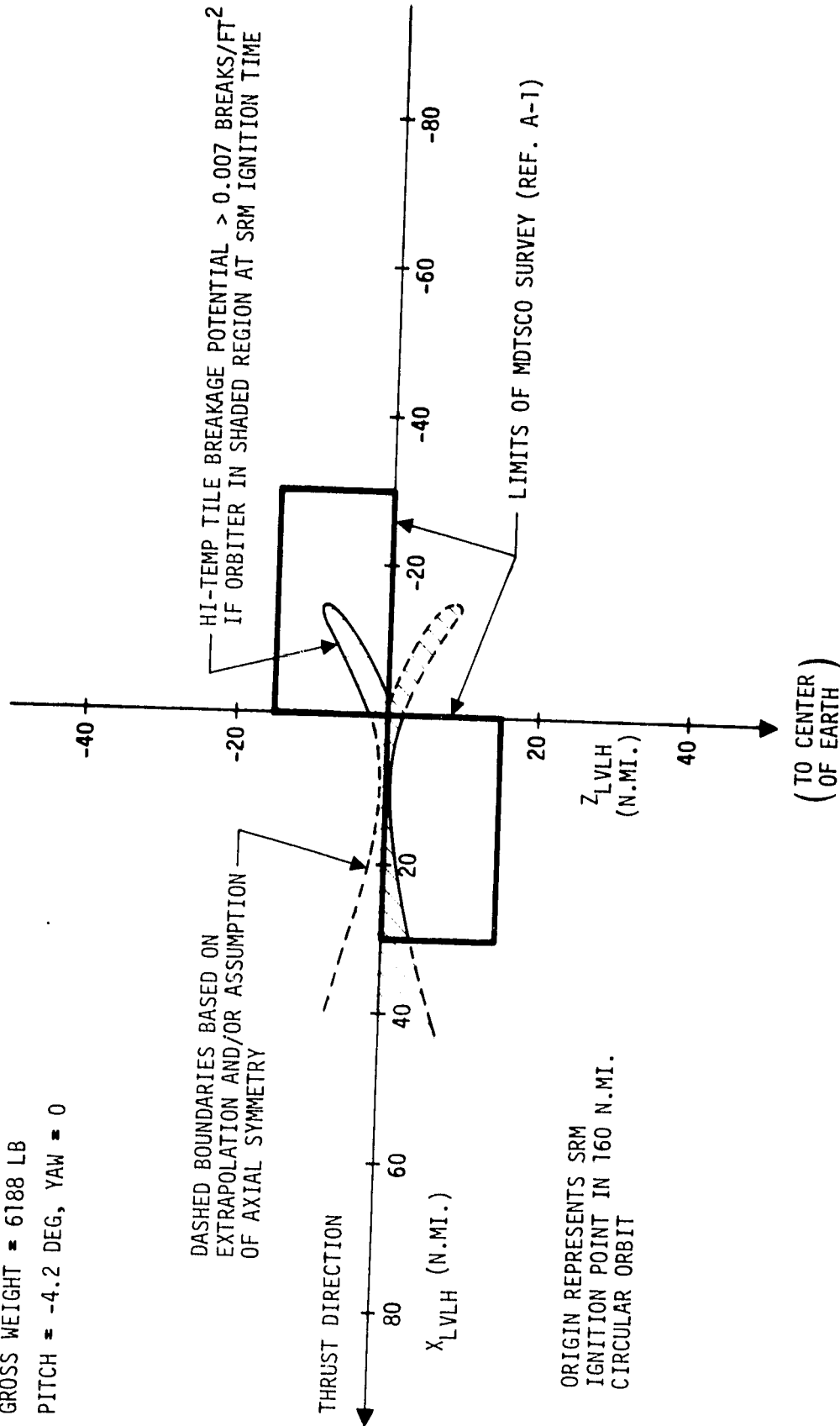
SSUS-D IGNITION CONDITIONS

PROPELLANT LOAD = 3700 LB

GROSS WEIGHT = 6188 LB

PITCH = -4.2 DEG, YAW = 0

ORBITER ASSUMED TO BE IN
CIRCULAR ORBIT



A-18

Figure A-7. Region of Excessive hi-Temp Tile Breakage Potential in SSUS-D Deployment Orbit Plane

SSUS-A IGNITION CONDITIONS

PROPELLANT LOAD = 7646 LB

GROSS WEIGHT = 12744 LB

PITCH = -4.2 DEG, YAW = 0

ORBITER ASSUMED TO BE
IN CIRCULAR ORBIT

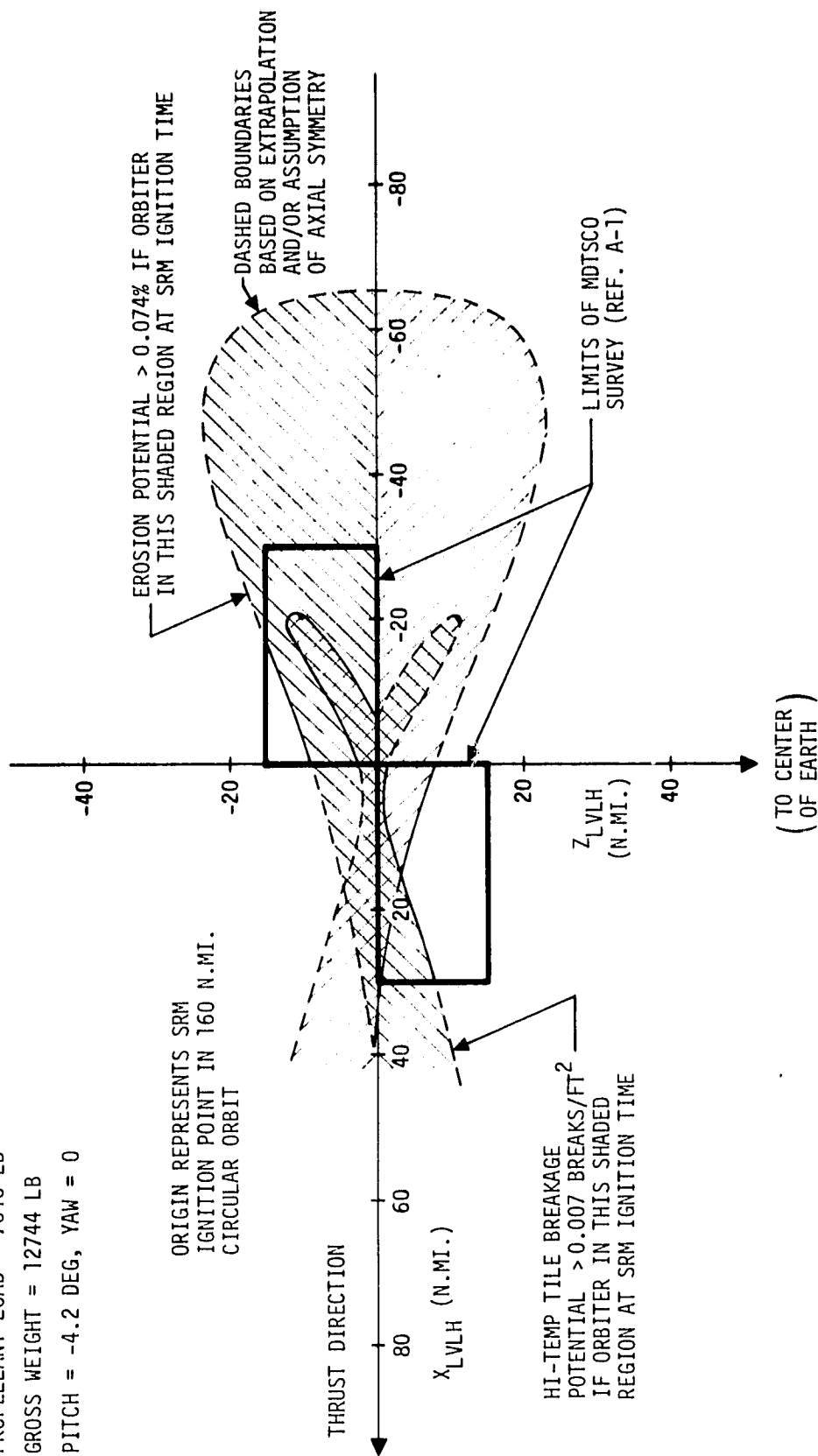


Figure A-8. Regions of Excessive Hi-Temp Tile Damage Potential in SSUS-A Deployment Orbit Plane

IUS IGNITION CONDITIONS

PROPELLANT LOAD = 21400 LB

GROSS WEIGHT = 34026 LB

PITCH = -4.2 DEG, YAW = 0

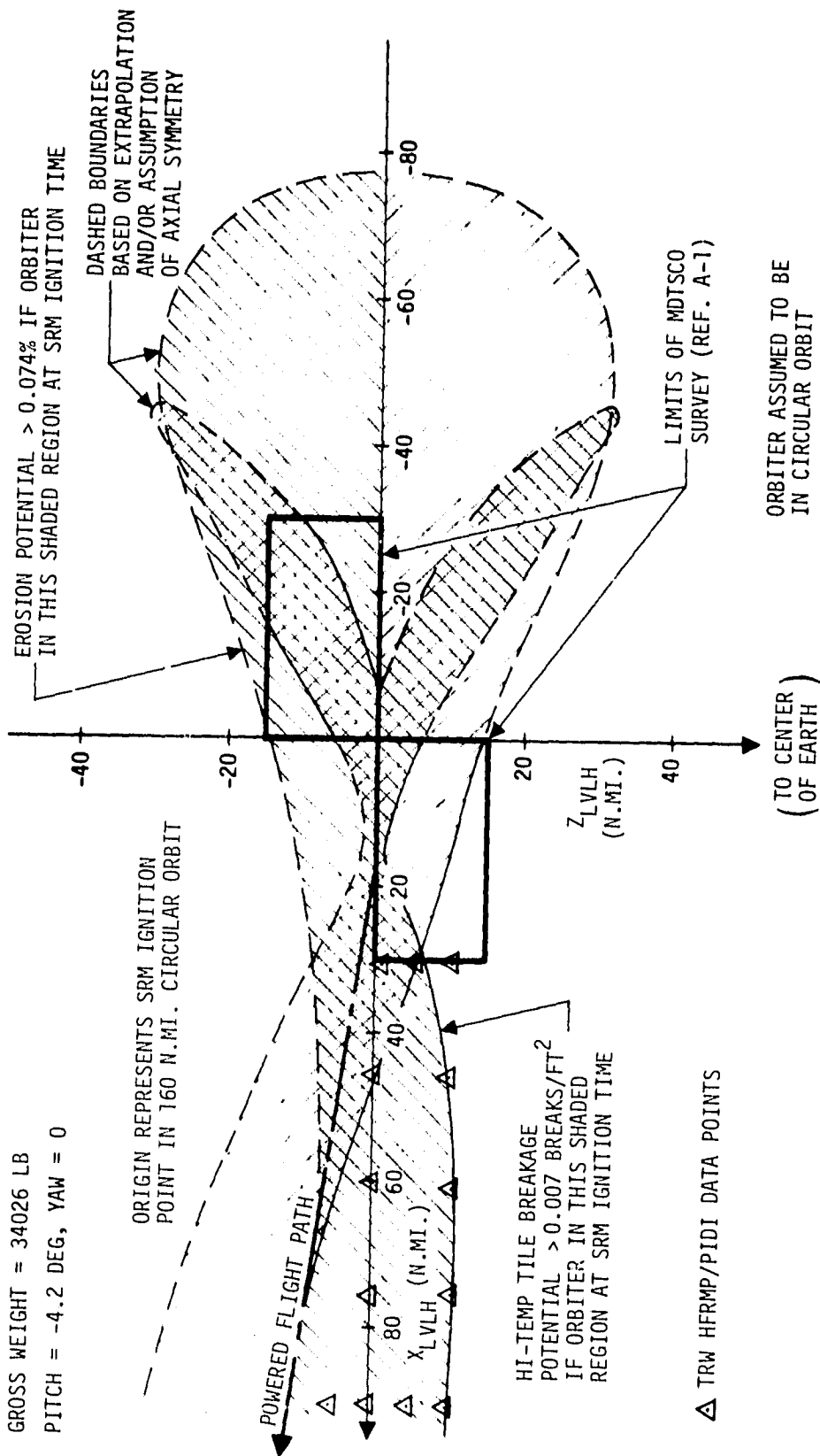


Figure A-9. Regions of Excessive Hi-Temp Tile Damage Potential in IUS Deployment Orbit Plane

position (relative to the upper stage) where the high-temperature tile damage potentials are acceptable.

It needs to be pointed out that the quantitative results shown in Figures A-6 through A-9 do not involve an assumption that the Orbiter is stationary in the LVLH frame. On the contrary, in the process of integrating the damage rates, the real motion of the Orbiter was calculated in accordance with the laws of orbital mechanics and the assumed state of the Orbiter's motion at SRM ignition time. However, for the purposes of the survey, the Orbiter's state of motion was arbitrarily assumed to be that corresponding to a geocentric circular orbit at the appropriate altitude (i.e., at 160-Z nautical miles). The point is that for any real in-flight situation, the Orbiter almost certainly will not be in a circular orbit at SRM ignition time, and therefore the results shown in Figures A-6 through A-9 should not be expected to apply precisely to real Orbiter/upper-stage separation sequences. That is to say, the damage potentials depend not only on the Orbiter's position at SRM ignition time, but also on its velocity.

In addition to the Orbiter's velocity at SRM ignition time, other parameters that can have a significant effect on the damage-potential contours include the SRM propellant load, the upper-stage payload weight, and of course the SRM thrust direction with respect to the LVLH frame. Reverting again to qualitative inferences, it seems reasonable that (in 3-dimensional space) the regions of excessive damage should look something like bodies of revolution about the SRM thrust axis. The shaded regions shown in Figures A-6 through A-9 can then be thought of as plane sections through these bodies of revolution. The gross effects of variations in thrust direction can be visualized by rotating the appropriate contours bodily about the origin of coordinates.

It must be emphasized that the supposed axial symmetries referred to in the preceding paragraph can be only qualitative at best, and that the accuracy of the attendant approximations has not been established. It is also emphasized that significant portions of the damage-potential contours (all of the dashed-line segments) shown in Figures A-6 through A-9 are based on the symmetry assumption and on the extrapolation of available data, in some cases far beyond the limits of the quantitative survey. In particular, the accuracy of the extrapolated tile breakage potential in the right-hand quadrants of Figure A-9 is very questionable.

A.5. REGIONS OF LVLH SPACE ACCESSIBLE TO ORBITER

To attain a position where the high-temperature tile damage potentials are acceptable, it is necessary to apply a major velocity increment to the Orbiter soon after the upper stage is ejected from the cargo bay. For reasons which are discussed in Section 2.4 of the main body of this report, it is generally not feasible to execute the Orbiter's main separation maneuver earlier than about 9 minutes after ejection. Distances traversed by the Orbiter prior to the major separation maneuver are insignificant on the scale of Figures A-6 through A-9. Therefore, for the purpose of preliminary damage-limitation analysis, the Orbiter's motion relative to the upper stage can be ignored during the first 9 minutes following stage ejection. Given a fixed interval of 45 minutes between ejection and SRM ignition, the available coast time in the large-scale separation trajectory is approximately 36 minutes.

The position reached by the Orbiter at the end of any particular coast interval depends on the orientation and magnitude of the main separation velocity increment ($\overline{\Delta V}$). For a given velocity increment magnitude, maximum separation is achieved when the $\overline{\Delta V}$ vector (hence also the resulting separation trajectory) lies in the plane of the geocentric deployment orbit.

Figure A-10 shows, as a function of coast time, the locus of all in-plane positions the Orbiter can reach by varying the direction of a $\overline{\Delta V}$ vector whose magnitude is fixed arbitrarily at a value of 60 feet per second. This figure indicates that nothing significant in the way of separation distance can be gained by coasting less than 36 minutes.

When the coast time and the direction of the $\overline{\Delta V}$ vector are held fixed, the orientation of the Orbiter's relative position vector at SRM ignition time remains constant and its magnitude is directly proportional to ΔV (the magnitude of the $\overline{\Delta V}$ vector). Therefore, variation of ΔV has the effect of expanding or contracting the outermost ellipse shown in Figure A-10, without changing its shape or its orientation. This effect is illustrated in Figure A-11. The radial lines emanating from the origin in Figure A-11 do not represent trajectories. Instead, they represent the loci of trajectory end points (at the end of a 36-minute coast) which can be attained by holding the direction of the velocity increment fixed and varying its magnitude.

There are two characteristics of the Orbiter accessibility contours that need to be emphasized. The first is that two distinct maxima exist when separation

ROTATING LVLH COORDINATE SYSTEM,
UPPER STAGE AT ORIGIN

$\Delta V = 60 \text{ FT/SEC}$

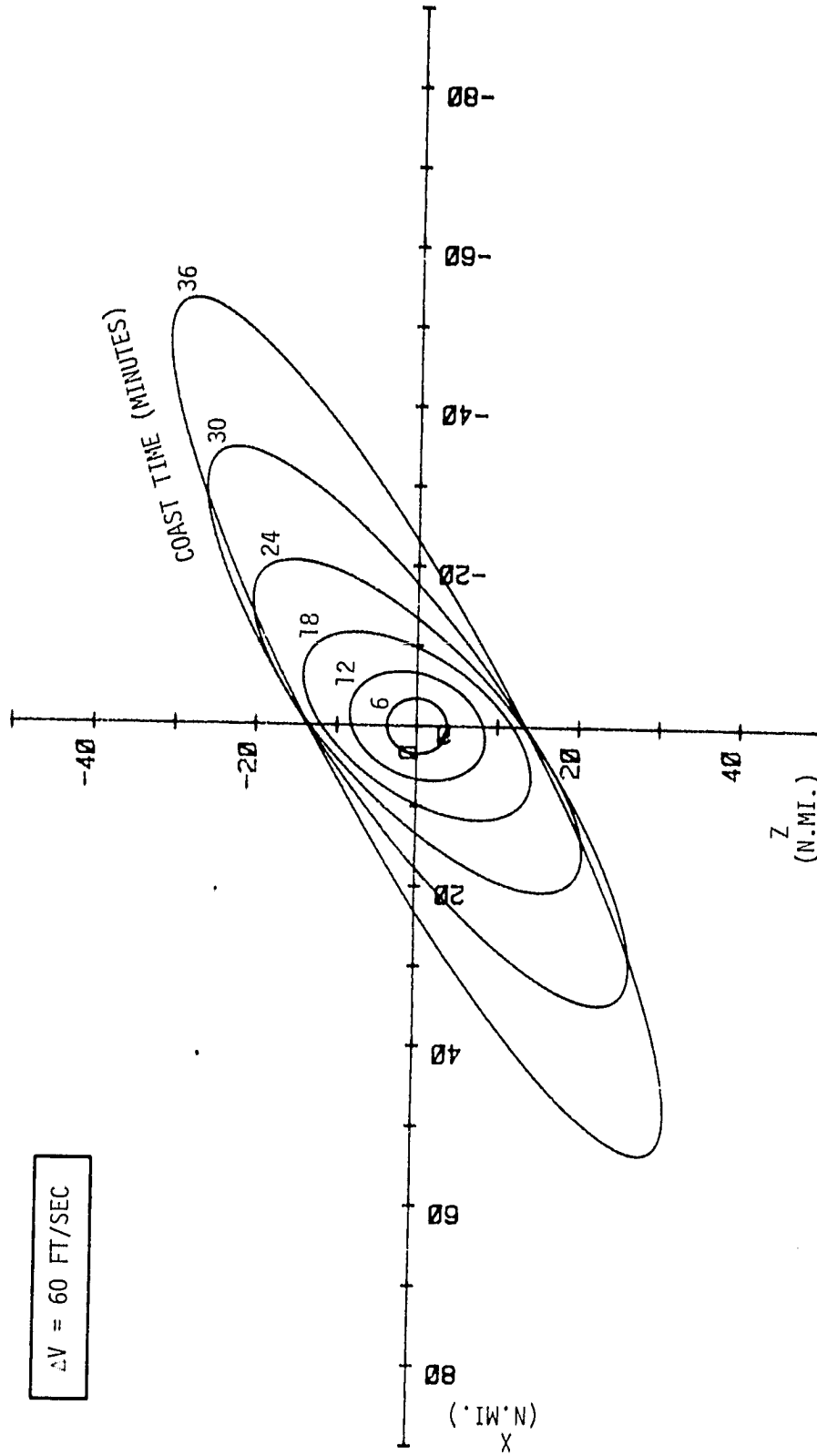


Figure A-10. In-Plane Positions Accessible to Orbiter,
with a Fixed ΔV Magnitude

ROTATING LVLH COORDINATE SYSTEM,
UPPER STAGE AT ORIGIN

COAST TIME = 36 MINUTES

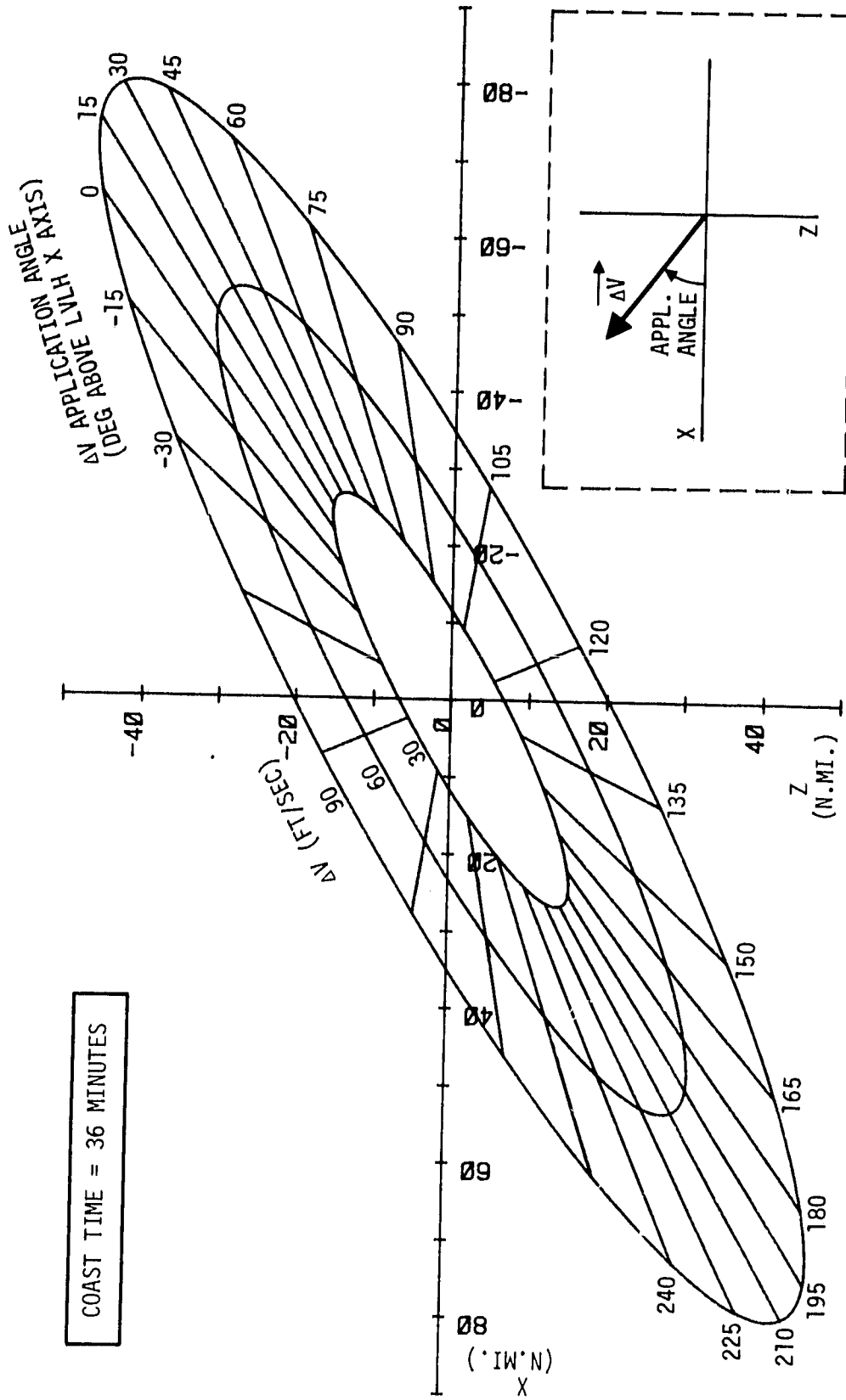


Figure A-11. In-Plane Positions Accessible to Orbiter,
With a Fixed Coast Time

distance is considered as a function of the direction of the ΔV vector. One maximum is achieved with an optimum ΔV application angle (measured as shown in Figure A-11) of approximately 30 degrees, and the other at another optimum application angle of approximately 210 degrees. The second characteristic, which is particularly important with respect to the trajectory design problem, is that the final position of the Orbiter (at SRM ignition time, of course) is relatively insensitive to deviations on the order of 20 degrees or so away from either of the two optimum directions.

A.6. SELECTION OF NOMINAL LVLH SEPARATION TRAJECTORY

We have seen in the previous section that the overall trajectory design problem divides itself naturally into two distinct chronological phases. During the first phase (which is the subject of the main body of this report), significant distances are measured in feet and the Orbiter's body-fixed axes represent the most pertinent system of reference. During the second phase, significant distances are measured in miles and the LVLH axes originating at the position of the upper stage represent the most pertinent system of reference. The velocity increment (ΔV) produced by the Orbiter's main separation maneuver is the one and only trajectory parameter that is of major significance in both phases*.

As so often is the case in trajectory design, the overall problem is most easily solved by working backward through the chronological sequence of events. Specifically, the question to be resolved now is: Where should the Orbiter be located at the time of SRM ignition? The answer will define the large-scale LVLH trajectory, which will define the direction and magnitude of the main separation velocity increment, which then will become a major input to the detailed design of the first (9-minute) phase of the separation sequence.

The magnitude of the separation velocity increment is roughly proportional to the amount of Orbiter propellant required to execute the separation sequence. Therefore, minimization of ΔV is the primary criterion for selecting a nominal LVLH trajectory, subject of course to the requirement for maneuvering the Orbiter to a position outside the excessive-damage region for the stage type under consideration, as defined approximately in Figures A-6 through A-9.

Figure A-12 is presented as a graphical aid for discussing the factors which relate to the selection of a nominal separation ΔV vector. It represents a superposition of the Orbiter accessibility contours (previously shown in Figure A-11) onto an extended region (the shaded area in Figure A-12) which is swept out when

*In this respect the current problem is similar to that of designing an interplanetary trajectory, where the hyperbolic excess velocity vector $\bar{v}_\infty$ (analogous to ΔV) is the major interface parameter between the planetocentric and heliocentric phases of flight.

ROTATING LVLH COORDINATE SYSTEM,
UPPER STAGE AT ORIGIN

COAST TIME = 36 MINUTES

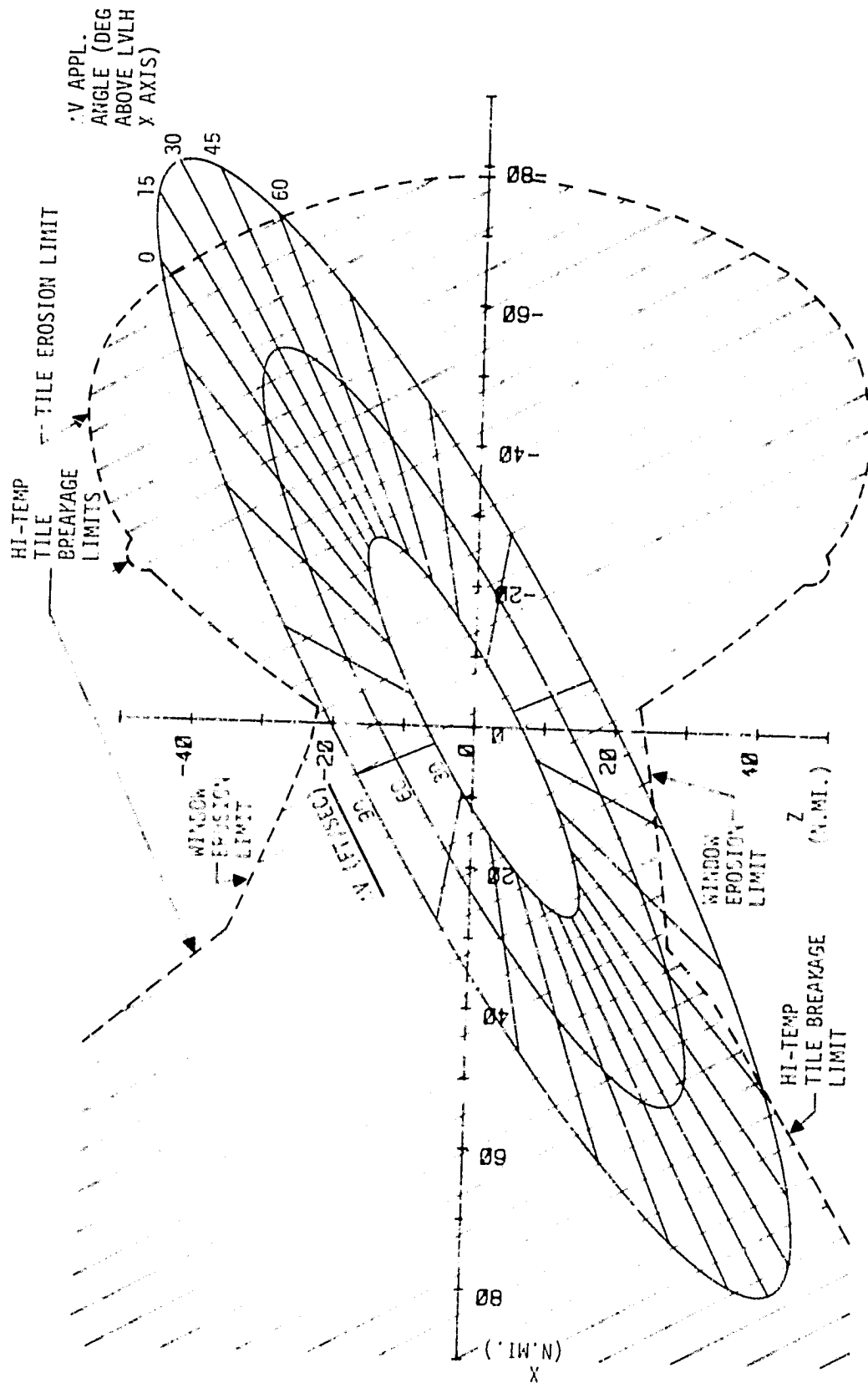


Figure A-12. US Excessive Damage Region (with ± 30 Degrees Pitch Variation)
Superimposed on Orbiter Accessibility Contours

the appropriate excessive-damage contours* for the IUS are rotated bodily through angles of 30 degrees about the origin of coordinates.

There are two reasons that a range of possible upper-stage thrust directions need to be considered. First of all, there is always the possibility that the upper-stage guidance and control system could malfunction. Of course, in such a case there is no reason to think that the error in thrust direction would be limited to the 30 degree range represented in Figure A-12. However, in the event of a malfunction it is reasonable to assume that small errors are more likely than large ones. By mentally picturing the result of extending the range of variation in thrust direction (beyond that shown in Figure A-12), it is readily apparent that, for a given ΔV magnitude, the Orbiter will achieve the safest possible position if the ΔV is applied at an angle of approximately 30 degrees above the LVLH axis. This places the Orbiter near the vertex of the accessibility ellipse, in the upper right quadrant of Figure A-12. Assuming the nominal SRM thrust direction is prograde, and that the magnitude of the separation ΔV is approximately 75 feet per second, at this position the Orbiter would be safe from any IUS orientation error of lesser magnitude than about 130 degrees.

The second reason for considering a range of SRM thrust directions (as opposed to one single direction) has to do with SRM energy management and flight phase standardization. Depending on the weight of the payload and its destination, significant variations in the nominal thrust direction may be required from one flight to another. Such variations are necessary in order to match upper stage trajectory requirements with the fixed total impulse of the SRM. According to Reference A-4, DOD payloads may require a flight-to-flight variation of 30 degrees in the nominal pitch angle of the IUS first-stage burn. Although for many specific flights it might be acceptable (if the possibility of an upper-stage malfunction could be ignored) to maneuver the Orbiter into the lower left quadrant of Figure A-12, each case would have to be analyzed individually. In line with the concept of standardization, it is deemed preferable to adopt a policy of always maneuvering the Orbiter into the right hand quadrant (i.e., to a position above and behind the upper stage), thereby tending to minimize the requirement for flight-specific analysis and training.

*Including window erosion (not shown in figure A-9) in the forward quadrants only. The reason for this is that, in the forward quadrants, the differences in the Orbiter-relative velocity vector directions of aluminum oxide and carbon particles (see Figures A-4 and A-5) would make it difficult or impossible to shield Orbiter windows from both erosion and breakage.

As a nominal position for the Orbiter at SRM ignition time, the upper right vertex of the accessibility ellipse offers not only the advantage of minimum Orbiter propellant requirement with minimum sensitivity to SRM thrust direction, but also minimum sensitivity to the direction of the separation velocity increment supplied by the Orbiter. This is extremely important because it affords a significant relaxation of the Orbiter's attitude constraints during the initial (9-minute) phase of the separation sequence. By examining Figure A-12, it can be seen that if the cited vertex is the Orbiter's target point, a deviation of 20 degrees or so from the optimum $\overline{\Delta V}$ direction (which is 30 degrees above the LVLH X axis) does not produce a very great change in the Orbiter's position at SRM ignition time. It can also be seen that, given precise execution of the separation maneuver (and assuming again that an upper stage malfunction does not have to be allowed for), there is an acceptable target region in the lower left quadrant which conceivably could be achieved with less propellant consumption on the part of the Orbiter. However, the accessibility ellipse is rather flat in this latter region, and a deviation of 20 degrees in the $\overline{\Delta V}$ direction would be intolerable. Aside from other implications, targeting to this region would require the design of (and crew training for) probably at least twice as many standard separation sequences as are defined in the main body of this report.

When the damage-potential contours for the SSUS-D and SSUS-A stage types are subjected to the kind of analysis described in the preceding paragraphs, the conclusion is essentially the same as it was in the case of the IUS, viz: given a 36-minute coast period, the optimum direction for the $\overline{\Delta V}$ produced by the Orbiter's major separation maneuver is approximately 30 degrees above the LVLH X axis. The magnitude of the required separation velocity increment is on the order of 25 feet per second in the case of SSUS-D, 65 feet per second in the case of SSUS-A, and 75 feet per second in the case of IUS. These figures are only approximate, being subject to certain assumptions and possible extrapolation errors which are discussed in Section A.4. Although the ΔV magnitudes may change as a result of (1) more extensive surveys of the type reported in Reference A-1, (2) more accurate SRM exhaust models, or (3) relaxation of the basic damage limitation criteria in Table A-1, only a change in the available coast time (from the assumed value of 36 minutes) is likely to alter the conclusion regarding the optimum $\overline{\Delta V}$ direction.

A.7. POSSIBLE RECONVERGENCE OF SRM EXHAUST PARTICLES

To date, in all of the numerical analyses of potential damage to the Orbiter from SRM exhaust particles, it has been tacitly assumed that the particles would continue to diverge indefinitely along linear trajectories after being expelled from their source (the rocket motor). This is a reasonable assumption in the case of the smaller (aluminum oxide) particles ejected during a posigrade burn when the inertial velocity of the upper stage never exceeds that which is required for a transfer to geosynchronous orbit altitude, because virtually all of the particles are given inertial velocities so small (see Figure A-4) that they are bound to enter the atmosphere before completing a full revolution in orbit. However, some of the heavier particles will be given inertial velocities sufficient to sustain them in orbit for several revolutions (see Figure A-5), and the same may be true of some aluminum oxide particles ejected during an interplanetary injection burn, or during a burn which is directed at right angles to the orbit plane.

The laws of orbital mechanics forbid the indefinite divergence of those particles which remain in orbit. In the absence of perturbations due to aerodynamics, gravitational harmonics, etc., it is easily deduced that those particles which remain in orbit must reconverge periodically whence they were ejected; i.e., on the geocentric inertial track which was traversed by the upper stage during the SRM burn. Conceivably then, if on a subsequent orbital revolution the Orbiter should happen to pass close enough to the inertial track of the SRM burn, it could be subjected to a significant flux of particles. Recognizing the facts that (1) perturbations will modify the particle orbits, (2) most of the particles will not survive their first perigee passage, and (3) not all of the surviving particles can reconverge simultaneously (i.e., that there must be a continuous variation of particle orbit periods), it is clear that the flux at a given position relative to the SRM powered flight track would have to be much weaker in such a case than it was during the primary period of concern (i.e., during and immediately after the time of the burn). However, it is also clear that in such a case the flux would be bi-directional (convergent/divergent), making it almost certain that the Orbiter windows would be subjected to it, no matter what the orientation of the Orbiter at the time. Given the tremendous potential of the original flux for causing window breakage (orders of magnitude greater than can be tolerated even at distances on the order of tens of miles from the SRM powered flight path), the problem is not obviously negligible.

There is one more aspect of the reconvergence problem that does seem to be clear: aerodynamic drag will cause the particle orbits to decay. Therefore, in the long term at least, the surviving particles should be expected to reconverge at altitudes progressively lower than the track of the SRM burn. This is considered to be another point in favor of maneuvering the Orbiter to a position above (and behind) the upper stage at SRM ignition instead of below (and ahead of) it. However, the most that can be said at the present is that this is believed to be a relative advantage, and even if it does tend to minimize the problem, it does not necessarily avoid it entirely.

It is not immediately obvious how a reliable numerical estimate of the delayed-action potential for window breakage can be obtained, but this is a problem that should be resolved.

A.8 REFERENCES

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