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Hazard Analysis of Commercial Space Transportation

Volume I: Operations
Volume II: Hazards
Volume III: Risk Analysis

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Cover illustration:
Near Earth Satellite Population of July 1, 1987
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EXECUTIVE SUMMARY

This report, entitled "Hazard Analysis of Commercial Space Transportation," is devoted to the review and discussion of generic hazards associated with the ground, launch, orbital and re-entry phases of space operations. Since the DOT Office of Commercial Space Transportation (OCST) has been charged with protecting the public health and safety by the Commercial Space Act of 1984 (P.L. 98-575), it must promulgate and enforce appropriate safety criteria and regulatory requirements for licensing the emerging commercial space launch industry. This report was sponsored by OCST to identify and assess prospective safety hazards associated with commercial launch activities, the involved equipment, facilities, personnel, public property, people and environment. The report presents, organizes and evaluates the technical information available in the public domain, pertaining to the nature, severity and control of prospective hazards and public risk exposure levels arising from commercial space launch activities. The US Government space-operational experience and risk control practices established at its National Ranges serve as the basis for this review and analysis.

The report consists of three self-contained, but complementary, volumes focusing on Space Transportation: I. Operations; II. Hazards; and III. Risk Analysis. This Executive Summary is attached to all 3 volumes, with the text describing that volume highlighted.

Volume I: Space Transportation Operations provides the technical background and terminology, as well as the issues and regulatory context, for understanding commercial space launch activities and the associated hazards. Chapter 1, The Context for a Hazard Analysis of Commercial Space Activities, discusses the purpose, scope and organization of the report in light of current national space policy and the DOT/OCST regulatory mission. It also introduces some basic definitions and outlines the approach to a generic Hazard Analysis for future commercial space operations. Chapter 2, Range Operations, Controls and Safety, discusses the tracking and flight control systems, as well as the mission planning and approval process. The chapter describes the prelaunch ground safety and launch flight safety procedures developed and enforced at the National Ranges to ensure launch and mission success, personnel safety and to protect the public from the potential impacts of a launch accident. Chapter 3, Expendable Launch Vehicles (ELV) Characteristics, introduces the basic propulsion technology, configuration and capability for operational US launch vehicles (Titan, Delta, Atlas/Centaur, Scout) likely to be commercialized in the near term. ELV historical launch performance, operational reliability data and the bearing this record has on public safety issues are also discussed. Chapter 4, Launch and Orbital Operations, describes the phases of space operations, from ground preparation to launch, through orbital transfer, operation and re-entry. It also provides the reader with sufficient background to understand possible ELV and mission failures during launch, orbital maneuvers and orbit insertion and operation.

Volume II : Space Transportation Hazards identifies and discusses the major and generic classes of hazards associated with each phase of space operations. Chapter 5, Pre-launch and Launch Hazards, identifies the types of hazards, such as explosions, fires, toxic vapors and debris, as a function of accident scenario and time after launch and defines their nature and severity indices. Further, a comparative perspective on potential ELV space launch accidents is provided by analogy to more common and socially accepted transportation and industrial accidents involving chemicals and fuels. Chapter 6 is devoted to Orbital Collision Hazards, shedding light on the Low Earth Orbit (LEO) and Geosynchronous Earth Orbit (GEO) space environment and the increasing threat of on orbit collisions to spacecraft. The sources and density of orbital debris are discussed and their implications for the probabilities of collisions involving operational satellites are quantified. Chapter 7 defines and reviews Re-Entry Hazards and their quantification by addressing the orbital lifetime and decay of space objects depending on their orbital characteristics, the behavior and survivability of space objects upon re-entering Earth's atmosphere and the uncertainties associated with predicting points of entry and ground impacts.

Volume III: Space Transportation Risk Analysis introduces the methods and uses of Risk Analysis as they apply to the qualitative evaluation and quantitative assessment of public risk exposure from commercial space operations. Chapter 8 introduces the concepts of risk acceptability and relative risk and the tools of Risk Analysis Methodology developed for a broad range of industrial and regulatory purposes. These include: failure analysis methods (which focus on failure modes and failure chains); consequence analysis methods (which focus on the severity of possible consequences of failures); hazard analysis methods (focused on the identification and ranking of hazards); and integrated probabilistic risk analysis methods, such as Fault Tree Analysis, which quantify risk as the mathematical product of an event probability and its consequence magnitude. Chapter 9 discusses the Applications of Risk Analysis to Space Launch Operations as used to date by the Government Agencies (NASA, DOD, DOE) concerned with assuring and maintaining high operability and safety standards for space launch operations. The chapter reviews the objectives, concepts, tools and uses of risk analyses conducted at the National Ranges by sponsoring agencies, in light of de-facto risk/safety goals, criteria and priorities. Finally, Chapter 10 provides an integrated Generic Risk Assessment of Representative Launch Scenarios background by reviewing the risk associated with typical ELV missions from current Range locations. Then the benefits of established Range Safety Controls are quantified, relative to their hypothetical absence, employing the framework of a simplified Community Damage (COMDAM) model in a typical Risk Matrix evaluation procedure.

Volume I: Operations

1. THE CONTEXT FOR A HAZARD ANALYSIS OF COMMERCIAL SPACE ACTIVITIES

1.1 POLICY AND MARKET CONTEXT

A new set of realities, shaping space activities worldwide, must be considered in order to provide the context for the nature, scope and thrust of commercial space efforts in the US. An extensive set of recent Congressional legislation, studies and reports[1-8] has documented the rapidly changing climate for international cooperation and competition in space activities and the need for greater political and economic flexibility in providing access to and services for space exploration and exploitation, if the US is to maintain its leadership in space. The arena of space technology, infrastructure development and new space applications has expanded in recent years to include more developed and third world nations.[2,8] In 1986 alone, the USSR had 91 successful space launches vs. the US with 6 and 2 each for China, Japan and ESA (European Space Agency). The US is revising and reshaping its space policy and priorities. These changes are needed if it is to provide the national and international leadership and foster the stability to ensure that, following the initial space exploration and utilization phase, the promise of commercial space development becomes a reality.[3-7] This will enable the US aerospace industry to capitalize on its technical superiority for the benefit of mankind and economic pay-back.

Both Congress and the Administration have proposed, enacted and promoted new space commercialization initiatives, most notably in privatizing remote sensing satellites and promoting the use of commercial expendable launch vehicles (ELV's) and launch services to place both government and commercial satellites into orbit.[6,7,9]

In May 1983, the President issued a new policy for commercialization of ELV's and in February 1984, by Executive Order 12465 ("Expendable Launch Vehicles in Space"), he designated the Department of Transportation (DOT) as the lead agency to facilitate and encourage commercial ELV activities and to license commercial space operations.

The STS-Challenger disaster and ensuing ELV accidents have severely limited the US access to space and indirectly provided new opportunities and incentives to ELV manufacturers and to commercial payloads and launch services providers.[7,10] As a result, all government agencies involved in space activities have been instructed to enable, foster and implement the new commercial space policies and laws and to develop the supporting regulatory framework and technology infrastructure for greater private sector participation in space transportation and development efforts.

1.2 REGULATORY CONTEXT FOR COMMERCIAL SPACE OPERATIONS

The Commercial Space Launch Act of October 30, 1984 (Public Law 98-575) (the Act), assigned to the Secretary of Transportation the responsibility for carrying out the Act.[6] The purpose of this Act is:

(1) to promote economic growth and entrepreneurial activity through utilization of the space environment for peaceful purposes;

(2) to encourage the United States private sector to provide launch vehicles and associated launch services by simplifying and expediting the issue of commercial launch licenses, facilitating and encouraging the use of excess Government-developed space launch capabilities and transferring technology to the private sector ;

(3) to designate an executive department to oversee and coordinate the conduct of commercial launch operations; to issue and transfer commercial launch licenses authorizing such activities; and to protect the public health and safety, safety of property, national security and foreign policy interests of the United States.

In 1984, the Secretary of Transportation created the Office of Commercial Space Transportation (OCST) and delegated to it the Secretary's responsibilities. As stated in Section 8(a)(2) of the Act, the Secretary is charged with prescribing "requirements as are necessary to protect the public health and safety, safety of property, and national security and foreign policy interests of the United States."

To carry out this responsibility, OCST established a program to develop safety and regulatory requirements for commercial space launch license applicants.[12] The Transportation Systems Center (TSC) is providing technical support to OCST to this end and has been assisting in the development of launch safety requirements based on the Preliminary Hazards Analysis embodied in this report.

However, it must be made clear that the focus of OCST licensing and regulatory activities is primarily on public safety and not on mission success.[6,12] This unique perspective and mandate for DOT is and will be reflected in the OCST safety research, rule making and licensing activities. DOT will have to regulate not just commercial launch sites and commercial launches, but payloads launched aboard these vehicles. These include retrievable materials processing, re-entry systems, non-government research activities and many other, as yet unforeseen, commercial space systems.

DOT/OCST will also license the construction and operation of new private launch Ranges, as well as any commercial Range Safety services.[12] OCST will also specify the certification requirements for Range Safety personnel and launch services providers, that might impact the public safety. Under the Act, DOT must also issue licenses for any launch vehicle or operation on foreign territory by a US citizen or company.

1.3 PURPOSE AND SCOPE OF REPORT: HAZARD ANALYSIS OR RISK ASSESSMENT

This report presents the results of a technical review and analysis of literature and information in the public domain, conducted to identify and evaluate the prospective hazards to the public and the environment, and to assess risk exposure levels associated with commercial space activities. Included in the report is a review of the present status of US space technology and practices (Vol. 1), as they relate to the hazards associated with commercial space missions and their mitigation (Vol. 2). In this analysis, a commercial space mission is comprised of four phases: prelaunch, launch, orbital and re-entry (Table 1-1). For each mission phase the potential classes of hazards which pertain to the people, procedure, equipment, facility and environmental elements are identified. These hazards have been identified and evaluated in light of DOT/OCST's mission, based on the review of existing literature and practice of space related risk analyses (Vol.3).

The following definitions will aid the reader with the assimilation of information in this report. An extensive Glossary of terms has been provided (Appendix A) and a discussion of terminology and procedures is given in Chapter 8 (Vol. 3).

An accident is defined as an undesirable event resulting from any phase of commercial ELV launch operations and space activities with the potential to cause injury or death to people, or damage to property.

Risk assessment is the systematic examination of an actual or proposed system or operation, to identify and evaluate potentially hazardous events and their consequences. The principal purpose of such an analysis is to assist policy makers, regulators and managers in deciding on risk avoidance, risk reduction or mitigation strategies. It can lead to either confirming the continued acceptability of a system or operation from the safety point of view, or setting new risk acceptability and regulatory thresholds for the protection of public safety (see Ch. 8, Vol. 3). Although the terms Risk Assessment and Hazard Analysis are both used in this report in nearly synonymous fashion, the latter is part of the former. There are other closely related terms used in the literature in similar contexts: "Hazard" ~~is often interchanged with "Risk", and "Analysis" for "Assessment", thus giving four common usage~~ expressions, namely: risk assessment, risk analysis, hazard assessment and hazard analysis.

i) An Analysis is typically a technical procedure following an established pattern;

ii) An Assessment is the consideration of the results of analysis in a wider context to determine the significance of the analytical findings;

iii) A Hazard is considered to be an existing property, condition, or situation, which has the potential to cause harm. For example, liquid hydrogen used as a rocket propellant is a hazard because of its chemical nature, and intrinsic flammability and explosiveness.

iv) Risk is related to both the consequences of an accident (i.e., hazard potential being realized and causing harm) and its likelihood of occurrence (Ch. 8, Vol.3). Risk is mathematically expressed as the product of the probability of an accident and the magnitude of its consequence. Thus, the risk from a liquid hydrogen tank is the product of the probability that its containment will fail and the magnitude of the resulting explosion and/or fire damage. Hence, people and property may be considered "at risk" from a nearby hazard.

v) An Accident occurs when the hazard potential for damage is activated by a stimulus and results in damage to a given system, component or operation, or in injury to people. Other operational and technical definitions for terms used throughout the report are given in the Glossary (Appendix A).

It must be kept in mind that a system or operation is considered to be "safe" when its risks are deemed economically, socially and politically acceptable, based on prevailing standards. These issues will be discussed and illustrated in detail in Vol. 3.

1.4 APPROACH TO HAZARD ANALYSIS FOR COMMERCIAL SPACE OPERATIONS

For over two decades, the US Government has been one of the world leaders in the development and exploration of outer space. In this role, the Government mission agencies (NASA and DOD) have developed and successfully implemented launch safety requirements in support of a wide variety of space missions (see Chs. 2 and 4 of Vol.1). Launch safety requirements have been established for both unmanned and manned space systems and operations, as well as for integration of specific payloads. As such, the standards presently in use at Government Ranges have evolved not only out of the need to protect the public safety and property, but also from the need to protect launch site personnel, facilities and on board astronauts; to ensure mission success; to evaluate launch vehicle performance; and to provide research results that would assist in expanding the national space exploration effort.

Since the only currently available launch sites are National Ranges owned and operated by US Government agencies (DOD and NASA as first parties), the basic launch and system safety regulations now in place at these facilities will probably continue to be observed in the near future by any commercial launch vehicle provider or operator that requires access to and use of Government launch facilities (second party). Cost, access and time constraints may influence the viability of commercial launch operations on these Ranges, while vehicle reliability and safety will remain major concerns. Recognizing this situation, OCST has undertaken an effort to examine ELV safety standards, launch hazards and risk analysis methods to ensure the protection of public safety and property[12] (third party), as opposed to Government launch facility (first party) and ELV or satellite manufacturers and operators (second party) who enter User Agreements.

As the initial effort in the development of a program to address the safety issues, this report focuses on the identification and evaluation of the safety hazards associated with ELV's and their launch operations from established and available Government Ranges as well as new launch sites that may be developed and operated in the future by commercial entities, or in partnership with states and federal entities.

Protecting the public health and safety as stated in the Act, requires that safety regulations be directed at preventing the occurrence of potentially hazardous accidents and at minimizing or mitigating the consequences of hazardous events. This will be accomplished by employing system safety concepts and risk assessment methodology to identify and resolve prospective safety hazards. The first step in applying system safety concepts is to define the commercial space launch hazards (preliminary hazard analysis, PHA). With the hazards defined, it is then possible to identify and rank those associated with each specific commercial space launch. Only after the hazards have been identified and satisfactorily assessed, will the goal of providing the public with the highest degree of safety practical have been accomplished. For the preliminary hazard analysis (PHA) presented in this report (Vol. 2), the operational commercial space launch phases have been defined as follows:

1. Prelaunch; 2. Launch; 3. Orbit; 4. Re-entry

For each of these life and operability phases of the commercial space launch process, it is possible to identify the generic classes of hazards that are associated with each phase (see Table 1-1) and to define appropriate regulatory oversight. To identify these hazards, a clear understanding of the system and its operation is necessary, as well as an analysis of the relevant accident history for specific launch systems and subsystems during each phase of launch operation. An analysis of previous accidents is necessary, but not sufficient, for the identification of prospective hazards, since both vehicle configurations (see Ch. 3, Vol. 1) and launch and Range Safety procedures (see Ch. 2, Vol.1) have improved with time. In 30 years of Government space launch activities and ELV operations to date, both the military and civilian sectors have had an excellent safety record and there have been no major accidents with reported public injuries. Therefore, the data base from which the hazards can be identified is limited, and known to be incomplete, with rare identical failures (see Ch.3). Furthermore, an examination of historical launch data can provide only a tentative list of probable causes and likely accident scenarios and may be incorrect for the purpose of projecting future performance. Special statistical methods may have to be used to account for "learning" from past failures in order to avoid repeating them (see Ch. 9, Vol.3).[7] Previous government ELV and space missions will, however, have to be used to generate a set of

TABLE 1-1. PHASES OF COMMERCIAL LAUNCH OPERATIONS

PHASE	PRELAUNCH	LAUNCH	POST-LAUNCH MISSION AND OPERATION		
			<u>PHASE A</u> ORBIT INSERTION OF PAYLOAD	<u>PHASE B</u> PAYLOAD ON ORBIT OPERATION AND STATION-KEEPING	<u>PHASE C</u> 1. DE-ORBIT AND RE-ENTRY 2. OR: MANEUVER TO HIGH STORAGE ORBIT
Representative Hazards or Events	• Damage to ELV in transit, storage, assembly and testing • Damage to Launch Facilities and Ground Support Equipment • Hazards to personnel • Environmental Damage	• On the pad explosion • Low altitude explosion • Failures of 1st, 2nd or upper stages • Failure of guidance and/or destruct system	• Malfunction in any of the boost stages, and/or motors • Malfunction of apogee / perigee kick motor	• Collision with debris, or other orbiting satellites • Malfunctions and operational failures	• Re-entry Hazards: Natural de-orbit and breakup • Rapid uncontrolled loss of altitude due to solar activity, or failure to maintain orbit • Damage to property or casualties in U.S. & abroad

representative, expected, and projected commercial space launch missions (see Ch. 10, Vol.3). This approach will allow us to examine and evaluate generic hazards associated with commercial space ELV missions (see Chs. 5-7, Vol.2).

1.5 OVERVIEW OF THE REPORT ORGANIZATION

This report is intended to inform and educate a broad readership on the generic sources and nature of hazards associated with space launch activities. Therefore, it is intended to provide both the necessary technical background and the specific hazard analysis methodology, in order to enable a non-technical reader to understand and appreciate the variety of technical issues involved.

<u>Volume 1: Space Transportation Operations</u> provides the background on Range Operations (Ch. 2), current Expendable Launch Vehicles (Ch. 3), and Space Launch and Orbital Missions (Ch. 4). Chapter 2 describes the Range Safety Control systems in place and established practices at the National Ranges. Chapter 3 introduces the basic technology, and typical proven and proposed configurations of ELV's likely to be used for commercial space missions in the near future. The historical reliability based on launch success/failure statistics for the major classes of operational ELV's in the US are also presented in Chapter 3. Chapter 4 describes the space launch and orbital operational phases.

Volume 2: Space Transportation Hazards introduces the generic classes of hazards associated with the use of these ELV's in space launch operations. Chapter 5 discusses fires, explosions, toxic vapor clouds and debris impacts. A relative risk context is provided in Chapter 5 to enable the reader to judge launch hazards by comparison with other common industrial and transportation hazards. Chapter 6 discusses orbital collision hazards to satellites in low and geosynchronous Earth orbits. Chapter 7 reviews and evaluates those hazards to people and property associated with both controlled, and uncontrolled re-entry of space objects.

Volume 3: Space Transportation Risk Analysis deals with the analytical tools available to assess public risks (Ch.8), the modeling and application of such tools to space operations (Ch.9) and illustrates the specific risks associated with commercial ELV launches in the near future (Ch.10). Since DOT/OCST will sponsor and perform risk assessment/risk management research to support commercial space launch licensing reviews and awards, Chapter 8 defines and introduces the standard methods of Risk Assessment. Chapter 9 reviews the published technical risk assessments conducted for selected space applications, focusing specifically on when, how and why such risk studies were conducted and on the software tools available for this purpose. Finally, in Chapter 10, an illustration of risk analysis is provided for representative ELV launch/mission scenarios which indicates how the public risk exposure from commercial space activities may be estimated, both with and without Range Safety controls in place. Also, a conceptual risk assessment and acceptability matrix is provided for comparing public risk levels associated with each phase of space launch operations. The benefits of Range Safety control systems and practices now enforced at Government Ranges as the key safeguards to manage and minimize the public risk exposure from future space activities to "acceptable" levels are made clear in Chapter 10.

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   (b) "The Expendable Launch Vehicle Commercialization Act," ibid., 1986.

   (c) "The Land Remote - Sensing Commercialization Act of 1986," (Report to accompany HR 5155), ibid, 1984.
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    (b) "Commercial Space Transportation; Licensing Regulations; Interim Final Rule and Request for Comments," 14 CFR Ch. iii, Feb. 26, 1986 (FR 51, No. 38, p. 6871 et seq.)

    (c) "Commercial Space Transportation; Licensing Regulations; Final Rule," April 4, 1988 (FR 53, No. 64, p. 11004 et seq.)

2. RANGE OPERATIONS, CONTROLS AND SAFETY

2.1 RANGE CHARACTERISTICS FOR SAFE OPERATION

2.1.1 US Government Launch Sites

The US Government has traditionally operated separate civilian and military space programs. NASA is the lead agency for civilian space activities, and assists as necessary, the Departments of Energy, Interior, Commerce, Transportation and Agriculture which also maintain space research and utilization programs.

The US Space Command (US SPACECOM) coordinates all military space activities, but the three services also have operational Space Commands. DOD recently established a Consolidated Space Test Center (CSTC) under the Space and Missile Test Organization (SAMTO). A very recent DOD regulation governing military Range activities designated the Air Force as the lead agency for the tri-service conceptual Space Test Range at Onizuka AFB, in California, with a special focus on safety issues.

The Eastern Test Range (ETR) is under the direction of the USAF Eastern Space and Missile Center (ESMC) at Patrick Air Force Base, Florida, and the Western Test Range (WTR) is under the direction of the USAF Western Space and Missile Center (WSMC) at Vandenberg Air Force Base, California. WTR launches are from Vandenberg Air Force Base; ETR launches are from the Cape Canaveral Air Force Station (CCAFS). NASA space missions are launched from the Florida Kennedy Space Center (KSC), also on Cape Canaveral and occasionally from WFF.

The United States has a major launch site in Florida at Cape Kennedy (NASA) and CCAFS (DOD) for manned, lunar and planetary launches, and for launching satellites to geostationary orbit (primarily for weather and communications). It has another major West Coast launch site at Vandenberg Air Force Base (VAFB), California, for satellites (including weather, Earth resources, navigation and reconnaissance) which must go into polar orbits. A smaller launch site for small space payloads and for sub-orbital research rockets is the NASA/Goddard Space Flight Center (GSFC) Wallops Flight Facility (WFF) site at Wallops Island, Virginia. Sub-orbital launches and short-range vertical testing are accomplished at White Sands, New Mexico, from the White Sands Missile Range (WSMR). In addition, the US Government has conducted launches from a number of other CONUS and off-shore sites.

Each of the National Ranges has unique capabilities related to its mission, siting and facilities, as well as specific requirements for the Range Users (see Vol. 3, Chs. 9, 10). The safety philosophy of ground and Range operations is generally that of dealing with controlled, managed and acceptable risks.

Procedures have been established to handle and store all materials (propellants, etc.) which may be a hazard, control and monitor electromagnetic emissions and govern transportation of materials to and from the facility.[4] The storage of propellants and explosives used in Expendable Launch Vehicles (ELV's) is controlled by quantity-distance criteria, as specified.[3] Failure modes and effects analyses (FMEA) are prepared, when necessary, for all potentially hazardous activities and devices (see Ch. 8). Quantitative risk analysis has rarely been used to establish launch and space operational risk because of the conservative philosophy of vehicle design, ground and launch procedures and the difficulty in developing realistic estimates of hazardous event probabilities and accident scenarios (see discussion in Vol 3, Chs. 9 and 10).

Since there are currently no private commercial space launch range facilities in the US, we will describe the past and current practices at US Government Range facilities. It is assumed throughout this report that the level of operational safety at licensed commercial space facilities will be comparable or equivalent to the level of safety maintained at US Government Ranges.

2.1.2 Ground Operations and Safety

One of the principal responsibilities of the launch Range is to perform all of those tasks which eliminate, or at least acceptably minimize, the hazards from an expendable launch vehicle (ELV), both prior to and during the launch.[1-3] This is accomplished by establishing:

(1) requirements and procedures for storage and handling of propellants, explosives, radioactive materials and toxics;

(2) performance and reliability requirements for flight termination systems (FTS) on the vehicle;

(3) a real-time tracking and control system at the Range; and

(4) mission abort, vehicle destruct or flight termination criteria which are sufficient to provide the necessary protection to people both within (on-Range) and outside (down-Range) the boundaries of the launch facility.

At each Range there is a hierarchy of regulations and requirements for Ground and Launch safety implementation (see also Chs. 6, 7, Vol. 2). Generally, the National Ranges take responsibility for the vehicle handling and safe operation from receipt until the time of orbital insertion. Safety issues associated with on-orbit impacts and re-entry from orbit are not normally the responsibility of the Range (see Chs. 6, 7, Vol. 2). Control of public risks from jettisoned stages and hardware prior to orbital insertion are a Range responsibility.

The following sections provide a general introduction to the various aspects of planning, ground operations and flight control, all with a specific emphasis on safety. Chapter 10 in Vol. 3 provides a more detailed discussion of launch hazards and their minimization by Range Safety controls.

2.1.3 Range Safety Control System

The NASA "Range Safety Handbook" states: "The flight safety goals are to contain the flight of all vehicles and preclude an impact which might endanger human life, cause damage to property or result in embarrassment to NASA or the US Government. Although the risk of such an impact can never be completely eliminated, the flight should be carefully planned to minimize the risks involved while enhancing the probability for attaining the mission objectives."[7]

The real-time Range Safety (or Flight) Control System must accurately and reliably perform the following functions:

1) Continually monitor the launch vehicle performance and determine whether the vehicle is behaving normally or failing;

2) Track the vehicle and predict (in real-time) where the vehicle or pieces of the vehicle will impact in case of failure and if flight termination action is taken;

3) Determine if there is a need to delay or abort the launch or destruct the vehicle, based on a comparison of predetermined criteria with the current vehicle status; and

4) If necessary to protect the public, send a command to abort the mission either by vehicle destruct or engine shutdown (thrust termination). Note that the term "destruct" is used generically in this report to denote flight termination actions for Range Safety purposes. In reality, thrust (and the flight) can be terminated on command for some ELV's without vehicle destruction.

Figure 2-1 describes pictorially the activities of the various elements of the Range Safety Control System.

Vehicle performance is determined at all Ranges by visual observation (early in flight) and by real-time telemetry measurements of vehicle status as a back-up to the computed (wind-corrected) behavior of the instantaneous impact point (IIP), discussed below in more detail. The actual location of the vehicle is less important than where the vehicle and its debris will land in case of both normal operation, accidental failure, abort or destruct. Therefore, in tracking a vehicle, velocity data must be obtained either directly or by differentiating successive measures of position. The most frequently used method of obtaining the velocity and position data has been the use of radar trackers, which measure the vehicle position in terms of azimuth, elevation and range relative to the tracker, expressed in a launch-pad centered reference coordinate system Radars are also capable of

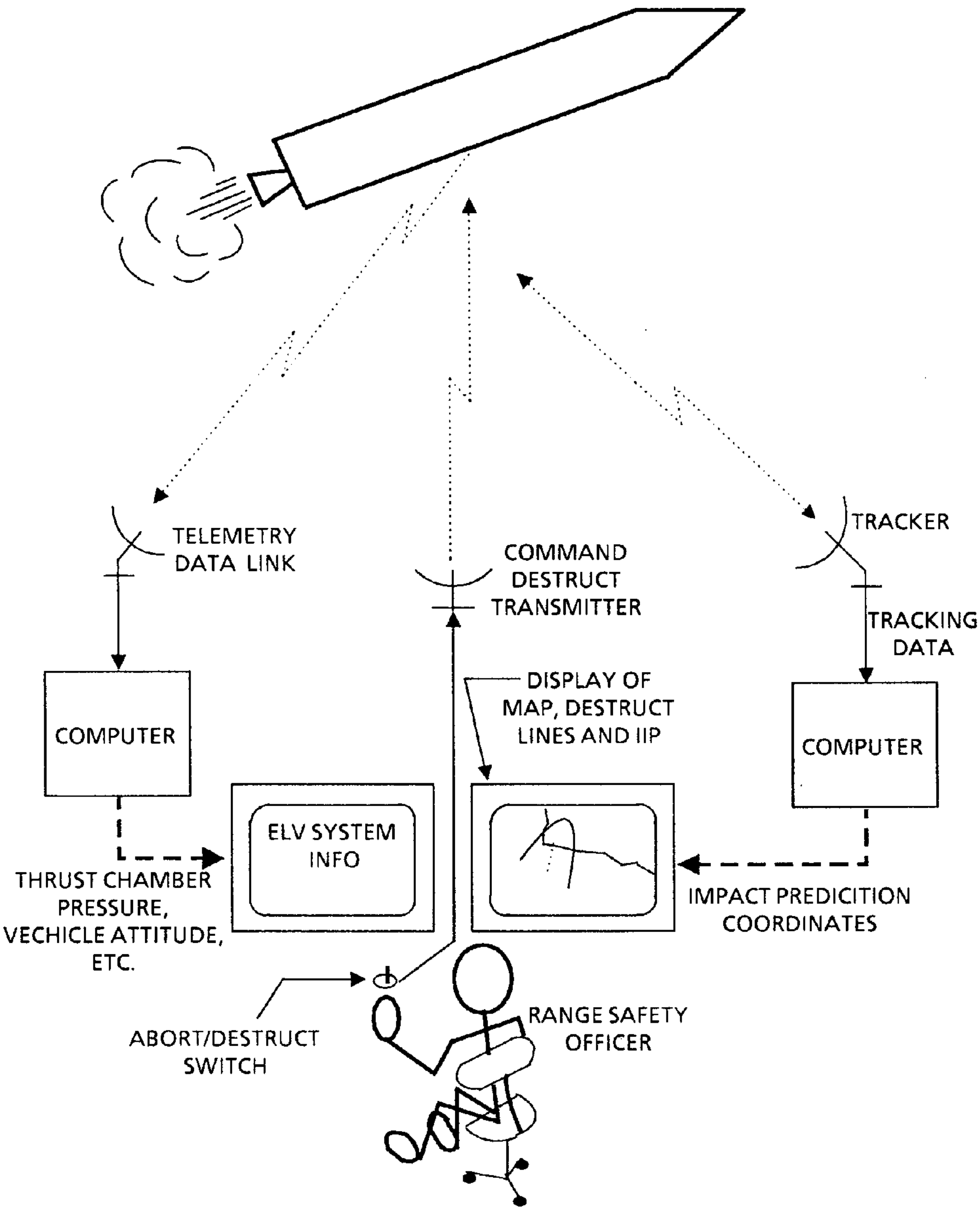


FIGURE 2-1. ELEMENTS OF THE FLIGHT CONTROL SYSTEM

determining range rate, i.e., the rate at which a vehicle is moving toward or away from the radar. A single tracker near the launch pad can provide satisfactory information for two or more minutes of flight depending on the rate at which the launch vehicle is traveling away from the tracker. The quality/accuracy of the tracking data is often affected by several factors, two of which are: (1) multi-path of returned signals which occurs at low antenna elevation angles; and (2) the plume signal attenuation due to high temperature ionization caused by the solid rocket motor exhaust. Multiple radar trackers are used to minimize these problems and to provide redundant measurements, so that failure of a single tracker will not jeopardize the mission. Early in flight, when the launch vehicle is still close to the ground, the radar may not be able to track the vehicle. In this case, visual observation and telemetry may be the only means of determining whether there is a malfunction and whether the vehicle maintains the correct attitude. Position and velocity data, along with the predicted instantaneous impact point (IIP) are typically displayed in real-time in the Launch Control Center (LCC).

Although not yet applied at the National Ranges, it is possible to use satellite information for determination of vehicle position and velocity. An electronics package on board the launch vehicle could collect information for calculating the range relative to several separately located navigation satellites and could be telemetered to a ground station, processed and converted into vehicle position and velocity. This will become practical when the Global Positioning System (GPS) satellites become operational. Some Ranges have used three or more geographically spaced telemetry antennas and associated computer equipment to infer the vehicle position and velocity from the Doppler phase shift of the received telemetry signals.

The launch vehicle velocity and position information are generally used to compute an instantaneous impact point (IIP). The IIP is displayed on a screen or chart indicating where the vehicle will impact on the surface if flight were to be aborted at that instant. This impact point is usually computed, assuming no atmosphere, as a vacuum IIP (VIIP) which allows simpler and more rapid trajectory computation. Inclusion of atmospheric drag is generally not necessary to satisfy the objectives of the real-time Range Safety. However, a drag and wind correction is applied in some cases.

Early in the flight the IIP advances slowly, but as the vehicle altitude, velocity and acceleration increase, the IIP change rate also increases. Very early in flight, the IIP change rate increases from zero to several miles per second. Later, it increases to tens of miles and then hundreds of miles per second. As the vehicle reaches orbital velocity, the IIP rate essentially goes to infinity because the vehicle will no longer come down. The difference between the advance of the IIP and the present position (sub-vehicle point) (SVP) is illustrated in Figure 2-2.

It is the advancing IIP that the Range Safety Officer (RSO) is usually observing during a launch. Prior to the launch, a map is prepared with lines drawn to represent the limits of excursion which, when exceeded, will dictate a command signal to terminate flight. A typical set of "destruct lines" is shown in Figure 2-3. The destruct lines are deliberately offset from land or populated areas to accommodate: (1) vehicle performance characteristics and wind effects; (2) the correction for using

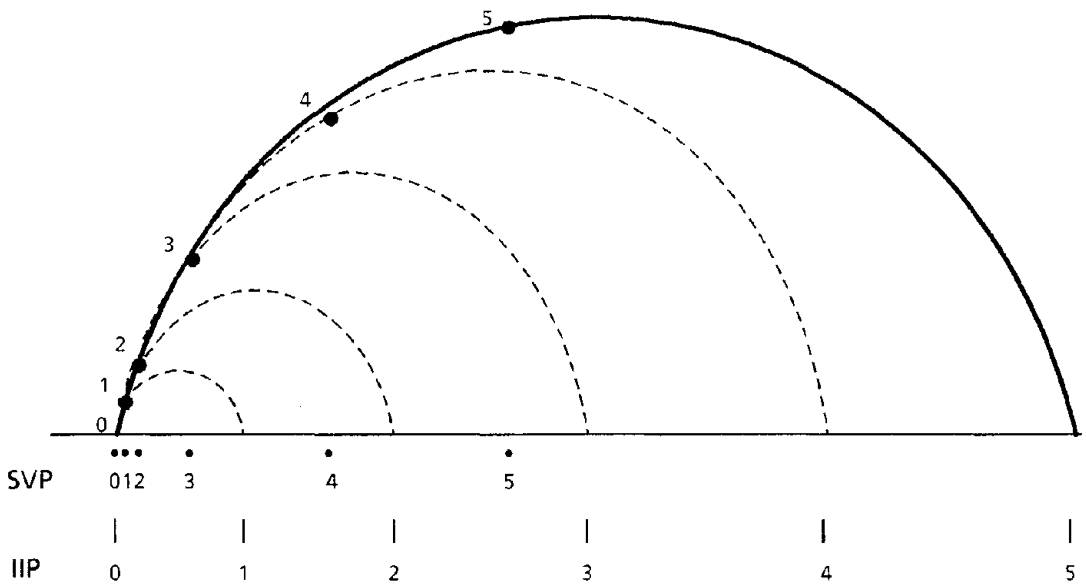


FIGURE 2-2. INSTANTANEOUS IMPACT (IIP) AND SUB-VEHICLE POINTS (SVP)

a vacuum instead of a drag-corrected impact point; (3) the scatter of vehicle debris; (4) the inaccuracies and safety-related tolerances of the vehicle tracking and monitoring system; and (5) the time delays between IIP impingement on a destruct line and the time at which flight termination actually takes place (i.e., human decision time lag). By proper selection of the destruct lines, debris can be prevented from impacting on or near inhabited areas.

The ability of the system to accurately predict the ELV impact point diminishes as the vehicle advances into the flight and the IIP is moving more rapidly along the ground track. Consequently, the difficulties in performing the Range Safety Control function increase with time, particularly if there are land masses or population centers that must be protected near the ground path of the launch trajectory. Regardless of the flight time, the Range Control problem is always more difficult if the flight plan is designed to move close to or over a populated area. If a flight plan requires violation of a prudently designed abort line, a risk analysis is performed to determine if the risk is acceptable. If the risk is small enough, the Range Commander may choose to permit a launch without an abort line for portions of the flight (for further discussion see Vol. 3, Ch. 10).

2.2 LAUNCH PLANNING

The principal mission of Range Safety personnel is the protection of life and property both off and on-site at the launch facility. In keeping with that objective, the Range must not be negligent, nor impose undue restrictions on launch conditions, that could result in a high probability of a good vehicle being destroyed. Minimization of the probability of terminating a "good" flight, and simultaneous minimization of the potential risk due to a malfunctioning ELV, is accomplished

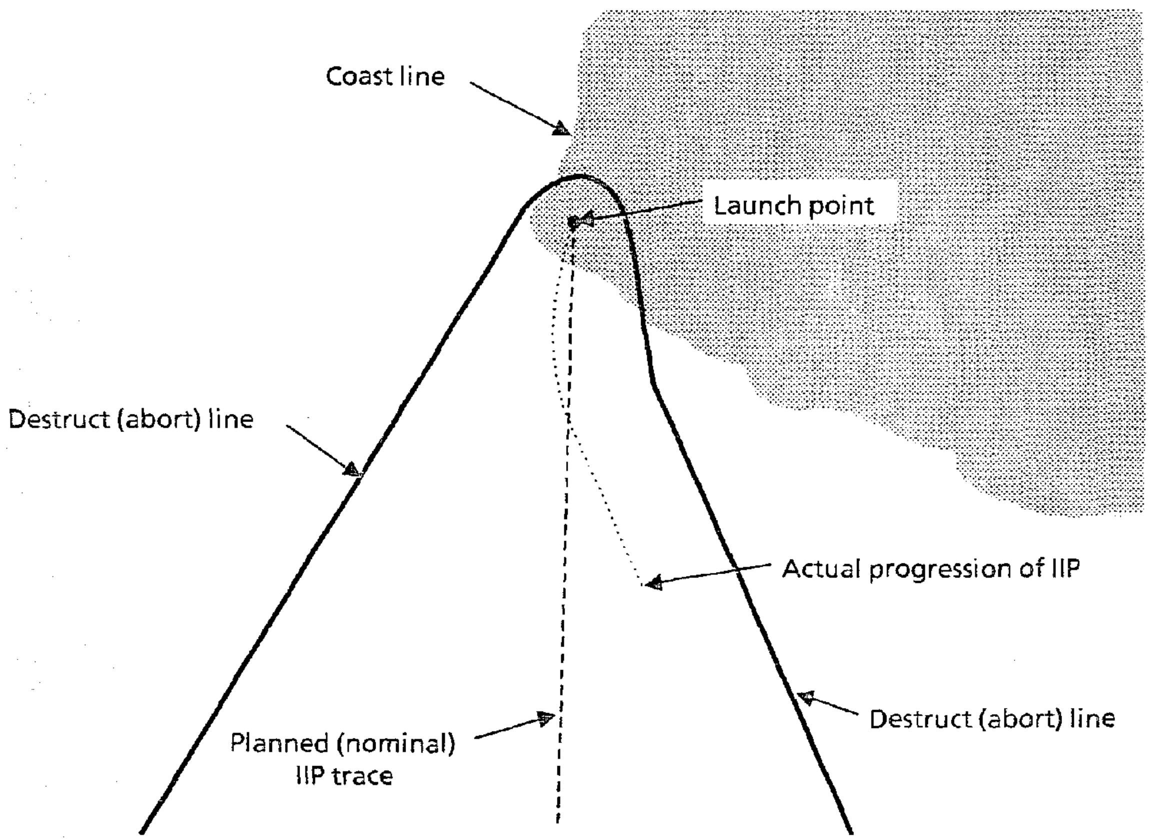


FIGURE 2-3. TYPICAL IIP AND DESTRUCT LINE DISPLAY

through careful mission planning, preparation and approval prior to the launch. The planning is in two parts: (1) mission definition such that land overflights or other risky aspects of the launch are avoided and/or minimized; and (2) development of data which support the real-time decision and implementation of active control and destruct activities. These two aspects are discussed in the following subsections.

2.2.1 Mission Planning

Figure 2-4 contains a map showing the ground trace of a hypothetical launch from Vandenberg Air Force Base (VAFB) on an azimuth which causes overflight of islands south of the base, flight along the coast and overflight of a portion of Chile and Argentina (in fact, such azimuths are restricted, as discussed in Ch. 10). The greatest risk is in the immediate vicinity of the launch area and to any occupants of the nearby islands. Since the overflight of these islands is planned, abort lines cannot protect their inhabitants. Abort lines can protect the coast from vehicle overflight and debris impacts, in case of destruct. However, if the intended flight path is too close to the coast and the abort lines are too close to the planned flight path, there is the possibility that the IIP of a good, but slightly drifting, vehicle will cross the abort line and thus require a commanded destruct. The overflight of the tip of South America is not as serious a problem because the rate of advance of the

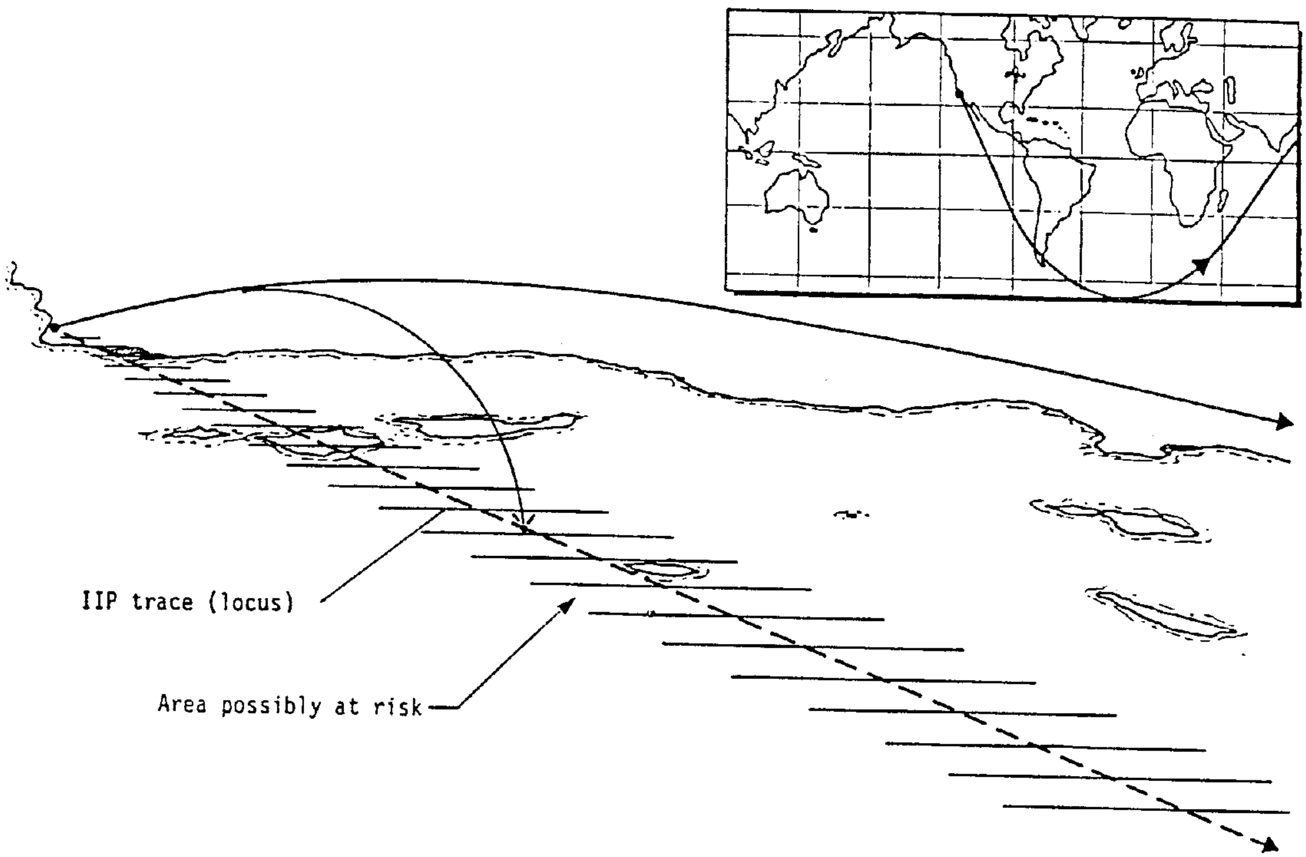


FIGURE 2-4. A HYPOTHETICAL AND RISKY TRAJECTORY PLAN

IIP is so rapid and the vehicle altitude is so high at that point in flight that there is a much smaller possibility of any hazard to that region. A failure would have to occur within a specific time interval (a second or two of flight) in order for any resulting debris to impact the region (see Ch. 10 for a more in-depth discussion of such risks).

In addition to considering where the aborted or destroyed vehicle will land, one must also consider where the debris from normally jettisoned spent stages will impact. For example, the vehicle might fly safely over the islands, but drop an empty rocket casing on one of them. Mission planning must consider and avoid all of the hazards associated with normal launch operations, as well as other potential hazards associated with potential accidental failures for the particular launch plan.

A Range user may request a particular trajectory to satisfy desired mission requirements (i.e., orbital inclination) or payload constraints. For example, a trajectory having a more easterly azimuth will enable the vehicle to put a heavier payload into orbit. If the launch vehicle is limited in lift capacity, the Range user may try to get the most favorable launch azimuth (in this case, eastern) in order to increase the amount of payload the vehicle can place into orbit. The Range Safety function in the mission planning stage is to limit the range of allowed launch azimuths to those which keep the risk to people on the ground at acceptably low levels. Another mission planning responsibility is to evaluate all other aspects of the planned launch, e.g., impact points of jettisoned stages, to assure the acceptability of the overall risk of the mission.

There are situations where the conflict between safety requirements and mission objectives require special studies to determine risks and define tradeoffs. In these cases detailed risk analyses are performed using models that consider the probability of the vehicle failing in a variety of modes and simulate the behavior of the missile during and after malfunction,including the effect of activating the flight termination system. Such risk analyses usually compute the land impact probability and associated *casualty expectation* (the average number of casualties expected per launch). Typically, missions with casualty expectations of less than one in a million are considered reasonably safe. If the risks are higher, the mission ordinarily comes under more scrutiny (see Chs. 9, 10 for more detailed discussion).

One of the options for maintaining a low risk for a launch is to move the abort lines away from the populated areas and closer to the trace of the IIP for the nominal trajectory. While this decreases the overall launch risk, it increases the probability of aborting a good vehicle. Considering the very high value of many of the launch vehicles and their payloads, these tight abort lines put additional pressure on the Range Safety Officer (RSO) who must decide on an active destruct command.

Another option to minimize the risk of a normal, or failed, launch to the population surrounding the Range is to place much tighter constraints on the tolerable wind and other meteorological conditions at the time of the launch.

2.2.2 Standard Procedures to Prepare for a Launch

The National Ranges have provided standards and requirements for organizations desiring to launch vehicles from their facilities. For example, the United States Air Force has specific safety requirements issued for each of the Ranges under USAF control. These documents describe the safety policy and procedures and also define the data submittal and launch preparation requirements for the Range user.[1,2] The categories covered by these requirements include ground safety (handling of propellants, ordnance, noise, hazardous operations, toxics, etc.), flight analysis (vehicle trajectory, mission, etc.), flight termination systems (FTS), ground operations and flight operations. Included in the flight analysis portion are requirements for trajectory modeling and descriptions along with the dynamic characteristics of the vehicle during a malfunction turn.This information is used by Range personnel to construct the abort lines. Ref. 5 is an example of the equipment requirements to support typical missions from a National Range.

REFERENCES TO CHAPTER 2

1. "Range Safety Requirements, Range Safety Regulation," ESMC Regulation 127-1, Dept. of the Air Force, Headquarters Eastern Space and Missile Center (AFSC), Patrick Air Force Base, Florida , July 30, 1984.

2. "Range Safety Requirements, Range Safety Regulation," WSMC Regulation 127-1, Dept. of the Air Force, Headquarters Western Space and Missile Center (AFSC), Vandenberg Air Force Base, California, May 15 1985.

3. Air Force Regulation (AFR) 127-100, Safety-Explosives Safety Standards (May, 1983; 2nd revision Jan., 1985).

4. Hazards of Chemical Rockets and Propellants Handbook: Vol. I, General Safety Engineering Design Criteria; Vol. II, Solid Rocket Propellant Processing, Handling, Storage, and Transportation; Vol. III, Liquid Propellant Handling, Storage, and Transportation. CPIA 394, Vol. I, II, and III. Chemical Propulsion Information Agency, 8621 Georgia Ave, Silver Spring MD 20910, September, 1984.

5. "Missile Flight Control Operations Requirements," WSMC MFCOR, Dept. of the Air Force, Headquarters Western Space and Missile Center (AFSC), Vandenberg Air Force Base, California, 3 January 1984.

6. "Space Activities of the US, Soviet Union and other Launching Countries: 1957-1986", M.S. Smith, C.R.S. US Congress Report 87-229 SPR, Feb. 1987.

7. "Range Safety Handbook", NASA GHB 1771.1, Sept. 1984.

3. EXPENDABLE LAUNCH VEHICLE (ELV) CHARACTERISTICS

3.1 GENERAL CHARACTERISTICS

US military and civilian space programs have developed a number of expendable launch vehicles suitable for use in commercial space activities.[1,2] These vehicles include the Delta, Atlas/Centaur, Titan and Scout rockets in various configurations, as required by the desired mission. Additionally, there are launch vehicles under development or proposed by Government agencies or private corporations. This chapter describes the technology and configuration for established US launch vehicles along with some of the proposed vehicles which are likely to become commercialized in the near future. Table 3-1 provides a partial overview of currently operational launch vehicles and launch vehicles (including upper stages) under development.[1]

Expendable Launch Vehicles (ELV's) are very different from aircraft in their operational objectives, environment and flight characteristics. Directional control is usually achieved by gimballing (tilting) the engines to redirect the thrust at small angles to the direction of motion, or by use of guidance thrusters. Directional control of rockets is assured by several methods other than gimballing or thrusters, such as liquid injection thrust vector control. Thus, guidance of the vehicle is only possible while thrust is applied. These vehicles are aerodynamically unstable. Radical controlled turns are possible, but guidance system or directional control system failures will usually result in tumbling. Propulsion staging, which refers to the several rocket engines in the vehicle being used sequentially in time, is used in most ELV's. This staging improves performance by separating a stage (rocket motors and case), after its fuel has been expended. The next stage can then maintain the acceleration of the smaller remaining mass using fewer or smaller rocket engines than were needed by the preceding stage. The discarded (spent and jettisoned) stages follow a ballistic trajectory to impact. In some cases, a second or third stage may attain orbital velocity before being discarded. The mechanics and dynamics of launch vehicles and orbital operations are discussed in Chapters 2 and 4 in Vol. 1, and will be further illustrated in the context of hazard analysis in Chapter 10.

Because of their unique hazardous characteristics (see Vol. 2, Ch. 5), the operational ELV's are only launched under tightly controlled conditions. Safety considerations dictate a carefully defined launch trajectory which minimizes the risks to people and property from both normally functioning vehicles, which drop stages and other equipment which will impact the Earth, and from abnormally functioning or failed vehicles. In the latter case, the vehicle may be destroyed by a radio signal command to prevent it from exceeding the bounds of the predetermined flight safety corridor as discussed in Chapter 2.

TABLE 3-1. CHARACTERISTICS OF US EXPENDABLE LAUNCH VEHICLES (Ref. 1)

Vehicle Name	User Agency	Vehicle Contractor	Propulsion	
			Stage No.	Engines
BASIC VEHICLES				
Titan 34D / Transtage	USAF	Martin Marietta	0 1 2 3	2 x 120-in. UA 1205 (strap-on) 2 x Aerojet LR-87-AJ-11 1 x Aerojet LR-91-AJ-11 2 x Aerojet AJ10-138
Titan 34D No Upper Stage	USAF	Martin Marietta	0 1 2	2 x 120-in. UA 1205 (strap-on) 2 x Aerojet LR-87-AJ-11 1 x Aerojet LR-91-AJ-11
Titan 2 / SLV	USAF	Martin Marietta	1	2 x Aerojet LR-87-AJ-5 1 x Aerojet LR-91-AJ-5
Titan 3	Commercial	Martin Marietta	0 1 2	2 x 120-in. UA 1205 (strap-on) 2 x Aerojet LR-87-AJ-11 1 x Aerojet LR-91-AJ-11
Titan 4 Centaur G Prime	USAF	Martin Marietta	0 1 2 3	2 x 120-in. UA 1207 (strap-on) 2 x Aerojet LR-87-AJ-11 1 x Aerojet LR-91-AJ-11 2 x P&W RL10A-3-3A
Titan 4 / IUS	USAF	Martin Marietta / Boeing	0 1 2 3	2 x 120-in. UA 1207 (strap-on) 2 x Aerojet LR-87-AJ-11 1 x Aerojet LR-91-AJ-11 1 x UTC solid rocket motor - 1 1 x UTC solid rocket motor - 2
Atlas / Centaur	NASA	GD Space Systems Div.	1/2 1 2	2 x Rocketdyne YLR-89-NA7 1 x Rocketdyne YLR-105-NA7 2 x P&W RL10A-3-3A
Delta 3914 / Delta 3924	NASA	McDonnell Douglas	1 1 2 3	1 x Rocketdyne RS-27 9 x Thiokol TX526-2 1 x TRW TR201 / 1 x Aerojet AJ10-118K 1 x Thiokol TE 364-4
Delta 3910 / 3920 / PAM-D	NASA	McDonnell Douglas	1 1 2 3	1 x Rocketdyne RS-27 9 x Thiokol TX526-2 1 x TRW TR201 / 1 x Aerojet AJ10-118K 1 x Thiokol Star 48
Delta 6920 / PAM-D IMLV	USAF	McDonnell Douglas	1 1 2 3	1 x Rocketdyne RS-27 9 x Thiokol TX-780 1 x Aerojet AJ10-118K 1 x Thiokol Star 48B
Delta 7920 / PAM-D	USAF	McDonnell Douglas	1 1 2 3	1 x Rocketdyne RS-27 9 x Hercules GEM 1 x Aerojet AJ10-118K 1 x Thiokol Star 48B
Scout / SLV-1A	NASA, USAF	Vought	1 2 3 4	1 x UTC Algol 3 1 x Thiokol Castor 2 1 x Thiokol Antares 3 1 x Thiokol Altair 3
UPPER STAGES				
Centaur D-1A / D-1T	NASA	GD Space Systems Div.	Varies	2 x P&W RL10A-3-3A
Transtage	USAF	Martin Marietta	Varies	2 x Aerojet AJ10-138
Stage Vehicle System	USAF	Fairchild / Space	2	2 x Thiokol TE-M-364-4
Orbit Insertion System	USAF	Fairchild / Space	1	1 x Thiokol TE-M-616
Stage Vehicle Sys. (SGS-II)	USAF	McDonnell Douglas	1-2	2 x Thiokol Star 48
STS / PAM-A	NASA	McDonnell Douglas	1	1 x Thiokol (MM3)
STS / PAM-D	Varies	McDonnell Douglas	Varies	1 x Thiokol Star 48
STS / PAM-DII	Varies	McDonnell Douglas	1	1 x Thiokol PAM-DII
IUS	USAF, NASA	Boeing	1-2	SRM-1 SRM-2
Transfer Orbit Stage	Varies	Orbital Sciences	Varies	SRM-1
Apogee & ManeuverStage	Varies	Orbital Sciences	Varies	Rocketdyne RS-51
TOS / AMS	Varies	Orbital Sciences	Varies	SRM-1, Rocketdyne RS-51

TABLE 3-1. CHARACTERISTICS OF US EXPENDABLE LAUNCH VEHICLES (CONT.) (Ref. 1)

Vehicle Name	Propellants (oxidizer/fuel)	Thrust (lb.) * = lb.sec.	DIMENSIONS & WEIGHT: Max dia. (ft.) (exclud. strap-ons)	Length (ft.) (exclud. payload)	Launch weight (lb.)	PERFORMANCE Payload (lb.): Orbital	Escape
BASIC VEHICLES							
Titan 34D / Transtage	Solid N_2O_4 / N_2H_4 - UDMH N_2O_4 / N_2H_4 - UDMH N_2O_4 / N_2H_4 - UDMH	246,288,000 * 529,000 101,000 16,000	10.2 10.0 10.0 10.0	90.4 78.6 37.0 14.7	1,514,600	4,200	-
Titan 34D No Upper Stage	Solid N_2P_4 / N_2H_4 - UDMH N_2P_4 / N_2H_4 - UDMH	246,288,000 * 529,000 101,000	10.2 10.0 10.0	90.4 78.6 31.3	1,492,200	27,600	-
Titan 2 / SLV	N_2O_4 / N_2H_4 - UDMH N_2O_4 / N_2H_4 - UDMH	430,000 100,000 (Vac)	10.0 10.0	70.2 23.4	340,000	4,200	-
Titan 3	Solid N_2O_4 / N_2H_4 - UDMH N_2O_4 / N_2H_4 - UDMH	246,288,000 * 529,000 101,000	10.2 10.0 10.0	90.4 78.6 31.3	1,492,200	31,600	-
Titan 4 Centaur G Prime	Solid N_2O_4 / N_2H_4 - UDMH N_2O_4 / N_2H_4 - UDMH LOX / LH_2	319,400,000 * 546,000 104,000 33,000	10.2 10.0 10.0 14.2	112.9 86.5 32.6 29.3	1,910,449	10,000	- - - -
Titan 4 / IUS	Solid N_2O_4 / N_2H_4 - UDMH N_2O_4 / N_2H_4 - UDMH Solid Solid	319,400,000 * 546,000 104,000 44,100 16,800	10.2 10.0 10.0 9.5	112.9 86.5 32.6 16.4	1,885,525	5,300	- - - - -
Atlas / Centaur	LOX / RP-1 LOX / RP-1 LOX / LH_2	377,500 60,000 33,000	10.0 -	140.5 / 104.7	360,600 / 293,000	5,200 / 3,000	3,500
Delta 3914 / Delta 3924	LOX / RJ-1 Solid N_2O_4 / N_2H_4 - UDMH Solid	205,000 767,000 9,850 / 10,000 10,000 15,000	8 3.3 8 3.2	73.4 36.6 19.3 6.8	420,500 / 425,300	2,065 / 2,430	1,390 / 1,670
Delta 3910 / 3920 / PAM-D	LOX / RP-1 Solid N_2O_4 / N_2H_4 - UDMH Solid	207,000 767,000 9,850 / 10,000 15,000	8 3.3 8 4	73.4 36.6 19.3 7.2	422,100 / 428,322	2,450 / 2,830	1,740 / 2,000
Delta 6920 / PAM-D IMLV	LOX / RP-1 Solid N_2O_4 / N_2H_4 - UOMH Solid	207,000 878,000 10,000 15,000	8 3.3 8 4	85.4 36.6 19.3 7.2	462,900	3,260	-
Delta 7920 / PAM-D	LOX / RP-1 Solid N_2O_4 / N_2H_4 - UDMH Solid	201,000 851,000 10,000 15,000	8 3.3 8 4	85.9 36.6 19.3 7.2	483,000	3,720	-
Scout / SLV-1A	Solid Solid Solid Solid	107,000 61,800 21,000 5,700	3.7 - - -	75.1 - - -	47,200	400 [10]	75
UPPER STAGES							
Centaur D-1A / D-1T	LOX / LH_2	33,000	10.0	30.0	35,000	5,200 / 17,500	3,500 / 13,000
Transtage	N_2O_4 / N_2H_4 - UDMH	16,000	10.0	15.0	27,000	4,200	4,000
Stage Vehicle System	Solid	15,500	4.6	10.3	5,520	-	
Orbit Insertion System	Solid	6,000	4.6	5.9	1,263	-	-
Stage Vehicle Sys. (SGS-II)	Solid	15,000	4.0	13.0	11,700	1,900	-
STS / PAM-A	Solid	35,200	5.0	7.5	12,760	4,400	2,530
STS / PAM-D	Solid	15,000	4.0	6.5	7,600	2,750	1,630
STS / PAM-DII	Solid	17,600	5.3	6.5	12,270	4,060	2,300
IUS	Solid Solid	44,100 16,800	9.5	16.4	32,311	5,000-6,000	11,023 / 3,307
Transfer Orbit Stage	Solid	44,100	9.6	10.7	24,010	13,400	7,930
Apogee & ManeuverStage	N_2O_4 / MMH	2,650	12.0	5.4	11,280	5,600	2,890
TOS / AMS	Solid, N_2O_4 / MMH	44,100 / 2,650	12.0	15.7	35,300	6,500	10,100

The US aerospace industry has responded recently to new commercial market opportunities and new military space programs by upgrading the capability of and reconfiguring proven ELV's. The lift capability to LEO depends on the orbital mission parameters (e.g., polar vs. equatorial), as well as on the launch site. This evolving lift capability for inserting payloads into Low Earth Orbit (LEO) and Geosynchronous Earth Orbit (GEO) orbits is shown in Table 3-2.[4]

TABLE 3-2. US ELV's EVOLVING LIFT CAPABILITY (REF. 4)

Vehicle	(Weight in lb.) Weight to Low Earth Orbit	Weight to Geosynchronous Transfer Orbit
Atlas Family - General Dynamics		
Atlas E	3,000	A
Atlas H	4,400	A
Atlas G/Stretched[B]	8,000	3,000
Atlas K	9,500	3,500
Atlas G/Centaur	13,500	5,200
Atlas Super G/Centaur	14,500	6,000
Titan Family - Martin Marietta		
Titan 2	4,200	A
Titan 34D/Transtage	31,650	9,500
Titan 34D/IUS	32,000	10,000
Titan 23E/Centaur D-1T	32,000	16,000
Titan 4/Centaur[B]	40,000	20,000
Delta Family - McDonnell Douglas		
Delta M-6	2,000	1,000
Delta 3920	5,500	2,800
Delta 4920[B]	C	3,900
Delta 5920[B]	C	4,400
Scout Family - LTV		
Scout	500-600	--

A. The rockets are not used in this orbit.
B. Proposed for development.
C. Not available.

3.2 LAUNCH VEHICLE TECHNOLOGY

Development of a reliable launch vehicle is an expensive and technologically demanding task, and only a limited number of nations have entered the field. This section provides a brief overview of launch technology in order to familiarize the reader with the terminology used in this report.[3-7]

Although there are many different and upgraded launch vehicle families, their basic technology is very similar. The fact that the same aerodynamics and gravity induced restraints apply to all such systems tends to dictate common design solutions. The fundamental elements of a space launch vehicle are: chemical propulsion systems, including support systems upper stages and tanks, guidance and navigation systems.

3.2.1 Propulsion Systems

Propulsion systems are based on Newton's laws of motion:

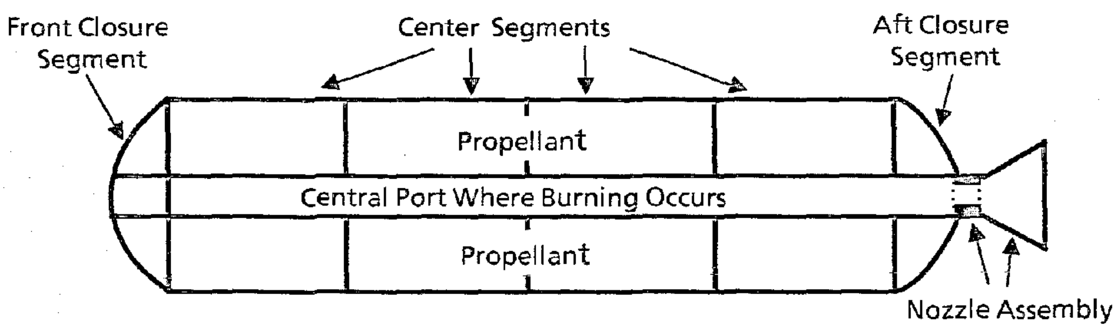
The first law requires that the engines overcome the inertia of the launch vehicle, and accelerate to the desired terminal (coast) velocity by the end of the thrust phase.

The second law requires applying net forces (F) to the launch system mass (m) (ELV plus payload) such that the thrust force upwards exceeds the downward gravity at launch; and such that the desired orbital plane change and velocity (v) is achieved during time interval Δt, using rocket impulse on orbit.

$$F \bullet \Delta t = \Delta m \bullet v + m \bullet \Delta v$$

The third (action-reaction) law is really the basis for a rocket motor. The motor expels "mass" (i.e., Δm due to exhaust gases) at high velocity through the rocket nozzle to achieve forward "thrust" for the launch system, at lower net velocity. As the ELV's mass decreases, it accelerates.[3]

The simplest propulsion system is the solid rocket motor (SRM), which contains fuel and an oxidizer in a binder that resembles hard rubber or plastic (Fig. 3-1). When ignited, a solid rocket motor burns



SOURCE: (Ref. 3).

FIGURE 3-1. A SEGMENTED SOLID PROPELLANT ROCKET MOTOR

rapidly until the fuel is exhausted, ejecting its burned products through a nozzle to generate the desired thrust. Such systems are reliable, relatively inexpensive to produce and may be stored for long periods of time. They do, however, have a lower specific impulse than liquid systems and therefore must be larger and heavier to obtain equivalent performance. In addition, solid rockets cannot be stopped and restarted once fired. As a result, some of the attractiveness associated with the lower cost of such systems is lost. Specific impulse (I_{sp}) is a key performance parameter, namely the thrust force (F) generated per unit mass of fuel burned per second or the weight rate flow of propellant. The specific impulse physical units are equivalent to the velocity of the exhaust gas at nozzle exit (m/sec, or ft/sec), although the convention is to express it in "seconds", given that thrust (F) is in lbs, and propellant flow rate (W) is in lbs/sec. If a propulsion system has an I_{sp} of 300 pound-seconds, it produces 300 pounds of thrust for every pound of propellant burned per second. A typical solid rocket motor operates in the 200 to 250 pound-second range, while the liquid system used in the Shuttle main engines can be as high as 450 pound-seconds.

The chemical rocket propellants commonly used are pairs of fuel/oxidizer compounds which react (during controlled combustion or "burning") to produce high temperature, high-pressure gases which escape from the combustion chamber with high nozzle velocities.[3] The chemical propellant mixture in rocket stages is tailored to provide a certain specific impulse and performance based on the efficiency of converting chemical to mechanical energy, the rate of burning (T,p) achievable and the mean molecular weight of combustion products. The characteristics and performance ranges for common solid fuel/oxidizer combinations and for typical liquid propellants are shown in Tables 3-3 and 3-4.

TABLE 3-3. SPECIFIC IMPULSE OF SOLID PROPELLANT COMBINATIONS

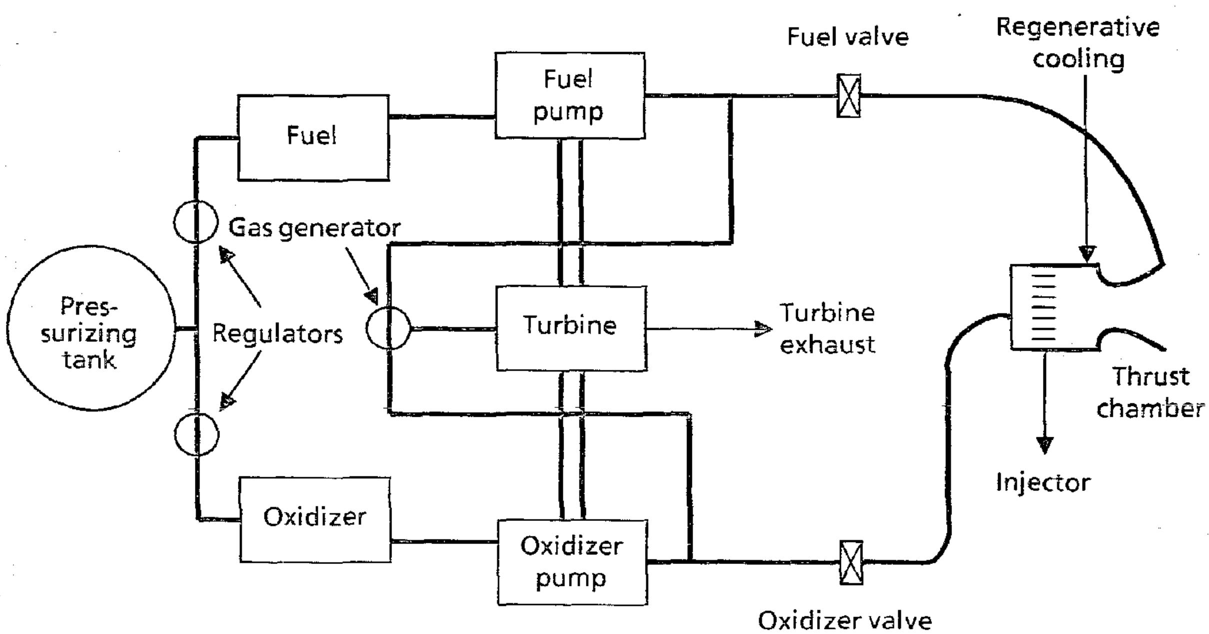
Fuel Base	Oxidizer	I_{sp} (seconds)
Asphalt	Perchlorate	200
Nitrocellulose and Nitroglycerine	----	240
Polyurethane	Perchlorate	245
Carboxy-terminated Polybutadiene (CTPC)	Perchlorate	260
Hydroxy-terminated Polybutadiene (HTPB)	Perchlorate	260
Cross-linked Double Base	----	270
Boron	Perchlorate	270
Metallic hydride	Fluoride	300

Liquid rockets typically keep the fuel and the oxidizer separated and bring these two elements together in a combustion chamber where they "burn"; the combustion products exit the system through a nozzle (Fig. 3-2). There are also "monopropellant" liquid rockets, where a single compound provides both fuel and oxidizer. Such systems are used for small auxiliary propulsion motors where the simplicity and lighter weight of a single tank counterbalances the lower specific impulse. Although the specific impulse of liquid fuels is significantly better than that of solids, liquid systems tend to be more complicated. For example, some require sophisticated pumps to maintain

TABLE 3-4. SPECIFIC IMPULSE OF LIQUID PROPELLANT COMBINATIONS*

OXIDIZER	FUEL					
	Ammonia	RP-1	UDMH	Aerozine-50 (50% UDMH & 50% Hydrazine)	Hydrazine (N_2H_4)	Hydrogen*
Liquid Oxygen (LOX)	294	300	310	312	313	391
Chlorine Trifluoride	275	258	280	287	294	318
95% Hydrogen Peroxide & 5% water	262	273	278	279	282	314
Red Fuming Nitric Acid (15% NO_2)	260	268	276	278	283	326
Nitrogen Tetroxide (N_2O_4)	269	276	285	288	292	341
Liquid Fluorine	357	326	343	----	363	410

*Assumes P_c = 1000 psia, optimum nozzle expansion ratio and P_o = 14.7 psia



SOURCE: (Ref. 3).

FIGURE 3-2. SCHEMATIC REPRODUCTION OF A LIQUID BIPROPELLANT SYSTEM

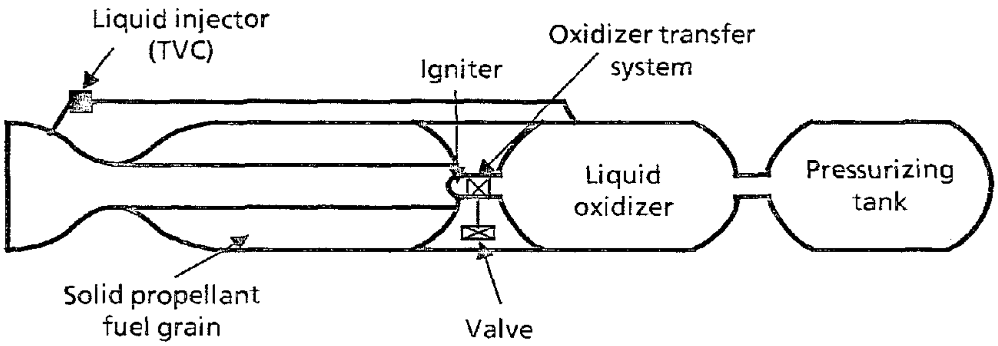
adequate flow of propellant to the combustion chamber (see Fig. 3-2). Such chambers must be designed to withstand extremely high temperatures and pressures, and sophisticated cooling systems are necessary to ensure high performance. However, not all liquid propellant systems require pump-delivery to the engine: For many upper stage bipropellant systems, including the Transtage, the propellant is fed into the engine by means of a high pressure inert gas head maintained in the propellant tank.

In addition, the most attractive propellants – liquid hydrogen and oxygen – are carried in cryogenic form and therefore require cryogenic systems to maintain their liquid state. This translates into complex and expensive systems for producing, storing, handling and loading these materials into the launch vehicle at the last possible moment to avoid excessive boil-off and loss of propellant. Cryogenic fuels also create special materials problems, such as embrittlement and fracture of tanks and other structural elements. The United States began using cryogenic technology in the late 1950's with the Atlas ICBM.

The difficulties involved in storing and handling cryogenic fuels prompted the development of "storable" liquid rockets. These have a lower specific impulse than rockets using cryogenic fuels, but can be fully fueled and held in readiness for an extended period of time. The obvious advantage of such rockets prompted the development of the Titan launch vehicle family as fully "storable" vehicles. The three liquid stages of the Titan/Transtage are fueled by Nitrogen Tetroxide (N_2O_4) and Aerozine-50 (Table 3-1).

The Titan launch vehicles series (see Sec. 3.3.1) is an example of a mixed system – i.e., one that has both solid and liquid stages. Solid and liquid propulsion systems can be mixed to match payload size to the mission velocity and orbital requirements. This is normally accomplished by adding solid "strap-ons" in addition to the liquid first stage engine. In this manner, basic core systems such as the Atlas or Titan can be upgraded and, when used with one of a number of existing upper stages, can produce a wide range of performance capability. This modular vehicle configuration affords greater mission flexibility and allows customization of lift performance according to orbital and launch mission profile.

In addition to the solid, liquid and mixed systems, hybrid rocket engines have been built that use a solid hydrocarbon fuel, but have a liquid oxidizer (LOX) (Fig. 3-3). Although less efficient than liquid or solid engines, hybrid systems are simpler, lower cost, safer and can have a start-stop-restart capability. Since the engine modules are inert until an oxidizer stream is fed in, termination of combustion is quick and simple (cut off oxidizer stream) and detonation/conflagration hazards are avoided. Work began on hybrid engines in the United States in the mid-1950's, but this technology has never been widely used. The American Rocket Company (AMROC) has tested hybrid rocket motors for its Industrial Launch Vehicle (ILV) family, which burn a solid hydrocarbon fuel (polybutadiene) with LOX oxidizer and offer some safety and efficiency advantages over operational ELV propellants[9] (see Vol. 2, Ch. 5).



SOURCE: (Ref. 3).

FIGURE 3-3. HYBRID ROCKET ENGINE

Along with the major rocket motors and propulsive stages, there are a variety of auxiliary propulsion systems used to control spacecraft or vehicle attitude. Such systems are used for station-keeping or for altitude and position adjustments, to clear away casings or protective structures, to provide spinning motion and for a variety of similar tasks that require small amounts of thrust.

A great deal of current research is devoted to advanced cryogenic liquid propulsion technology, for longer life, greater reliability and lower cost (e.g. redesigning high-pressure fuel and oxidizer turbopumps). A storable propellant upper stage is also under development, to enhance lift capability to Geostationary Earth Orbit (GEO).

3.2.2 Support Systems and Tanks

Launch vehicles are designed to be as light as possible in order to provide the best possible thrust-to-weight ratios. However, the vehicle must be strong enough to resist bending, absorb the thrust of the main propulsion system and survive various mechanical, thermal and environmental stresses. The tension between these two goals has stimulated creative designs (e.g., the use of a pressurized tank skin as a structural element in the Atlas vehicle) and the development of several new high-strength-to-weight materials.

Solid fuel rockets are normally heavier than liquid fuel rockets because the entire propellant casing functions as a combustion chamber and must be able to contain the internal heat and pressure of the burning fuel. In liquid systems, the tank can be much lighter (thinner "skin") since combustion takes place in a confined region in the rocket engine. As a result, the structure of a liquid fueled vehicle is dictated by the necessity to support the payload weight and aerodynamic loads during prelaunch and launch operations. For cryogenic liquid rockets, the need for tank insulation to reduce loss of propellant can result in heavier structures. Newer materials such as graphite fiber composites are lighter and tougher and are replacing steel for nose cones and tanks. However, shielding requirements still "weigh down" launch vehicles. New solid rocket propellants under development will be less hazardous to the environment (produce less hydrogen chloride by-products). Other new formulations (beryllium and beryllium hydride) are being tested. X-ray non-destructive test and acceptance methods have long been used to certify solid rocket motors, but new computer tomagraphy technology is now being introduced for SRM testing.

Better understanding of vehicle structural dynamics has developed as a consequence of the cumulative experience of the United States with a large number of different types of launch vehicles. Problems associated with high performance launch vehicle structures require further R & D efforts and technology transfer by the US Government. Conservative design or over design of structures is likely in such cases, with consequent loss in performance.

3.2.3 Guidance Systems

The guidance system of a very sophisticated launch vehicle provides information on the position, altitude, acceleration and velocity of the vehicle, and may allow adjustment of the direction and magnitude of thrust. Sophisticated guidance systems are based on an inertial platform with precise acceleration and attitude sensors that provide information to an on board computer that controls the subsystems.(3)

Inertial guidance systems are a sophisticated technology; there are, however, simpler radio or radio inertial systems that can be readily obtained. Such systems were used by the United States, the Soviet Union, Japan and China when they first developed expendable launch vehicles. Relatively simple inertial guidance systems have become more widely available and later generation space vehicles, such as the Ariane and India's SLV-3, have incorporated them from the beginning of their operations.

3.2.4 Upper Stages

Expendable launch vehicles such as the Titan, Delta and Atlas/Centaur are built in sections called "stages." As each stage completes its burn during the launch process it is discarded, thereby increasing the velocity that can be imparted to the payload by succeeding stages during thrusting. The first stage is usually called the "booster stage," the second stage the "sustainer stage," and subsequent stages "upper stages" (also see Ch. 4). The upper stages of most expendable launch vehicles are capable of placing payloads into highly elliptical "transfer orbits." Once this is accomplished, the payload can either remain in this orbit or move to a higher orbit, by use of a payload assist module (PAM), or to a circular orbit by use of a small rocket (apogee kick motor, AKM, perigee kick motor, PKM). Generally, communication satellites, which are placed in a geosynchronous stationary orbit, (see Chs. 4, 6) require such rocket motors for orbital insertion and station-keeping.

3.3 REPRESENTATIVE ELV's

This brief and sketchy description of the basic US launch vehicles is based on References 1-5. Although the ELV industry is introducing new design concepts and upgrading capabilities, the basic vehicle characteristics remain valid.

3.3.1 Titan

Designed originally by the Air Force for its own needs, the Titan was first manufactured under contract by Martin Marietta in 1955. Since then, the Titan has been upgraded and configured in several different ways. Titan I was a two stage Intercontinental Ballistic Missile (ICBM) using liquid oxygen and RP-1. Titan II (larger than Titan I and also an ICBM) was deployed in 1962; this version, using storable liquid propellant, was converted to the Titan/Gemini launch vehicle and used for 12 launches from 1964 to 1966. The Titan III family was developed to provide a wide spectrum of satellite mission capabilities. Titan 34D is a two stage solid and liquid propellant launch vehicle designed to launch heavy (30,000 pounds) payloads into LEO. With the addition of a third (upper) stage, called a Transtage, additional capability can be achieved. Without the Transtage, 31,600 pounds can be placed into a 115 statute mile circular orbit. With the Transtage, about 9,500 pounds can be placed into a geosynchronous equatorial orbit (see Table 3-2). The Transtage contains an inertial guidance system and an attitude control system. It has a multi-start capability and provides the propulsive maneuvers for achieving a variety of circular and elliptical orbits. As Tables 3-1 and 3-2 indicate, the IUS is an alternative upper stage for the basic Titan. The Titan III type vehicle is being adapted for commercial use as the Titan 3. Figure 3-4 illustrates the Titan IIIC (similar to the 34D plus the Transtage). Titans can be launched from both East Coast and West Coast pads, although, commercial missions will be primarily from Florida.

3.3.2 Delta

The Delta, manufactured by McDonnell Douglas, was derived from the Thor Intermediate Range Ballistic Missile (IRBM) in 1959 and has become one of the most used US launch vehicles, especially by NASA. It has been constantly upgraded during its history and is capable of placing over 2,800 pounds into a geosynchronous transfer orbit, depending upon its configuration (see Table 3-2).

A two or three stage configuration can be used with the Delta. The first stage, or booster, is an elongated Thor missile with 3, 6 or up to 9 Castor 4, 4A or graphite epoxy (GEM) solid strap-on motors. The second, or Delta stage, is liquid fueled with restart capability. Vehicle guidance is accomplished by an inertial system in the Delta stage. The third stage is a payload assist module (PAM) with a solid rocket motor. Figure 3-5 shows a three stage configuration for the Delta.

Designated as its Medium Lift Vehicle (MLV) by DOD, the Delta II configuration will also be available for commercial launches.

Delta payloads have included scientific, meteorological, earth resource and communications satellites. It can be launched from the Eastern Space and Missile Center (ESMC) in Florida and the Western Space and Missile Center (WSMC) in California.

3.3.3 Atlas/Centaur

Manufactured by General Dynamics, the Atlas is based on the Atlas ballistic missile and was first used as a space booster in 1958. The Centaur is a family of liquid fueled upper stages specifically designed for the Atlas which can accommodate a wide variety of payloads and upper stages. Centaur has also been adapted to Titan boosters. The Atlas/Centaur has been used primarily to launch communications satellites and also for low earth orbit, lunar, planetary and synchronous transfer orbit missions. In the latter role, the Atlas/Centaur version can place up to 5,200 pounds into geosynchronous transfer orbit (see Table 3-2). General Dynamics is offering the Atlas G/Centaur for commercial use. Figure 3-6 depicts a standard Atlas/Centaur configuration.

3.3.4 Scout

The Scout is a research and/or sounding rocket developed in the late 1950's to launch small payloads into low Earth orbit and to conduct suborbital studies on reentry. Currently, the Scout employs four solid propellent stages. Produced by Vought Industries, the vehicle is capable of lofting a 400 pound payload into a 345 statute mile orbit launched due east. There are currently 11 vehicles in inventory, all of which are committed; Table 3-2 details planned upgrades. Figure 3-7 shows a typical Scout launch vehicle configuration.

3.3.5 Vehicles Under Development

Several other firms are developing vehicles specifically for commercial use. Space Services Inc. is developing the all solid rocket Conestoga launch vehicle based on the Minuteman and Aries motors (off-the-shelf tested technology), but as an upgrade to Scout vehicles, to carry low and medium class payloads (300-2,000 pounds, typical of remote sensing satellites) into low earth orbit (see Figure 3-8).[10]

American Rocket Company (AMROC) has developed the Industrial Launch Vehicle (ILV) family: 1) the ILV-1 is a four stage hybrid rocket designed to place 1,400 kg in a 250 km circular polar orbit; 2) the ILV-S is a three stage variant capable of placing 225 kg in the same orbit; and 3) the SMLV, a derivative of the single module test vehicle, will be used for suborbital missions[9] (see Figure 3-9). E'Prime Aerospace Corporation is developing STAR A (a modified Scout rocket) and Star B (based on

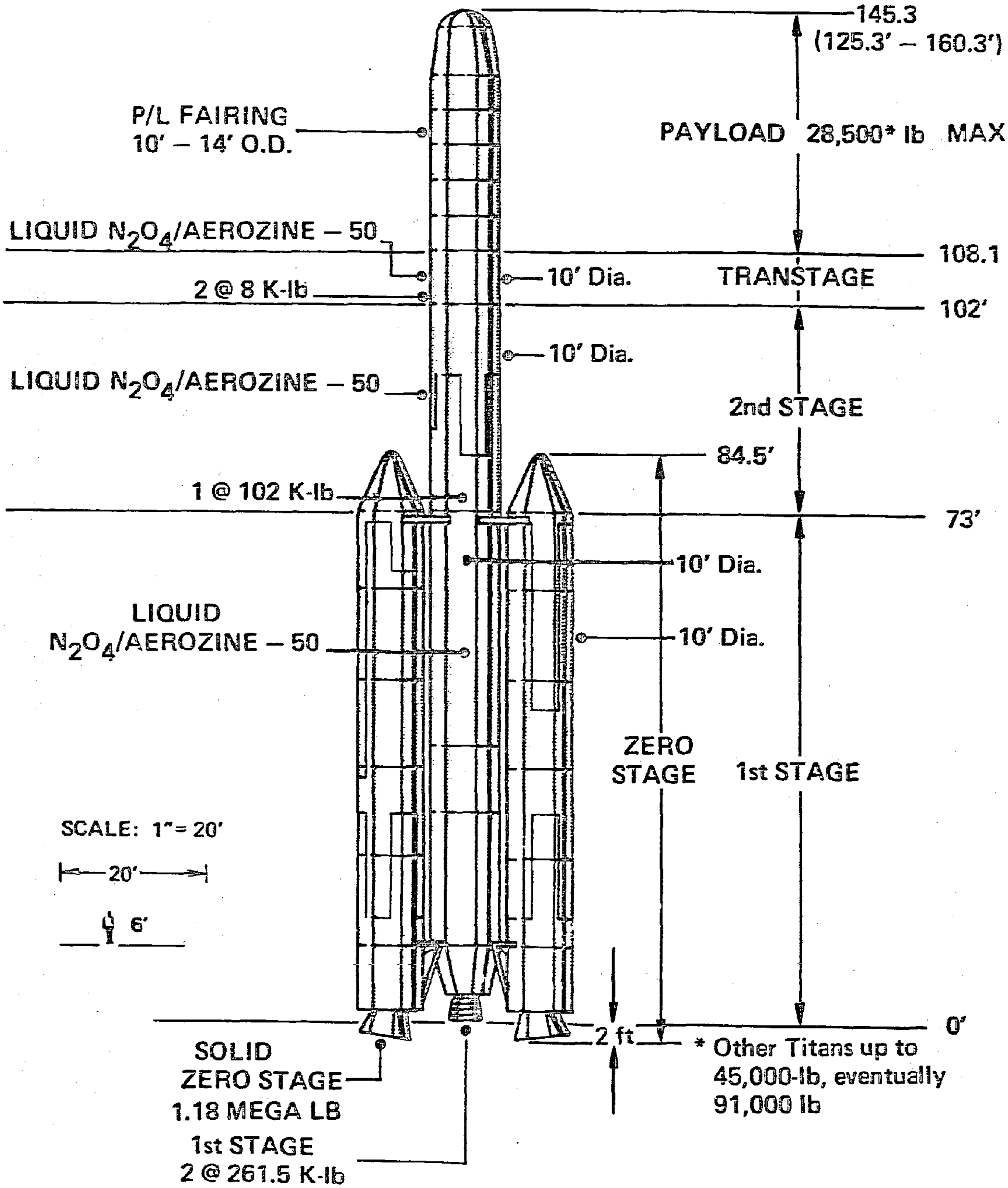
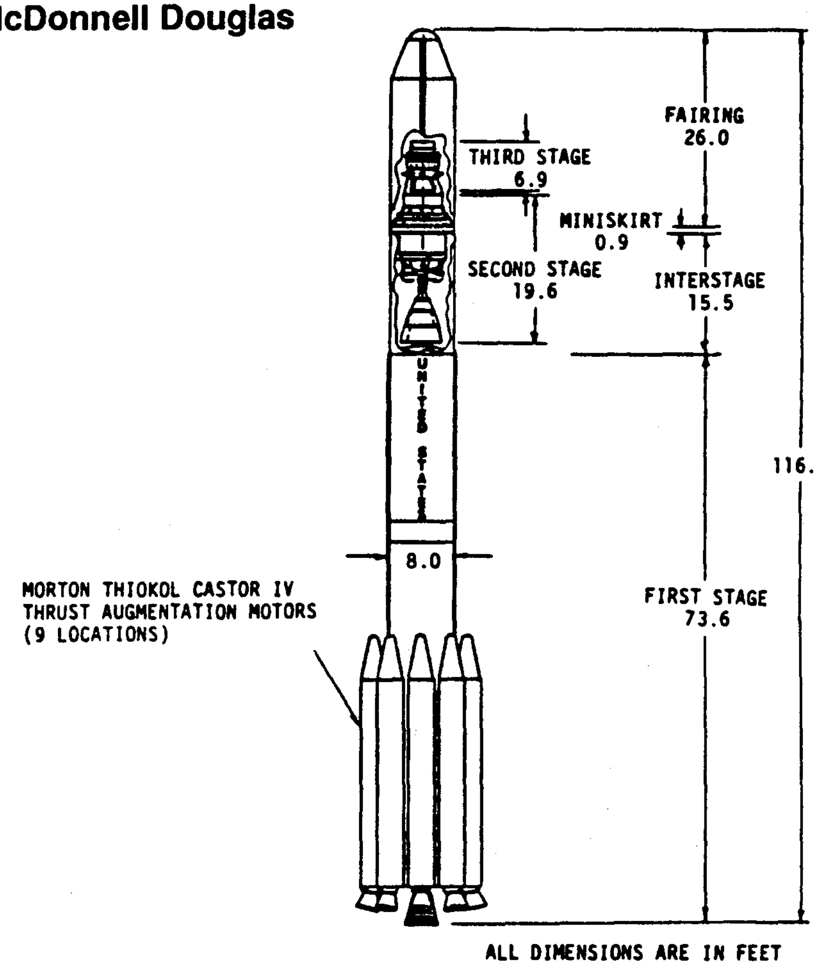


FIGURE 3-4. SLV-5C, TITAN IIIC (REF. 4)



DELTA 3920/PAM
LAUNCH VEHICLE CONFIGURATION

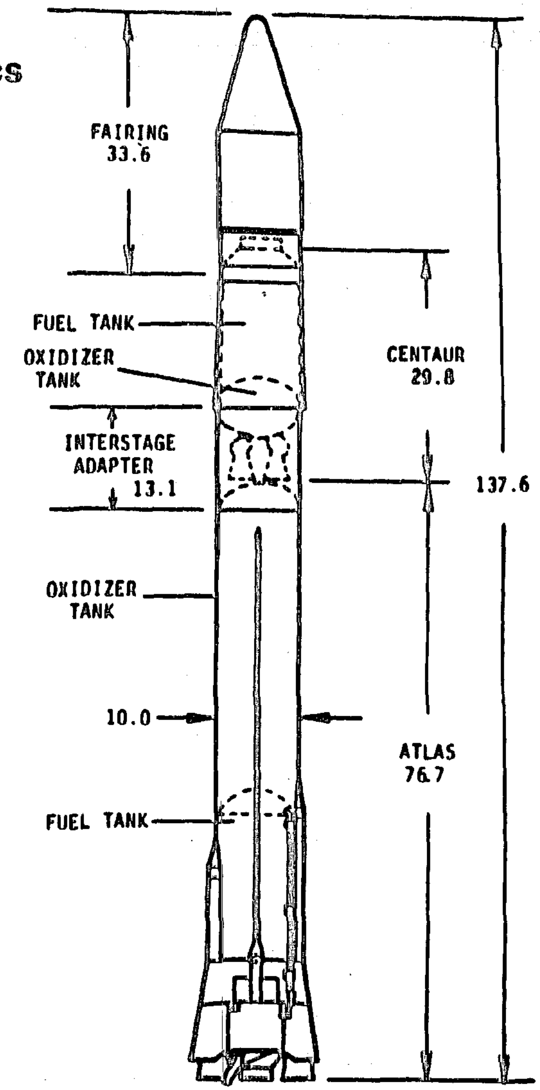
General Stage Data
Delta 3941/3920-PAM Vehicle Delta

	Boosters	Stage 1	Stage 2 (3914/3920)	Stage 3
Stage Data				
Designation	Castor IV	ELT Thor	Delta	TE-M-364-4/PAM
Stage Mass, Klbm	214	187	13.5/15.1	2.5/4.8
Usable Propellant, Klbm	185	175	10.0/13.2	2.3/4.4
Number of Engines	9	1	1	1
Stage Length, inch	439	1,071	232/235	77/82
Stage Diameter, inch	40	96	55/68 (Excluding Miniskirt)	37/49
Guidance Data				
Manufacturer			McDonnell Douglas	
Type			Strap-Down Inertial	
Engine Data				
Manufacturer	Thiokol	Rocketdyne	TRW/Aerojet	Thiokol
Designation	TX-526-2	RS-27	TR-201/AJ10-118K	TE-M-364-4/STAR48
Number of Starts Possible	1	1	Multiple	1
Fuel	Solid	RP-1	Aerozine 50(1)	Solid
Oxidizer	Solid	LOX	N_2O_4	Solid
Mixture Ratio	--	2.23	1.6/1.9	--
Average Thrust per Engine, lbf				
Sea Level	83,640	204,990	--	--
Vacuum	92,400	229,600	9,530/9,710	14,800/15,130
Average Chamber Pressure, psia	540	650	100/114	550/570
Specific Impulse, sec				
Sea Level	230	261	-/-	-/-
Vacuum	256	294	303/320.2	283/293
Total Burn Time, sec	56	227	318/435	44/85.3
Nozzle Expansion Ratio	8.0	8	46/65	33/54.9
Nozzle Exit Area, ft^2	6.1	12.0	17.4/19.9	3.14/4.75
Engine Cant Angle, deg	11	0	0	0
Thrust Vector Control	--	Gimballed Engine	both Gimballed Engine	both Spin Stabilized

(1) Aerozine 50 is an equal mixture of UDMH and hydrazine.

FIGURE 3-5. DELTA 3920/PAM LAUNCH VEHICLE CONFIGURATION AND DATA

General Dynamics

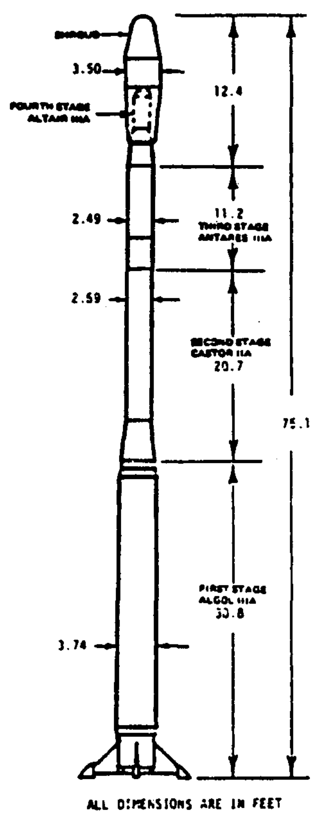


General Stage Data
Atlas Centaur Launch Vehicle

	Stage 1/2		Stage 1	Stage 2
Stage Data				
Designation		Atlas G		Centaur D-1A
Stage Mass, Klbm		320.675		33.777
Usable Propellant, Klbm		300.632		29.734
Stage Length, ft		76.7		29.8
Stage Diameter, ft		10		10
Number of Engines	2		1	2
Guidance Data				
Manufacturer				Honeywell
Type				Four Gimbal Inertial
Engine Data				
Manufacturer	Rocketdyne		Rocketdyne	Pratt and Whitney
Designation	YLR-89-NA-7		YLR-105-NA-7	RL-10A-3-3A
Number of starts possible	1		1	2
Fuel	RP-1		RP-1	LH_2
Oxidizer	LOX		LOX	LOX
Mixture Ratio, O/F	2.25		2.22	5.0
Average Thrust per Engine, lbf				
Sea Level	188,750		60,500	—
Vacuum	—		—	16,500
Average Chamber Pressure, psia	640		733	474
Specific Impulse, sec				
Sea Level	259		220	
Vacuum	292		312	446.4
Total Burn Time, sec	153		283	404
Nozzle Expansion Ratio	8		25	61
Nozzle Exit Area, ft^2	11.24		11.54	8.22
Engine Cant Angle, deg	0		0	0
Thrust Vector Control	Gimballed Engines and Verniers			Gimballed Engine

FIGURE 3-6. ATLAS/CENTAUR LAUNCH VEHICLE CONFIGURATION AND DATA

Ling - Temco - Vought



General Stage Data
Scout Launch Vehicle

	Stage 1	Stage 2	Stage 3	Stage 4
Stage Data				
Designation	ALGOL IIIA	CASTOR IIA	ANTARES IIIA	ALTAIR IIIA
Mass, Klbm	32	11	4	0.7
Usable				
Propellant, Klbm	28	8	3	0.6
Number of Engines	1	1	1	1
Guidance Data				
Type			Open Loop Inertial	
Manufacturer			Honeywell	
Engine Data				
Manufacturer	UTC	Thiokol	Thiokol	Thiokol
Designation	N/A	TX354-3	TE-M-762	TE-M-640
Number of Starts	1	1	1	1
Fuel/Oxidizer	Solid	Solid	Solid	Solid
Average Web Thrust				
Sea Level, lbf	93,250	*	*	*
Vacuum, lbf	105,112	63,970	18,700	5,710
Average				
Chamber Pressure, psia	450	700	700	670
Specific Impulse, sec				
Sea Level	229	*	*	*
Vacuum	259	280	295	288
Web Burn Time, sec	56.11	35.32	43.96	29.30
Nozzle Expansion Ratio	6.48	21.2	46.0	50.3
Nozzle Exit Area, ft^2	5.67	7.95	4.48	1.5
Engine Cant Angle, deg	0	0	0	0
Thrust Vector Control	Aerodynamic Fins and Jet Vanes	H_2O_2 RCS	H_2O_2 RCS	Spin Stabilized

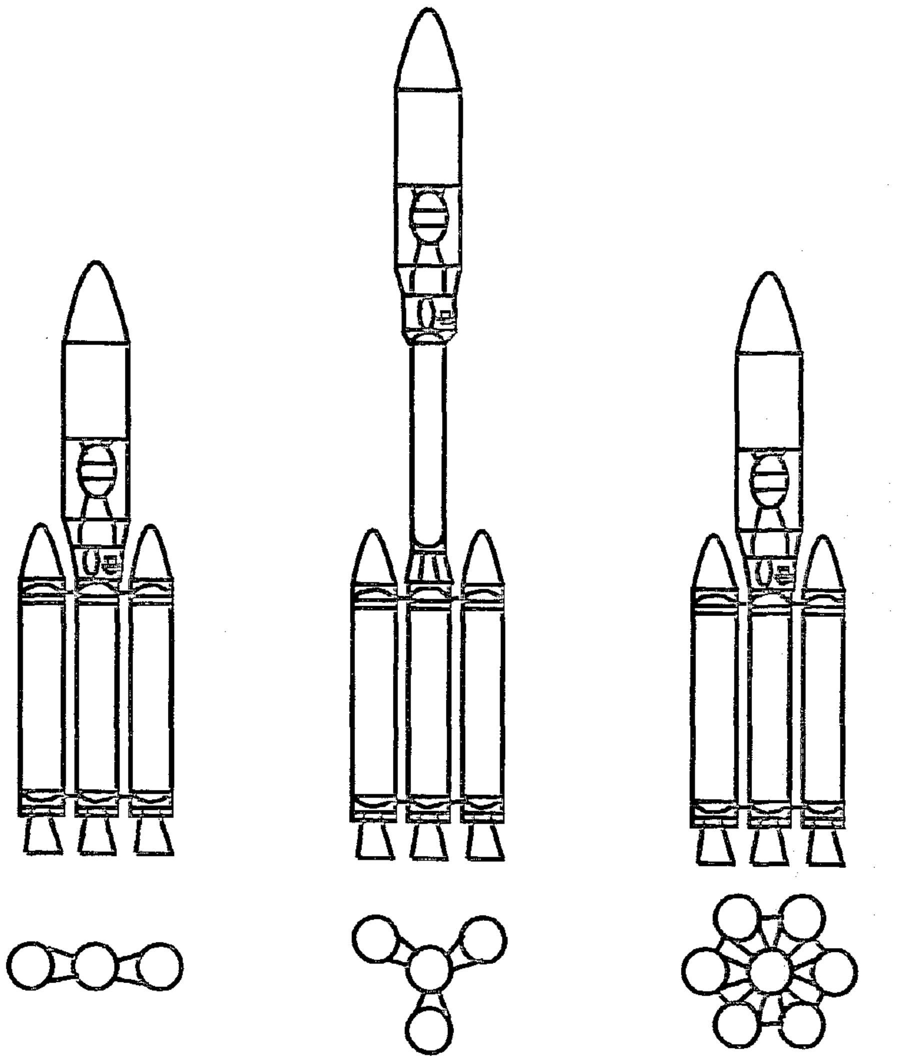
* = Not Applicable

FIGURE 3-7. SCOUT G-1 LAUNCH VEHICLE CONFIGURATION DATA

	CONESTOGA II	CONESTOGA III	CONESTOGA IV

Vehicle Profile

Base Configuration



Motors by Stage

	CONESTOGA II	CONESTOGA III	CONESTOGA IV
Stage One	2 Castor V	3 Castor V	4 Castor V
Stage Two	1 Castor V	1 Castor V	2 Castor V
Stage Three	1 STAR 37FM	1 Castor 1VA	1 Castor V
Stage Four		1 STAR 37FM	1 STAR 37FM

FIGURE 3-8. CONESTOGA LAUNCH VEHICLE CONFIGURATIONS FOR THE ILV FAMILY

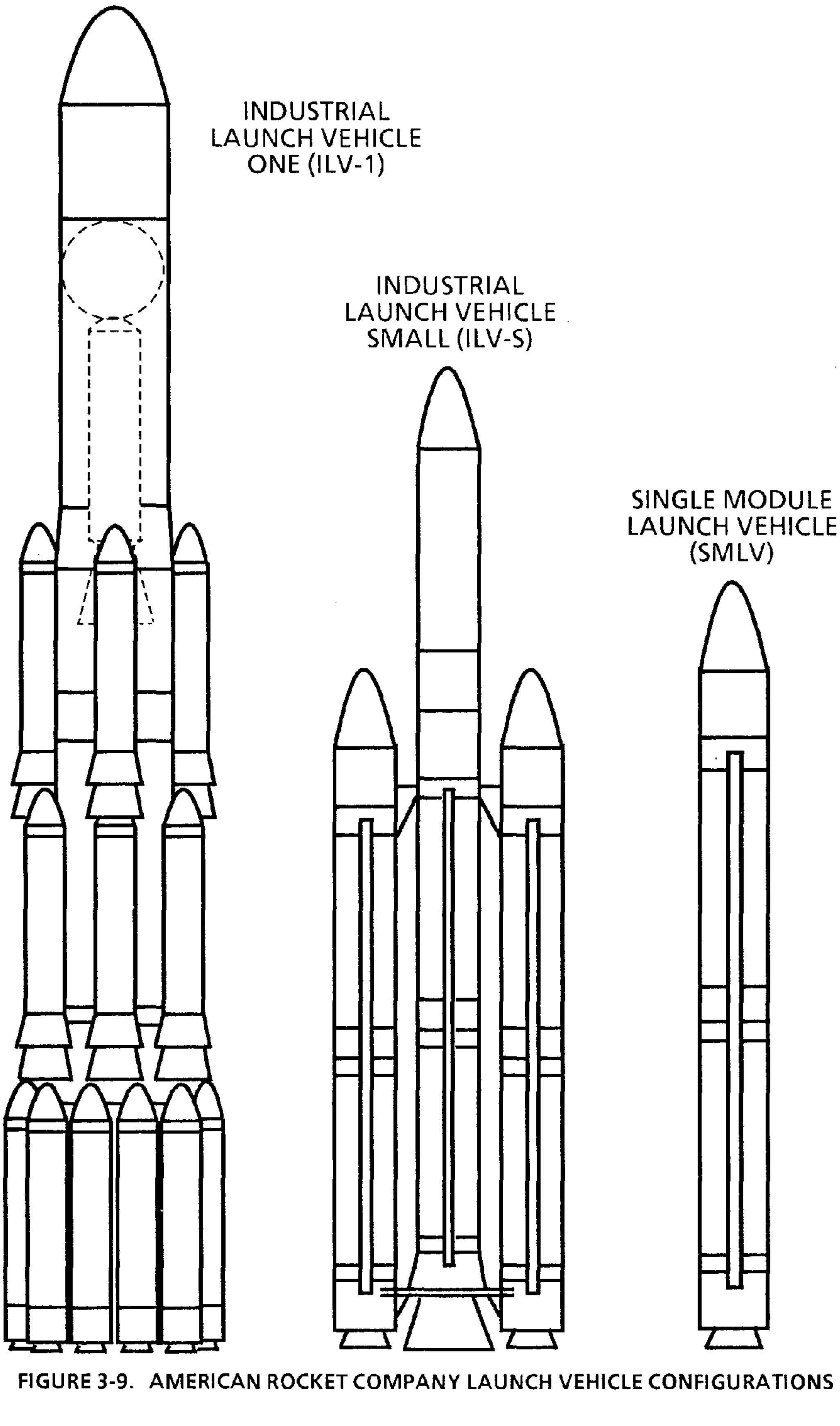


FIGURE 3-9. AMERICAN ROCKET COMPANY LAUNCH VEHICLE CONFIGURATIONS

Peacekeeper missiles) with lift capacities of 900 and 35,000 lbs to LEO; and 400 and 12,000 lbs to GEO, respectively. Other small entrepreneurial companies are offering a full range of launch services, including purchase of launch vehicle and access to launch facilities. An example is Conatec, which will launch retrievable payloads using Black Brant suborbital sounding rockets from the White Sands Missile Range.

Historical US launch vehicles are shown to scale in Figure 3-10. While the discussion in this report focuses on historical experience with ELV's that are currently operational, new configurations and innovative designs will probably be developed commercially in the future.

3.4 ELV LAUNCH EXPERIENCE

Table 3-5 shows the publicly available performance/reliability statistics for some of the US launch vehicles that being considered for commercial space launches.[8] This table excludes launches of ICBM-type vehicles that are in the same family as vehicles used for orbital applications. It is apparent that all ELV's currently have achieved over 90%, and some up to 98%, operational performance reliability, figures which are highly competitive internationally.

Such vehicle reliability information is useful for providing worst case inputs to casualty and damage expectation estimates, which supplement specific engineering design failure probability data. Analysis of actual historical launch failures also provide information that can be used to improve assumptions and calculations built into the casualty expectation models, as discussed in Vol. 3, Chapters 9 and 10. These data, while useful, contribute only indirect information relevant to the analysis of public risk exposure. Detailed failure mode statistics by vehicle, and those accident scenarios with potential public safety impacts, are needed as inputs to any computation of casualty expectation. Launch or Mission failures, as reflected in Table 3-5, may not necessarily have any adverse public safety impacts, given the Range Safety process (Chapter 2). For example, out of a statistical data sample of 486 launches through 1986, including Titans, Atlas, Delta, Ariane and STS missions, 442 (or about 91%) were successful.[7] Out of the 44 launch failures, only one (the Titan 34D April, 1986 catastrophic accident at VAFB, see Ch. 10) resulted in any land impact, with substantial damage to the launch site facilities. None involved any third party damage and/or casualty claims. In the case of NASA, out of 290 launches (of Delta and Atlas/Centaur) conducted, and 33 failures (P_f = 11%), only 1 had catastrophic consequences ($P_{cat} = 3 \times 10^{-3}$) (damage to the VAFB launch facilities), but had only negligible public impacts (see Ch. 10).

Here is another example of how reliability figures can reflect on public safety: According to NASA records on the Delta vehicle launch performance and failure history, out of the 12 failures experienced in 25 years in over 180 launches (P_f = 6.6%), only 4 required destruct actions by Range

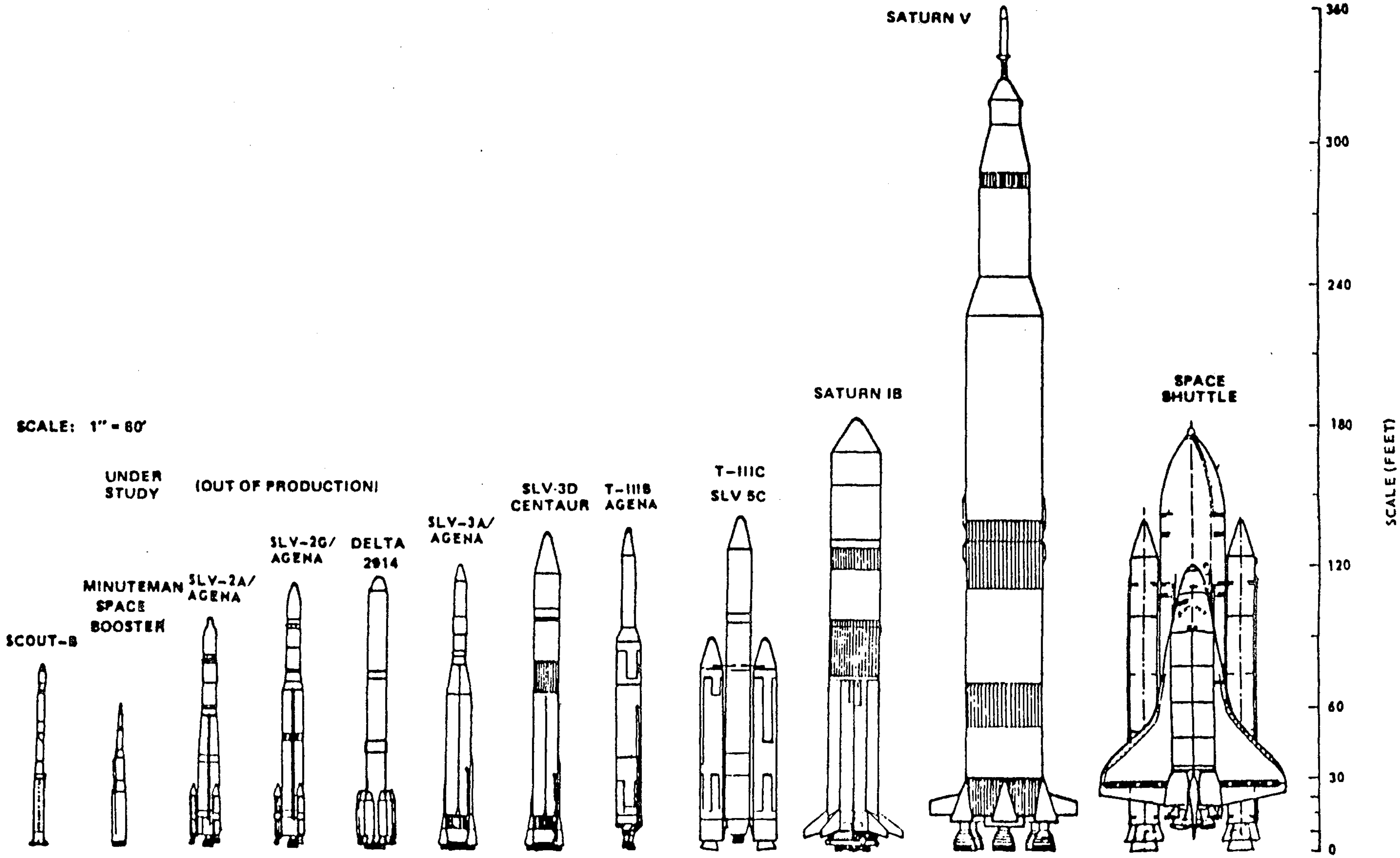


FIGURE 3-10. HISTORICAL U.S. LAUNCH VEHICLES

TABLE 3-5. ELV LAUNCH RELIABILITY (Ref. 8)
(As of December 1987)

	% SUCCESS	% FAILURE
SCOUT		
TOTAL PROGRAM STATISTICS SINCE 1960: (INCLUDING DEVELOPMENT FLIGHTS)		
• 108 LAUNCHED	87	13
• 14 FAILED		
RECENT PROGRAM STATISTICS		
• 54 LAUNCHED SINCE 1967	96.3	3.7
• 2 FAILED		
DELTA		
TOTAL PROGRAM STATISTICS: (INCLUDING DEVELOPMENT FLIGHTS)		
• 181 LAUNCHED	93.4	6.6
• 12 FAILED		
RECENT PROGRAM STATISTICS		
• LAST FAILURE OCCURRED IN MAY 1986; PRIOR TO THAT 43 LAUNCHES WERE SUCCESSFUL, 1 SINCE. IN THE PAST 10 YEARS, 46 SUCCESSES OUT OF 47 LAUNCHES	97.9	2.1
ATLAS/CENTAUR		
TOTAL PROGRAM STATISTICS: (INCLUDING DEVELOPMENT FLIGHTS)		
• 67 LAUNCHED	85.1	14.9
• 10 FAILED		
RECENT PROGRAM STATISTICS		
• 25 LAUNCHED SINCE 1977 (FAILURE IN 1977)	92.0	8
• 2 FAILED (AC-62) 1984		
ATLAS E, F - MODIFIED BALLISTIC MISSILES		
TOTAL (USAF + NASA) PROGRAM STATISTICS:		
• 80 LAUNCHED	90.0	10.0
• 8 FAILED		
RECENT NASA ONLY		
• 9 LAUNCHED	88.9	11.1
• 1 FAILED		
TITAN III		
TOTAL		
• 137 LAUNCHED	96.4	3.6
• 5 FAILED		
ARIANE		
TOTAL		
• 19 LAUNCHED	79.9	21.1
• 4 FAILED		

Safety (P_d = 2.2%), and only 1 (P_{li} = $3X10^{-3}$) involved re-entry of fragments with land impact with no damage reported, although 5 led to re-entry of spacecraft stages and debris ($P_{re\text{-}entry}$ = 2.7%). Hence ELV reliability figures are only a very rough guide to the potential for adverse public impacts of launch failures, because only a small fraction of these has the potential to inflict public property damage and injuries. There was only one launch site Delta fire in the early 1960's which resulted in personnel injury and death (1^{st} party), but the recurrence of such an event has been made very unlikely by improved on-site safety procedures.

It is important to note that past performance is not necessarily a good indicator or predictor of future performance. Typically, in Government space operations to date, components and subsystems that have been identified as contributing to or causing a system failure were upgraded or procedures were modified, so as to avoid their recurrence. This is a major reason why the recent ELV reliability statistics in Table 3-5 show considerable learning and performance improvement over total and average program statistics. From such launch failure statistics and from ELV manufacturer and sponsoring Agency responses to Mishap Reports and to accident investigations for failed launches, it also appears that in the early operational life of an ELV, the failure rate may reach or exceed 25 percent, but gradually decrease to under 5 percent as experience results in corrective improvements in the launch vehicle system, and in both ground and Range Safety procedures. This "failure, analyze and fix" philosophy, in combination with technical improvements (e.g., in guidance, avionics, communications and control) and upgrades to each generation ("block") within a family of ELV's, has contributed to few recurring or "common cause" failures over the years and to demonstrated ELV reliability growth over time.

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8. a) Delta Launch Vehicle Accident Investigation, Hearing Before the Subcommittee on Space Science and Technology, House of Representatives, Ninety-Ninth Congress, Second Session, June 5, 1986, No. 125; b) also Aviation Week and Space Technology, Dec. 7, 1987, p. 73 for Scout and Nov., 1987 issues for Titan and Ariane updates in fall 1987.

9. "Design Status of the American Rocket Company Industrial Launch Vehicle (ILV) Family", J.R. French and M.D. Griffin; presented at 1st Annual Utah State University Conference on Small Satellites, Oct. 1987; updated Feb. 1988.

10. a) "Conestoga Vehicle and Launch Facilities", Space Services Inc. of America; attached to 4/22/1988 D.K. Slayton letter to Courtney Stadd, OCST Director; b) "Range Safety Report for the Conestoga I Sub-Orbital Launch Vehicle", June 1982, by Space Vector Corporation for Space Services Inc.

4. LAUNCH AND ORBITAL OPERATIONS*

4.1 PHASES OF LAUNCH THROUGH ORBITAL OPERATION

Launch and orbital operations can be divided into two or three phases:

(1) the initial launch and boost phase which terminates when the vehicle obtains the velocity and altitude necessary to achieve Earth orbit;

(2) the orbital transfer phase, during which properly timed firings of rocket motors move the satellite into the desired final orbit; and

(3) depending upon the mission, return from orbit. Re-entry is further discussed in Vol. 2, Ch. 7.

4.1.1 Launch Phase

The prime objective during the launch phase is for the boost vehicle to overcome Earth's gravitational pull, rise through the atmosphere and overcome frictional heating. It must provide a satellite with an initial vertical and final orbital velocity (almost parallel to the surface of the Earth) using sustainer and upper rocket stages which will keep it in orbit. Depending on the latitude of the launch point, the desired orbital inclination and altitude, the initial orbit may not be the final orbit for the satellite. To change inclination the boost and higher stages of the ELV must rotate the attitude of the vehicle, so that it will be moving in the proper direction, and then pitch over to the orbital plane gradually as it gains velocity and altitude.

The gradual programmed pitchover (called a gravity turn) is carefully designed so that the angle of attack (the angle between the axis of the vehicle and the vector of the aerodynamic forces) is kept as close to zero as possible. The gravity turn is preceded by a small pitchover maneuver called the "kick angle." If this is not accomplished, the aerodynamic loads on the vehicle will build up and overcome the guidance and control system, thereby producing a deviation from the planned flight path. If the angle of attack becomes too large, the airloads may over-stress the vehicle and cause its structural failure. The aerodynamic force effects are proportional to one half of the product of the local atmospheric density and the vehicle velocity squared (called dynamic pressure or "q"). In some vehicles, failure can begin at less than 10 degrees angle of attack during the "high q" portion of flight. A typical trajectory profile is shown in Figure 4-1.

*The information in this chapter was developed using the references listed at the end of the chapter. This material is intended for readers with little or no background in either orbital mechanics or rocketry. Others can proceed directly to Chapter 5.

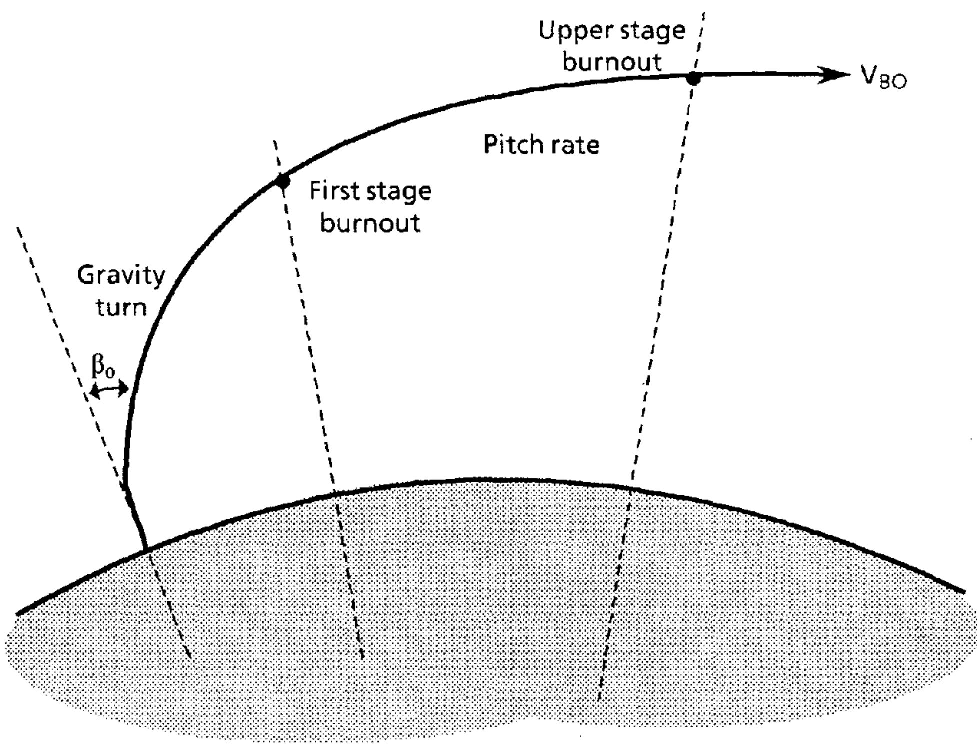


FIGURE 4-1. TYPICAL TRAJECTORY PROFILE

When the vehicle reaches a very high altitude, the atmospheric density becomes so low that the dynamic pressure is essentially zero regardless of the velocity. After this, the zero angle of attack is no longer required and different pitch attitudes and pitchover rates can be used.

Control of all launch vehicles is maintained by gimballing (tilting) the engine nozzles or some equivalent way for changing the direction of the engine thrust. Launch vehicles must be controlled continuously because they are, without exception, aerodynamically unstable, i.e., a slight increase in angle of attack will cause the aerodynamic forces to attempt to increase the angle of attack even further. Severe wind shears during the early post-launch period of flight create difficulty for most vehicles, as the guidance and control systems must act to minimize the pitching or yawing due to abrupt angle-of-attack changes which they create.

Most launch vehicles contain several stages. Thrust is initially provided by the lowest (and largest) boost stage. When the fuel for this first stage is consumed, the spent fuel casing is jettisoned to Earth, the remainder of the vehicle separates from it and the next stage is fired to continue the flight. Part of the preparation for any mission is the planning for the impact location of the spent stages (and other jettisoned equipment) in order to minimize the risk to people and property on the ground (see Ch. 2).

Most of the current launch vehicles use solid rockets fastened to a central core vehicle which is usually a liquid propellant stage. These "strap-on" solid rocket motors (SRM's) augment the first stage thrust and are jettisoned when their propellant is consumed.

4.1.2 Orbital Insertion and Orbital Operations

It is not possible to describe the myriad of possible orbital parameters which may be desired or designed for different mission objectives. This discussion will only briefly cover the very simplest example. Consider the sequence of events illustrated in Figure 4-2. In the first illustration (a), a satellite (with a booster stage) is placed in a low "parking" orbit around the Earth. The rockets are fired in orbit and then shut off. The result of this orbital correction firing is the creation of a new elliptical "transfer" orbit which has an apogee (greatest distance from the Earth) which is at a higher altitude above the Earth than the original orbit (Figure 4-2(b)). If the satellite has no further propulsion, it will continue to follow this elliptical orbit indefinitely, passing (ideally) through its initial perigee point once very revolution. If the objective is to reach a higher circular orbit, the built-in rockets (apogee kick motors, AKM) can be fired again (for a specified period of time) when the satellite reaches the apogee of the elliptical orbit, and the new orbit will be as shown in Figure 4-2(c).

4.1.3 Orbital Decay and Re-entry

Once out of the densest portion of the atmosphere, the ELV and its payload (satellite) has only very small drag forces acting upon it to reduce the satellite velocity. Consequently, the satellite will continue to orbit until reverse thrust (retro-propulsion) is applied for a planned re-entry or decay forces eventually cause an uncontrolled re-entry. Controlled descent from an orbit reverses the firing sequence for orbit transfer. Rocket engines fire for a determined interval and angle and the vehicle/satellite now follows an elliptical orbit with apogee at the original orbital altitude and perigee at an altitude much closer to the Earth. If the perigee is within denser portions of the atmosphere, the vehicle/satellite will start to slow down gradually because of aerodynamic drag and descend to the Earth sooner due to orbital decay (see Vol. 2, Ch. 7). Aerodynamic heating is intense because of the very high vehicle velocity as it is coming out of orbit and the slow initial braking during re-entry. Objects not designed to withstand this heat by protection from a heat and ablation shield generally break up and, often, vaporize altogether. Re-entry vehicles (RV's) similar to those provided for ICBM's have been proposed for recoverable payloads.

Satellites which are placed in very low Earth orbit may not need any propulsion to return from orbit. Even at an altitude of 200 miles, the very low density of air molecules still applies a small, but continuous drag force. These satellites will very slowly lose both velocity and orbital altitude and the decay will gradually increase until the object is traveling slow enough to re-enter the Earth's

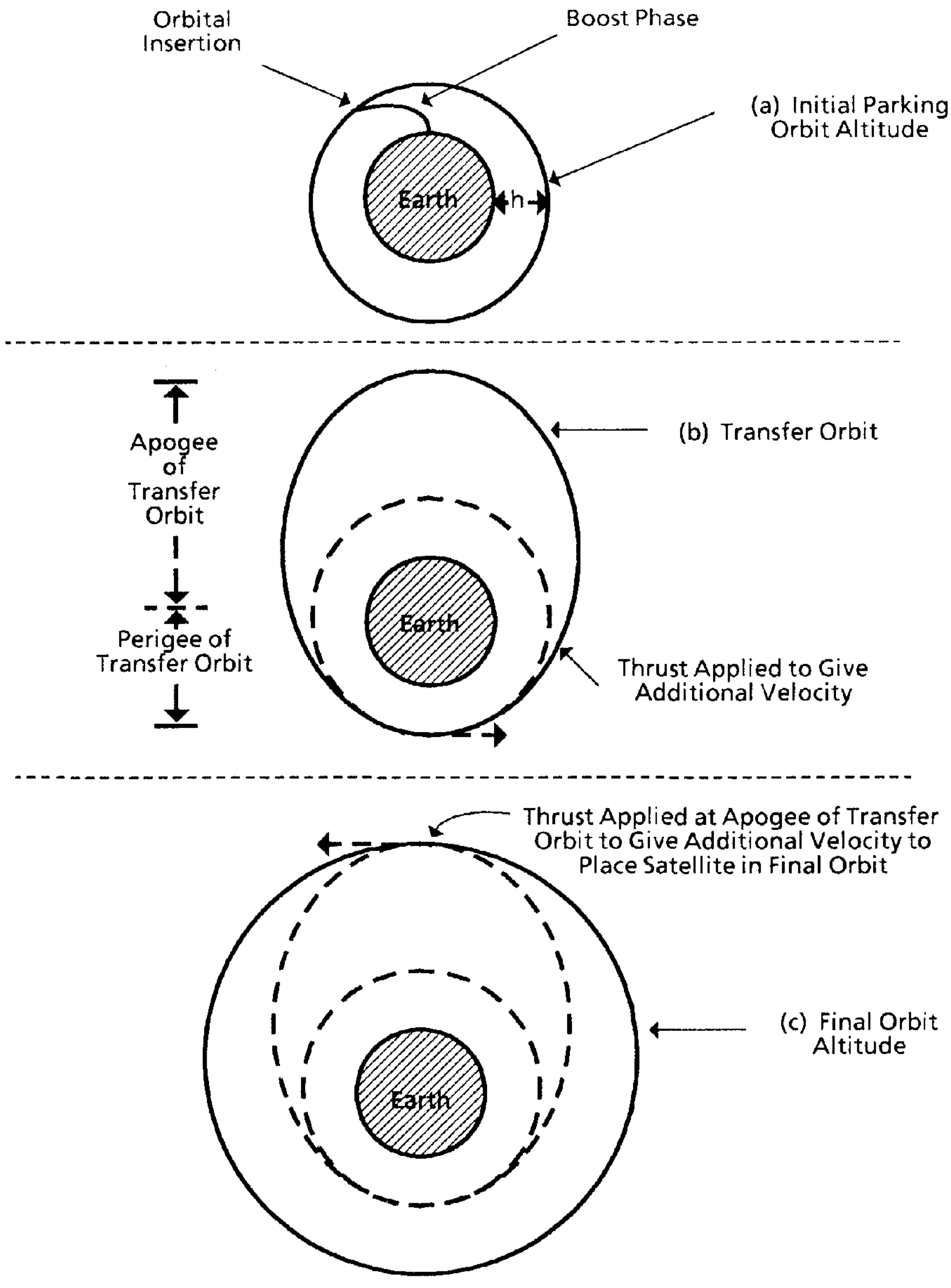


FIGURE 4-2. ORBITAL TRANSFER

atmosphere. This unplanned re-entry is discussed further in Ch. 7. Figure 4-3 shows approximate orbital lifetimes for satellites in circular orbits. Orbital lifetime is a direct function of the mass to drag ratio of the satellite. This ratio is represented by the ballistic coefficient β which is equal to W/C_DA; where W is the weight, C_D is the drag coefficient of the body, and A is the cross-section area. The shaded area in the figure shows the range of lifetime in orbit for objects whose ballistic coefficients range from 10 to 300 lb/ft^2. The larger values of ballistic coefficient correspond to the longer lifetimes in the shaded region shown in Figure 4-3.

If rocket engines are used to de-orbit, as proposed for recoverable payloads that use re-entry vehicles (RV's), the potential hazard from the re-entering spacecraft is controllable. However, the hazard from a decayed satellite re-entry is uncontrolled and usually cannot be predicted with any accuracy (see Ch. 7).

4.2 BASIC ORBITAL CHARACTERISTICS

A satellite stays in orbit because the centrifugal (outward) force equals the Earth's gravitational pull (inward). The centrifugal force is proportional to V^2/R, where R is the distance from the center of the Earth to the satellite and V is the component of satellite velocity which is perpendicular to the radius R. The gravitational pull decreases with distance and is proportional to $1/R^2$. For low Earth orbits (LEO), the gravitational pull is stronger and, consequently, satellites must have a higher velocity to compensate and, thus, circumnavigate the globe much more rapidly. Figure 4-4 shows the relationship between orbital velocity and altitude above the surface of the Earth for circular orbits. Figure 4-5 gives the period (the time required to complete one circular orbit) as a function of altitude above the surface of the Earth.

Not all orbits are circular; many are elliptical and are employed in orbital transfer and other mission applications. The *perigee* of an elliptical orbit is the minimum altitude of the orbit; the *apogee* is the maximum altitude (see Figure 4-6). The eccentricity is a measure of the ellipticity of the orbit. The formula for eccentricity is:

$$e = \frac{r_a - r_p}{r_a + r_p} \tag{4-1}$$

where r_a is the distance from the center of the Earth to the apogee altitude and r_p is the distance from the center of the Earth to the perigee altitude. The apogee and perigee altitudes for a circular orbit are equal, hence a circular orbit has zero eccentricity. Elliptical orbits having the same perigee altitude as a circular orbit always have a longer period, with the period increasing with the eccentricity.

The free flight path of a suborbital rocket or an expendable launch vehicle (ELV) is also elliptical. These vehicles, after completion of powered flight, follow a ballistic trajectory with an apogee above the surface of the Earth and a perigee which is below the surface of the Earth (see Figure 4-6).

The concepts of energy and angular momentum are essential in understanding orbital mechanics. The total mechanical energy has two components, *kinetic energy* (K.E.) and *potential energy* (P.E.).

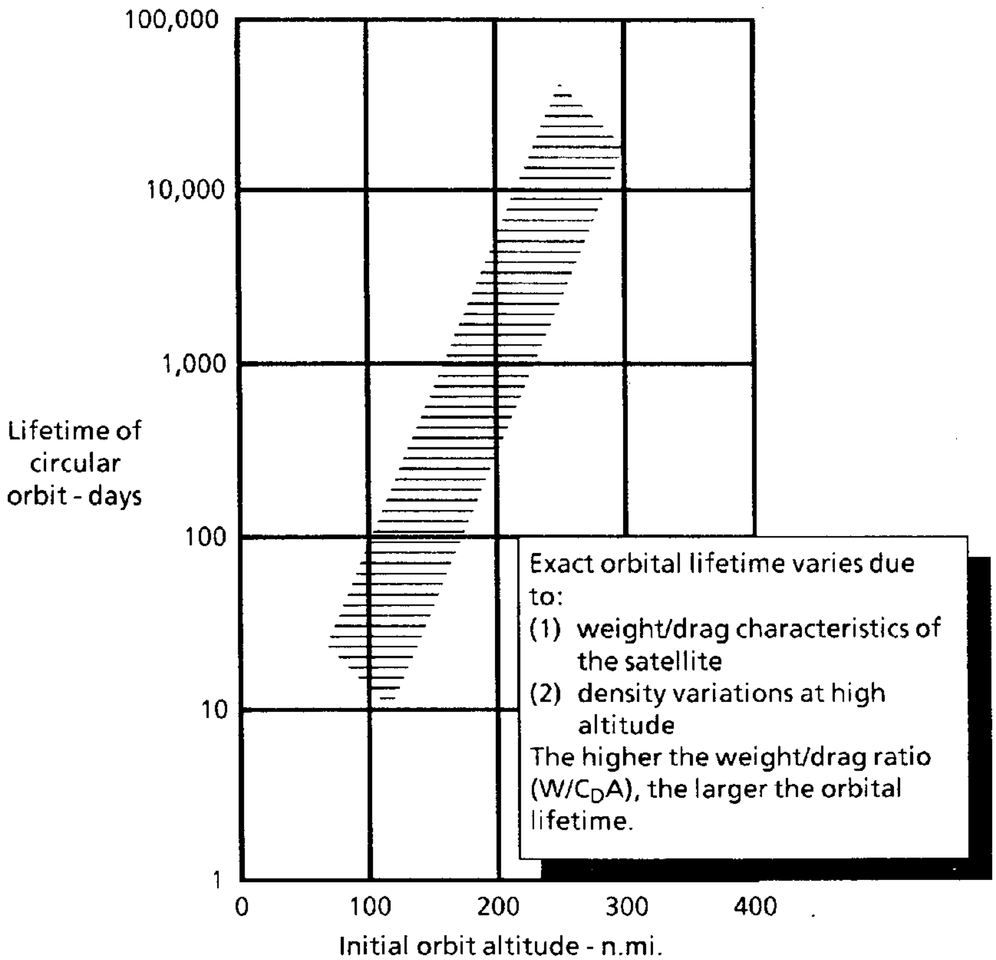


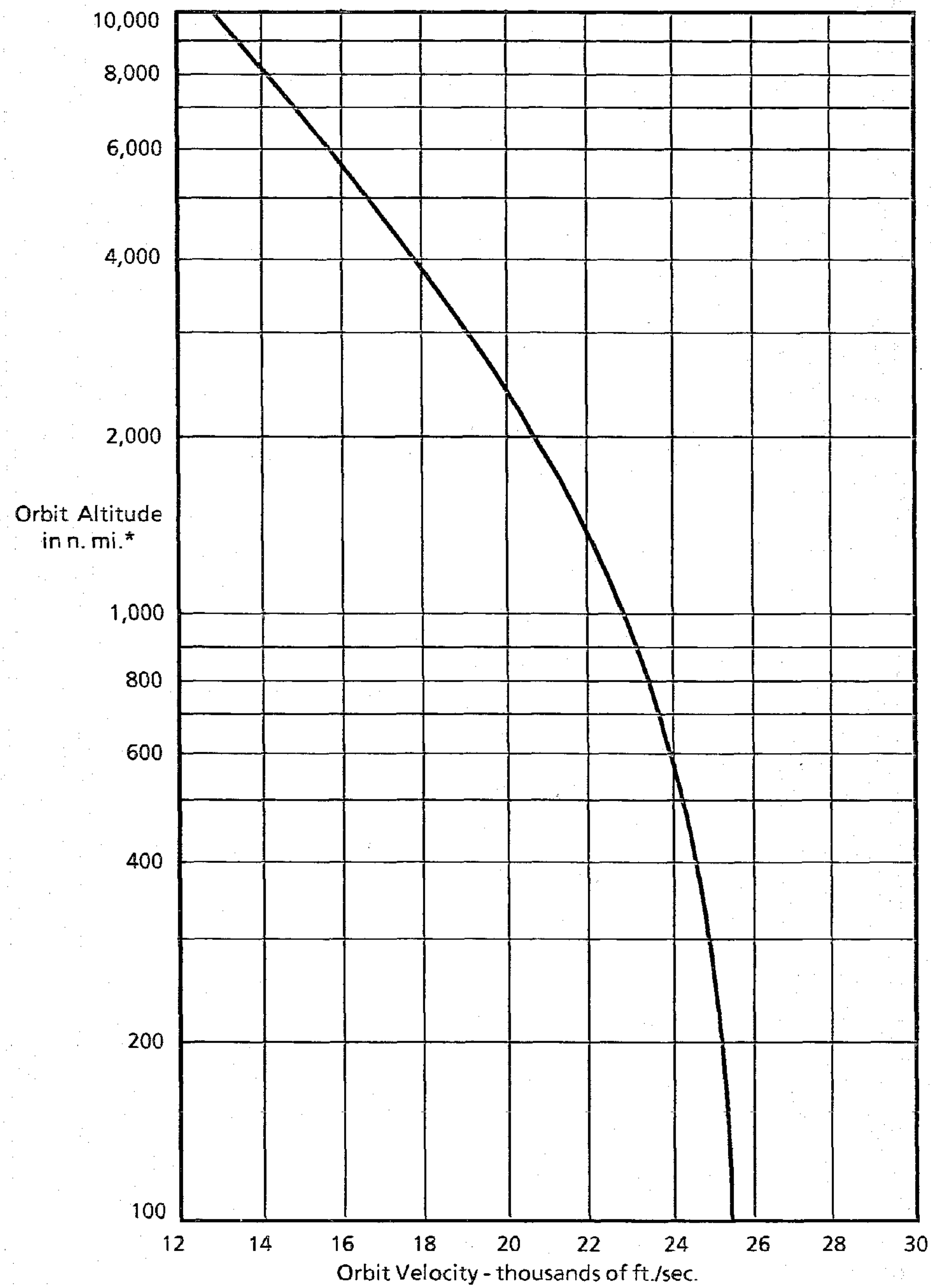
FIGURE 4-3. APPROXIMATE LIFETIMES FOR SATELITES IN CIRCULAR ORBITS

As long as no additional force is being applied to the satellite (e.g., aerodynamic or rocket thrust), the total energy of the satellite remains constant, i.e.,

$$\text{Total mechanical energy} = \text{K.E.} + \text{P.E.} = \text{constant} \tag{4-2}$$

The kinetic energy is proportional to the square of the velocity of the satellite. Potential energy results from the combination of gravitational attraction and distance to the gravitational source. The total energy per unit mass, E, will remain constant throughout the orbit (circular or elliptical) unless a force impulse, such as rocket thrust or drag, is applied to the satellite. Thrust in the direction of the velocity vector will increase the energy and thrust or drag in the direction opposite to the velocity vector will decrease the energy.

Hence, an orbiting satellite has both Kinetic Energy: $KE = mv^2/2$ and Potential (Gravitational) Energy: GmM/r at its orbit altitude ($r = R + h$; where R is the Earth's radius, h is altitude above the Earth and M is the Earth's mass). The constant $\mu = GM$ in $(ft/sec)^3$ or $(m/sec)^3$ is the constant product of the Universal Gravitational constant and Earth's mass.



* A **nautical mile** (6076.166 feet) is a measure commonly used in orbital mechanics. There are 60 nautical miles to a degree of latitude. In contrast, a statute mile has 5280 feet or 1760 yards.

FIGURE 4-4. SATELLITE VELOCITIES IN CIRCULAR ORBITS

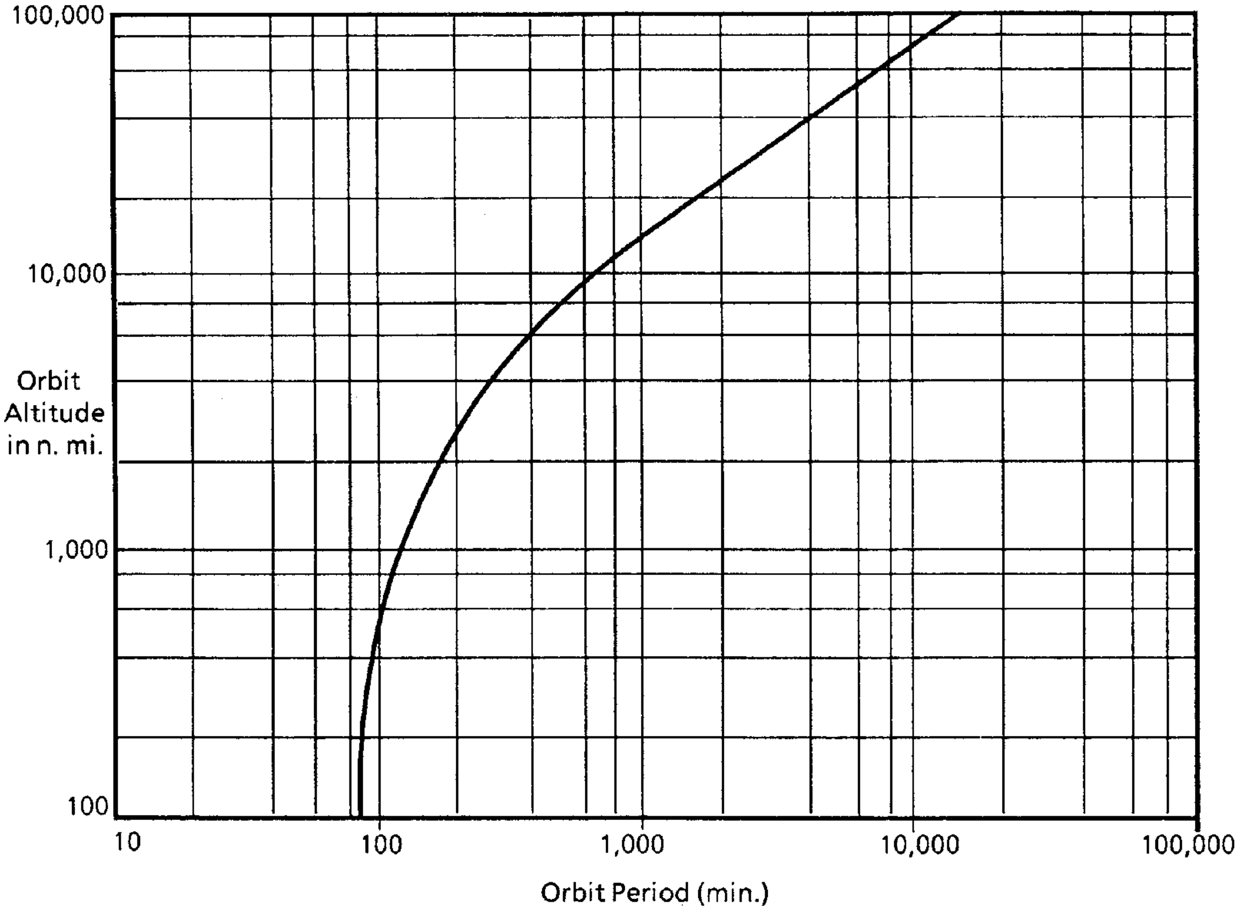


FIGURE 4-5. PERIODS FOR CIRCULAR ORBITS

This simplifies the total energy per unit mass for an orbiting satellite to a specific mechanical energy:

$$E_s = \frac{E}{m} = \frac{KE + PE}{m} = \frac{v^2}{2} - \frac{\mu}{r} = \text{const.} \qquad (4\text{-}3)$$

If $E_s<0$, the path is parabolic; if $E_s = 0$, the satellite is in a captive orbit (elliptical, or circular). If $E_s>0$, the path is hyperbolic and the satellite will escape Earth's gravitational pull. The escape velocity is obtained from:

$$\frac{v_{esc}^2}{2} - \frac{\mu}{r} = 0 \qquad v_{esc} = 36{,}700 \text{ ft/sec or approximately 12 km/sec} \qquad (4\text{-}4)$$

For launch velocities below v_{esc} , the satellite will either return to Earth (suborbital injection velocities) and follow a ballistic (parabolic trajectory) or orbit in a circular or elliptical orbit with a speed (v) and period (P) determined as below:

$$v = \sqrt{\frac{\mu}{r}} \qquad P = \frac{2\Pi r}{v} \qquad (4\text{-}5)$$

Two body gravitational interactions and no energy dissipation are assumed for the present discussion. The effects of solar wind, atmospheric drag and luni-solar perturbation on orbital

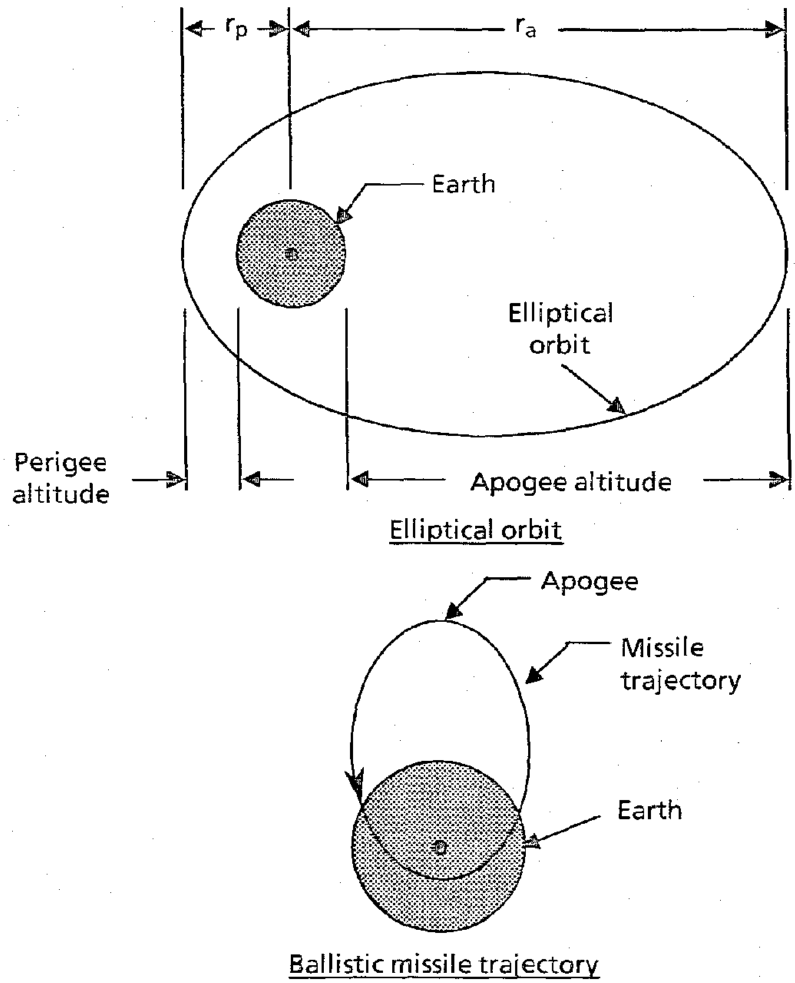


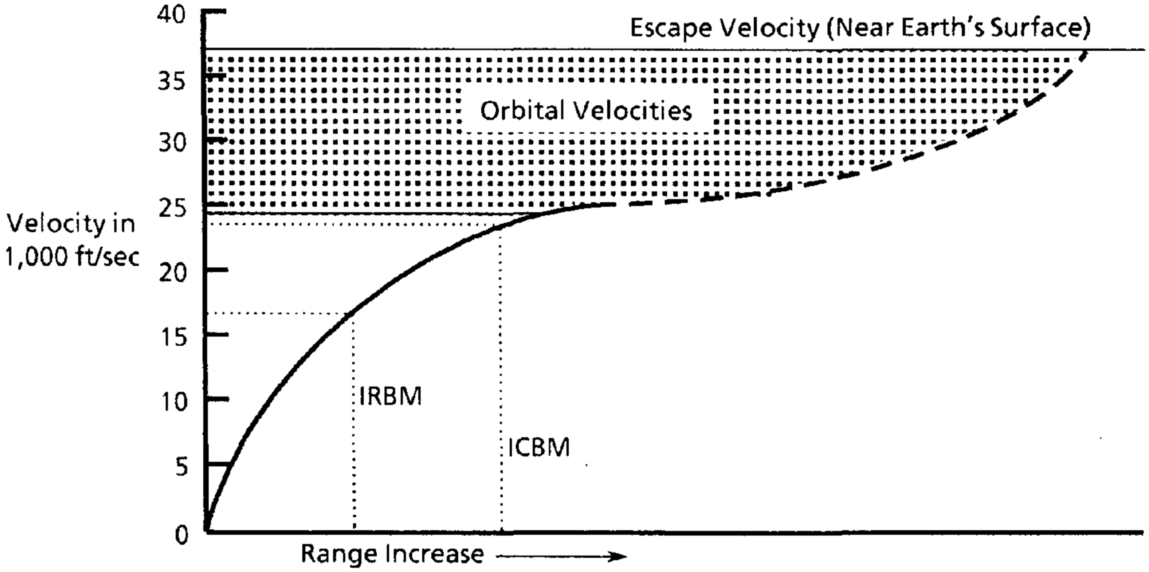
FIGURE 4-6. GEOMETRY OF SATELLITE ORBITS AND ELV TRAJECTORIES

parameters and decay are discussed in Ch. 7. Figure 4-7 shows the velocity vs. range for a rocket and payload.

Since energy is conserved, it is now possible to visualize the exchange between potential and kinetic energy in an elliptical orbit. When the satellite is nearest to the Earth (perigee), the potential energy is least and the kinetic energy is at its peak. Hence the satellite reaches its highest velocity at the perigee and its lowest velocity at the apogee (where the potential energy is highest).

The kinetic energy required for different orbits can be related to a *characteristic velocity*. The characteristic velocity is also the summation of all the velocity increments attained by propulsion to establish the desired orbit. Table 4-1 (from Ref. 1) describes the characteristic velocities for a number of missions.

Angular momentum is also a conserved quantity, so that without an external application of torque for a period of time, a spinning body will neither increase nor decrease its rate of spin. Satellite



(Source: Ref. 2)

FIGURE 4-7. VELOCITY vs. RANGE OF A ROCKET

orbits have an angular momentum, which is about an axis through the center of the Earth. The orbital angular momentum, H, is given by:

$$H = R \cdot v \cdot \cos\theta \tag{4-6}$$

where H is the angular momentum, R is the distance from the satellite to the center of the Earth and θ is the angle between the velocity vector and a line in the orbital plane which is perpendicular to the position vector (see Figure 4-8). The product "$v \cdot \cos\theta$" can also be referred to as the *tangential velocity*. H is constant except when the satellite is accelerated or decelerated by thrust or drag.

The equations for conservation of energy and angular momentum are necessary to analyze the dynamics of satellite orbits. The oblateness of the Earth requires some additional terms over those shown in Equation 4-3 for the potential energy expression, to obtain more accuracy in the orbital predictions; and the gravitational fields of the Moon and the Sun, in particular, should also be considered in increasing prediction accuracy.

The plane of the orbit is defined by the longitude of the *ascending node* and its *inclination*. These are shown in Figure 4-9. The ascending node is the point where the projection of the satellite path crosses the celestial equator from south to north. The inclination is the angle formed by the plane of the orbit and the equator. It is measured counterclockwise from the eastern portion of the equator to the ascending node. Thus, satellites which orbit west to east (normal or prograde) have an inclination $<90^\circ$; orbits going east to west (retrograde) have an inclination $>90^\circ$. An alternate method sometimes used to designate retrograde inclination is to measure the angle clockwise from the western portion of the equator and state it as an X° retrograde inclination (see Fig. 10-8). A third

TABLE 4-1. MISSION VELOCITY REQUIREMENTS

MISSION	Characteristic Velocity V_{CH} ft/s	Excess Velocity Over Reference Orbit V_{ex}, ft/s
100 nmi Reference Circular Orbit	25,570	0
200 nmi Circular Orbit	25,922	352
500 nmi Circular Orbit	26,900	1,325
1000 nmi Circular Orbit	28,296	2,726
Synchronous Transfer Ellipse	33,652	8,082
Lunar Impact	36,035	10,465
Escape, Parabolic	36,164	10,594
ETR - Synchronous - 28.5° Inclination*	38,490	12,920
WTR - 24 Hour Orbit - "Polar"*	38,503	12,933
Synchronous Equatorial (28.5° Plane Change)	39,791	14,120
Venus Flyby	37,570	12,000
Mars Flyby	37,770	12,200
Mercury (Via Venus Flyby)	38,570	13,000
Jupiter Flyby	36,070	20,500
Saturn (Via Jupiter Flyby)	46,570	21,000
Uranus (Via Jupiter Flyby)	47,270	21,700
Neptune (Via Jupiter Flyby)	50,070	24,500
Pluto (Via Jupiter Flyby)	54,270	28,700
Lunar Landing Return	56,000	N/A

* For an ETR orbit altitude of 19,323 nmi and a WTR orbit altitude of 19,355 nmi.

term often used to describe orbits is the right ascension (Ω). This is the arc of the celestial equator measured eastward from the direction of the vernal equinox to the ascending node.

The choice of orbit depends upon the mission of the satellite. Low Earth orbits (LEO) serve a variety of purposes and do not necessarily operate close to the plane of the equator. In fact, orbits with higher inclinations (near polar) provide the satellite the opportunity to cover a larger portion of the Earth's surface (see. Figure 4-10).

Communications satellites are generally placed in geosynchronous Earth orbits (GEO) where they complete one revolution of the Earth in 23 hours, 56 minutes and 4 seconds. A satellite in a geosynchronous orbit on the equatorial plane will appear stationary to observers standing on the

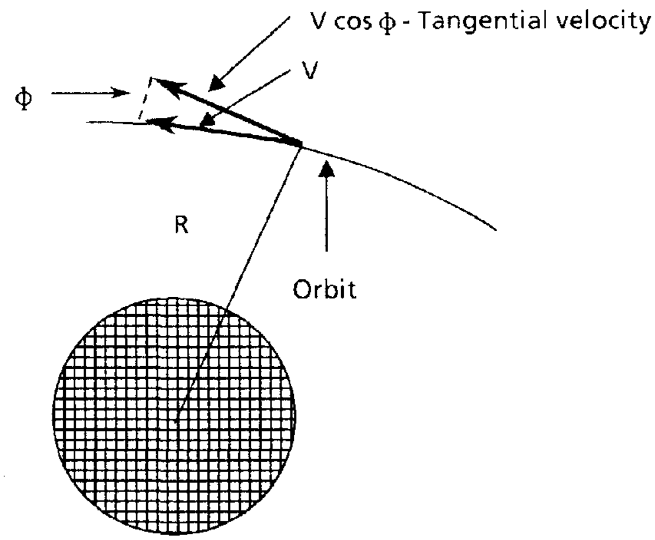


FIGURE 4-8. DEFINITION OF TANGENTIAL VELOCITY

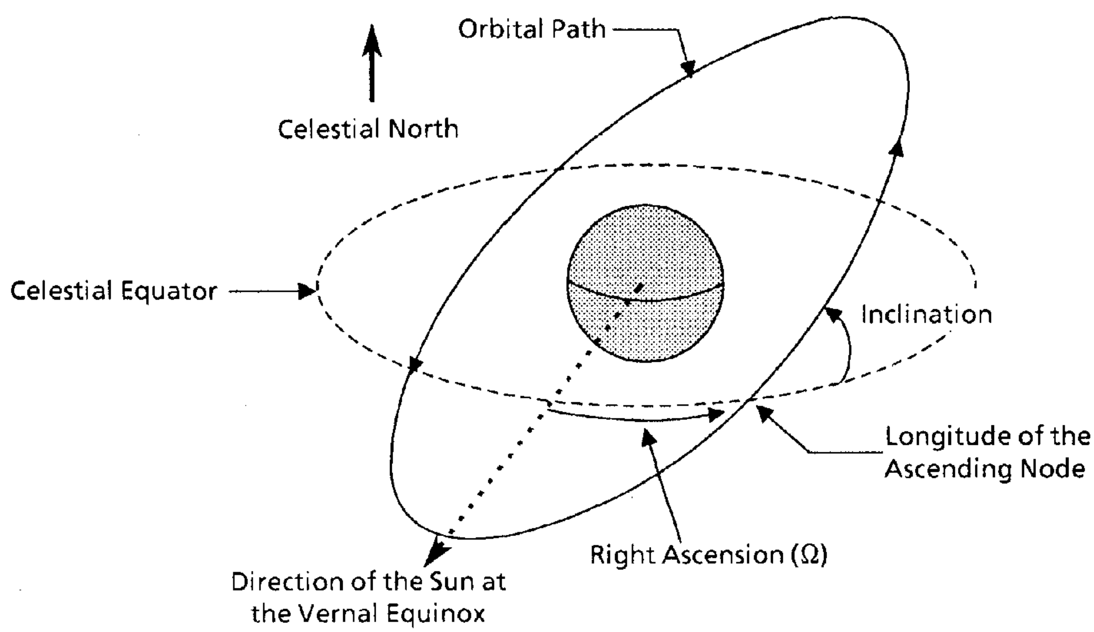
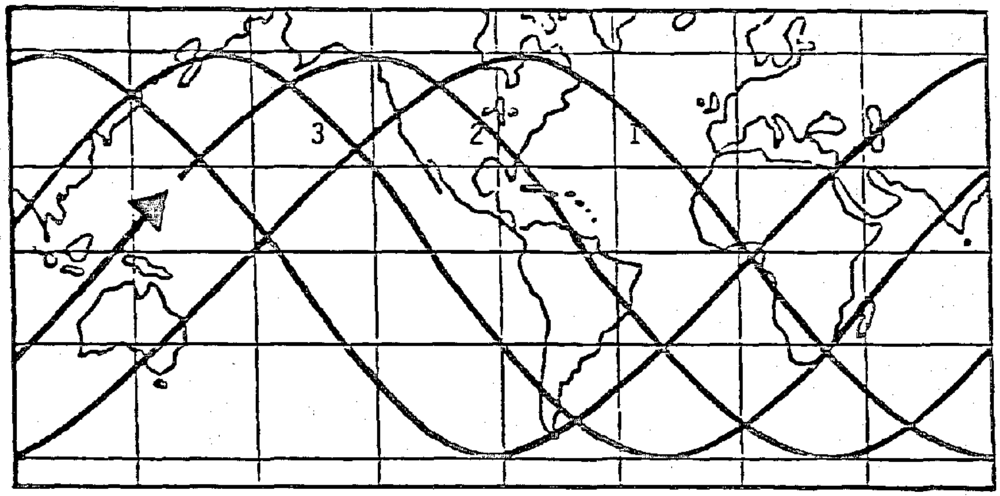


FIGURE 4-9. ORBITAL CHARACTERISTIC DIAGRAM

equator. In order to have this day-long orbital period,* a satellite must be at an altitude of roughly 19,300 nautical miles above the surface of the Earth (5.6 Earth radii). The plane of the orbits of these satellites is either the same as the plane of the equator or at some relatively small inclination angle to the equator. Ideally, equatorial orbits can be achieved directly, with no mid-course corrections,

*Our "solar day" of 24 hours corresponds to the Earth's apparent spin period, but the Earth actually rotates approximately 1 and 1/365 turns in that time. One rotation of the Earth takes 23 hours, 56 minutes and 4 seconds. Time on a scale based on exactly one rotation of the Earth is referred to as *sidereal time*. One 24 hour day of *sidereal time* is equivalent to 23 hours, 56 minutes and 4 seconds of *solar time*.



*Numbers on ground tracks indicate sequence of the orbit, 1 being before 2, etc.

FIGURE 4-10. WESTWARD REGRESSION OF THE ORBITAL GROUND TRACE

only by launches from the equator. Launches from points north and south of the equator have a minimum inclination which is related to the launch site latitude. Thus, equatorial orbits are normally achieved by maneuvers whereby the satellite is reoriented and a rocket motor is fired perpendicular to the plane of the current orbit to create a new orbit orientation (see Figure 4-11).

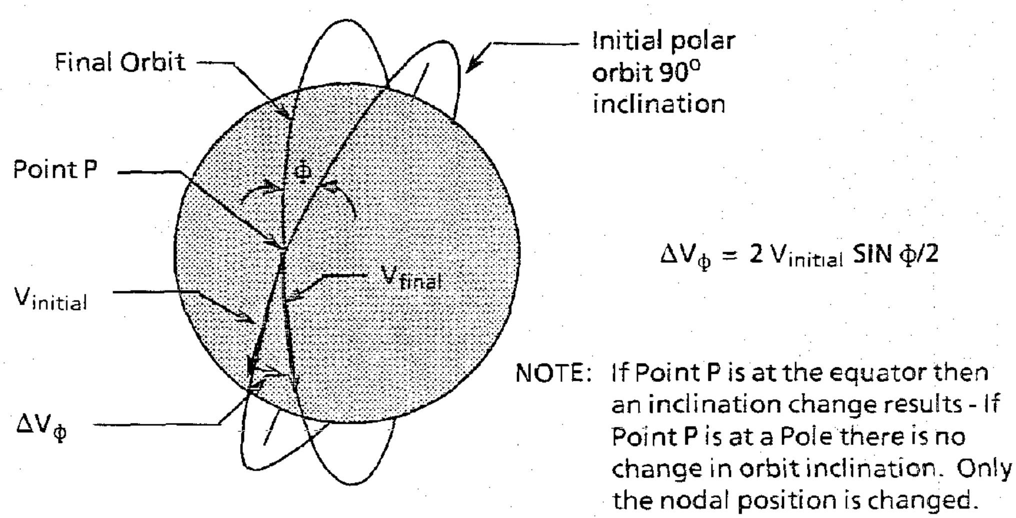


FIGURE 4-11. ORBITAL PLANE CHANGE

Since the orientation of the orbit is relatively motionless in space while the Earth turns inside it, the ground track of the orbit will recess (fall behind). The rate of recession is based on the number of degrees the Earth rotates while the satellite is completing one orbit. The northern-most and southern-most range of the ground track are equal to the inclination of the orbit. A typical ground track is shown in Figure 4-10. The width of the ground track, as seen by the satellite from orbit, is also called a "swath" or "footprint" of the satellite.

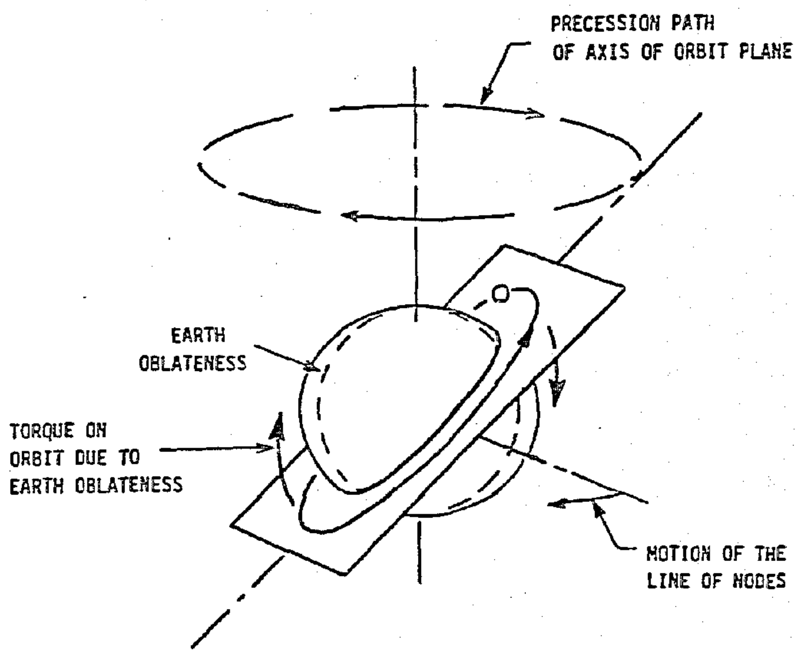
There are external forces which perturb the otherwise stationary orbital plane and cause it to change orientation. The largest effects are caused by the oblateness of the Earth and the gravitational pull of both the Sun and the Moon, called luni-solar perturbations. Their relative importance varies with the altitude of the orbit. The relative effects in terms of acceleration (Earth gravitational units, or g's) for a satellite 200 n mi. above the Earth are shown in Table 4-2. As the altitude of the orbit increases, the relative effect of the Earth's oblateness decreases and the Sun and the Moon's influence increases.

TABLE 4-2. COMPARISON OF RELATIVE ACCELERATION (IN G's) FOR AN EARTH SATELLITE AT 200 NM ALTITUDE (2)

Source of Perturbation	Equivalent Acceleration (in g's) on 200 n mi. Earth satellite
Earth's attraction	0.89
Earth's oblateness	0.001 (approx.)
Sun's attraction	0.0006
Moon's attraction	0.0000033

While the attraction of bodies other than the Earth can distort the orbit, the oblateness of the Earth will cause the plane of the orbit to *precess* around an axis through the pole of the Earth. The additional girth of the Earth around the equator (oblateness) produces a torque on the orbit and the result is a precessional motion not unlike that of a gyro or top. The precession rate can be defined as the number of degrees the line of nodes moves in one solar day. The nodal precession rate for circular orbits is shown in Figure 4-12. Note that the effect of the Earth's oblateness lessens with the altitude of the orbit and also with the inclination of the orbit. A polar orbit will not precess.

The rotation of the Earth has an influence on the ability to launch satellites into desired final orbits. Looking down upon the North Pole, the Earth rotates counterclockwise. At the time of launch, the rocket already has a horizontal component of velocity which equals in magnitude the product of the Earth's rate of rotation and the distance to the axis through the poles of the Earth. If the ELV is launched in the direction of this velocity vector (eastward), it reaches orbital velocity easier than if it is launched in a westerly direction, in which case this surface velocity must be first overcome. (This effect varies with the latitude of the launch point. It is greatest at the equator and absent at the North or South Poles.) This factor is one influence on selection of a site for conducting launches. Therefore, in the United States, eastward launches of satellites into equatorial orbits from ETR,



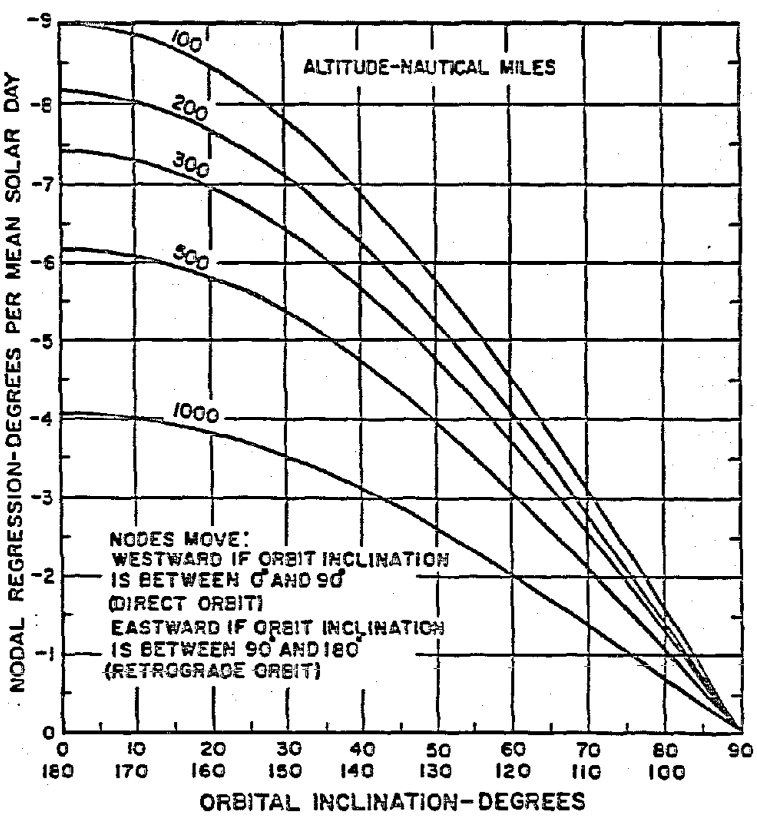


FIGURE 4-12. ORBITAL PLANE PRECESSION DUE TO EARTH'S OBLATENESS (Ref. 2)

Florida augment the ELV thrust. More payload can be placed into orbit than from an identical launch made from, for example, Maine. The satellite launches from the West coast are almost always to the south to achieve polar (high inclination angle) orbits. Polar orbits are perpendicular to the velocity provided by the Earth's rotation, thus the rotation neither helps nor hinders the polar launch. However, the launch corridors used at both ETR and the West coast are chosen primarily for safety considerations. Launches eastward from ETR and southward from the West coast fly over water rather than inhabited territory and do not pose hazards to populated areas due to jettisoned stages or other debris.

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Volume II: Hazards

5. PRE-LAUNCH AND LAUNCH HAZARDS

5.1 INTRODUCTION

5.1.1 Background and Objectives

A hazard is the existence of any property or condition which, when activated, can cause injury, death, or result in damage to property. Of interest to this study are launch-related hazards which could affect third parties, namely people or property not connected with ELV operations. Thus, hazards which have effects contained within the boundaries of the Range are not discussed explicitly in this context.

A hazard potential exists because large quantities of liquid and/or solid propellants are part of the ELV and they could be unintentionally released in case of a launch accident. This hazard decreases with time into the flight because the quantities of on-board propellants decrease as they are consumed and the vehicle moves away from both the launch site and nearby populated areas. The exposure to launch accident hazards is greatest during the first few minutes after launch.

The major generic hazards in the event of an accident involving propellants during pre-launch and launch operations are:

1. <u>Explosions</u>: uncontrolled combustion of these propellants at a very fast rate per unit volume such that part of the chemical energy is converted to mechanical energy and part to heat. The mechanical energy is produced in the form of a blast wave with the potential of causing damage by crushing forces and winds (Sec. 5.2).
2. <u>Debris</u>: vehicle fragments that may land upon structures or populated areas. Fragments may include burning propellants which could explode or burn upon landing thus posing additional hazards of types 1 and 3 (Sec. 5.3).
3. <u>Fires</u>: uncontrolled combustion of the propellants at a slower rate than occurs in explosions, thus converting their chemical energy into heat only. The corresponding hazard is thermal radiation to people and property in the proximity of the fire (Sec. 5.4).
4. <u>Toxic Vapor Clouds</u>: some hypergolic propellants (such as monomethylhydrazine, nitrogen tetroxide and Aerozine-50) are toxic and corrosive. If released in an accident, unreacted vapors and aerosols may be transported by prevailing winds in the form of clouds. Hydrazine vapors are colorless and become white when combined with atmospheric moisture; nitrogen tetroxide vapors are reddish brown. Such clouds may pose a health hazard to people and are potentially harmful to animals and vegetation (Sec.5.5). Othe toxic propellants include fuming nitric acids, liquid fluorine, anhydrous

ammonia, nitromethane, ethylene oxide, chlorine trifluoride, chlorine, nitrogen trifluoride, hydrogen peroxide, hydrogen chloride and hydrogen cyanide.

Hazards associated with noise, sonic boom and small quantity releases of toxic materials are not considered in the same severity category as the hazards listed above and are not addressed in this report.

In a given accident, one or more of these hazards may occur and prevail in importance over the others, depending on the specific circumstances of the event such as: vehicle design, accident location, failure mode, propellant type, amount of propellant released, mode of release, environmental conditions and proximity of people and property. Sometimes, the occurrence of one hazard may preclude another because they compete for the same propellant. For example, when most of the propellant is consumed in a fire, a vapor cloud will not form. Other times, the hazards may be sequential -- such as the formation of toxic vapors in a fire or an explosion which may later pose a toxic vapor cloud hazard. The possible off-range impacts of launch accidents are illustrated in Sec. 5.6.

This chapter presents a generic discussion of the major types of hazards associated with the ground preparation and launch of ELV's namely: explosions, debris, fires and vapor clouds. The objective is to provide an overview of the mechanisms involved in these hazards, the types of analyses used and the damage criteria. The hazards are considered to be of very low likelihood. Their applicability to, and magnitude in, any launch operation should be established by detailed analyses of the specific circumstances in each case. Such analyses for typical launch operations are discussed in Ch. 10, Vol. 3. A second objective is to provide a perspective on launch hazards by comparison with industrial and transportation accidents.

5.1.2 Major Information Resources on Rocket Propellant Hazards

In order to assess public risk exposure derived from launch hazards, information must be drawn from reports of major experimental and theoretical studies of the behavior of accidentally released propellants and fuels.[1,3] These studies include test programs carried out by government agencies (NASA and DOD) where realistic accident scenarios were simulated on a large scale. Two notable test programs were projects PYRO[2] and SOPHY.[3] Both are summarized briefly below to illustrate the experimental basis for the information that follows in this chapter:

1. Project PYRO tested the explosive yield and flammability of liquid propellants namely:

- hypergolics (Aerozine-50 & Nitrogen Tetroxide used as fuel and oxidizer in both the Titan and Delta vehicles) in mass ratio of 2.25/1, in several configurations and with total weights of up to 200 to 1000 lb (90 to 450 kg);
- Liquid Oxygen/RP-1 (used in the Atlas vehicle) in mass ratio of 2.25/1 and with a total weight of up to 25,000 lb (11,000 kg);
- Liquid Oxygen - Liquid Hydrogen (used in the Centaur vehicle) in mass ratio of 5/1 and in total weights of up to 100,000 lb (45,000 kg);
- Full-scale Saturn S-IV and a modified Titan I first stage.

Also, three accident conditions were simulated to produce different types of mixing effects:

- failure of an interior bulkhead separating fuel and oxidizer;
- fall back of a space vehicle on the launch pad with complete tank rupture and subsequent ignition;
- high velocity impact of a space vehicle after launch.

2. Project SOPHY addressed the hazards associated with handling, transporting, testing and launching of solid propellants. Solid propellants were tested in various geometries, sizes and weights (the latter varied from a few hundred to half a million pounds). Shock initiation was produced with a TNT charge centered on the end face of the propellant. Air blast and fire ball data were collected and analyzed statistically to develop scaling relationships. The critical charge diameter required to sustain a detonation in a typical composite propellant was determined to be between 60 and 72 inches.

These two test programs and their results were discussed extensively in a Chemical Propulsion Information Agency (CPIA) publication entitled "Hazards of Chemical Rockets and Propellants".[1] The results were analyzed to identify and quantify the resulting hazards and to develop methodologies for use in hazard analysis. Their findings are drawn upon extensively without having reviewed in detail the original reports of project PYRO and SOPHY.[2,3] Other references of interest to such analyses are safety standards AFR 127-100[4] and DOD 6055.9-STD.[5]

Against this background, we will present a generic discussion of the explosion, debris, fire and vapor cloud hazards associated with the accidental release of propellants. Hazard analyses of specific launch operations will also be discussed in Vol. 3, Chapters 9 and 10.

5.2 EXPLOSION HAZARDS

Explosion of an ELV can occur accidentally, as with the Titan 34D event in April, 1986, or as a result of a destruct command using the flight termination system. In some cases, flight termination is accomplished simply by shutting off the fuel supply to liquid fuel engines. In this case, an explosion may not occur unless the intact vehicle and its remaining fuel impact the ground sharply.

An explosion is a very rapid expansion of matter into a volume greater than its original volume. The cause of the expansion might be combustion, electrical discharge (such as lightning) or a purely mechanical process such as the bursting of a cylinder of compressed gas. The faster the energy is released, the more violent the explosion.

Rocket motors are designed to burn their fuels and release their energy in a controlled combustion process called a deflagration, or simply, a flame. In a deflagration the reaction front is driven by diffusion mechanisms. At steady state, it proceeds in the material at a rate lower than the speed of sound.

Under some conditions, the rate of energy release can increase significantly, leading to an explosion. The combustion process is then called a detonation. In a detonation the reaction front consists of a shock wave followed by a flame. The reaction front is driven by a shock compression mechanism. At steady state, it proceeds in the material at a rate faster than the speed of sound.

There is a spectrum of reaction possibilities between steady state deflagrations and detonations, such as a fast deflagration and a weak detonation, with the potential of a transition from one reaction to another. The deflagration-to-detonation transition is referred to as DDT. A shock-to-detonation transition is also possible and is referred to as SDT.(6,7)

For solid propellants (see Table 3-3, Vol. 1, Ch. 3), cross-linked double base hybrid materials (DOD Class/Division 1.1 -- old Class 7) were always considered in the past to represent a detonation hazard; most composite propellants (Class/Division 1.3 -- old Class 2) were considered to represent a fire (deflagration) hazard. However, recent trends in rocket motor design include: more energetic composite propellants, higher solid loading densities, larger grain diameters and greater mass. The net effect is that composite propellants may also detonate inadvertently under the dynamic conditions of accidents. Although, they may require a larger initiation energy than Class/Division 1.1 propellants and their detonation may not be self-sustaining, resulting in lower yields(7).

A number of conditions influence the likelihood of solid propellant detonation:(6,7)

- propellant toughness;
- motor geometry, core configuration, diameter, length to diameter ratio, chamber pressure, case bonding technique and propellant residual strain;
- propellant critical diameter and geometry;
- propellant granular bed characteristics (pyrolysis and ignition) both thermally and mechanically induced, leading to faster combustion terminating in a detonation (DDT);
- propellant response to shock (SDT);

- propellant response to delayed reduced shock (referred to as XDT)
- impact velocity and surface impacted (water, sand or concrete).

A question of particular interest is whether activation of the destruct system is likely to detonate solid rocket boosters. This subject was studied recently by the Naval Surface Weapons Center (NSWC) for a filament wound graphite case material.(8) They tested:

- linear shaped charge (LSC)/propellant case interactions;
- detonability and shock sensitivity;
- material response (breakage of propellant).

They concluded that activation of LSC would not detonate the Solid Rocket Booster propellant. At most, a rapid burn is expected.

For liquid propellants, the likelihood of detonation is influenced by chemical composition and conditions such as:

- degree of fuel and oxidizer mixing and size of the mixture prior to initiation;
- confinement of the products of combustion;
- presence of obstructions or flow instability that generate turbulence and result in increased reaction areas.

Such conditions are encountered in accidents to various degrees. Thus, it is usually very difficult to predict with certainty whether or not a detonation will occur.

Still, overpressure can result if the reaction is fast enough, even though it is not an ideal, steady state detonation. The main difference is in the near-field where a detonation generates a much higher overpressure. This difference decreases further away from the center of the explosion. The far-field is of particular importance to this study which focuses on potential damage to the public (third parties) off-range. Overpressure estimation methods are presented in the next section.

5.2.1 Blast Waves

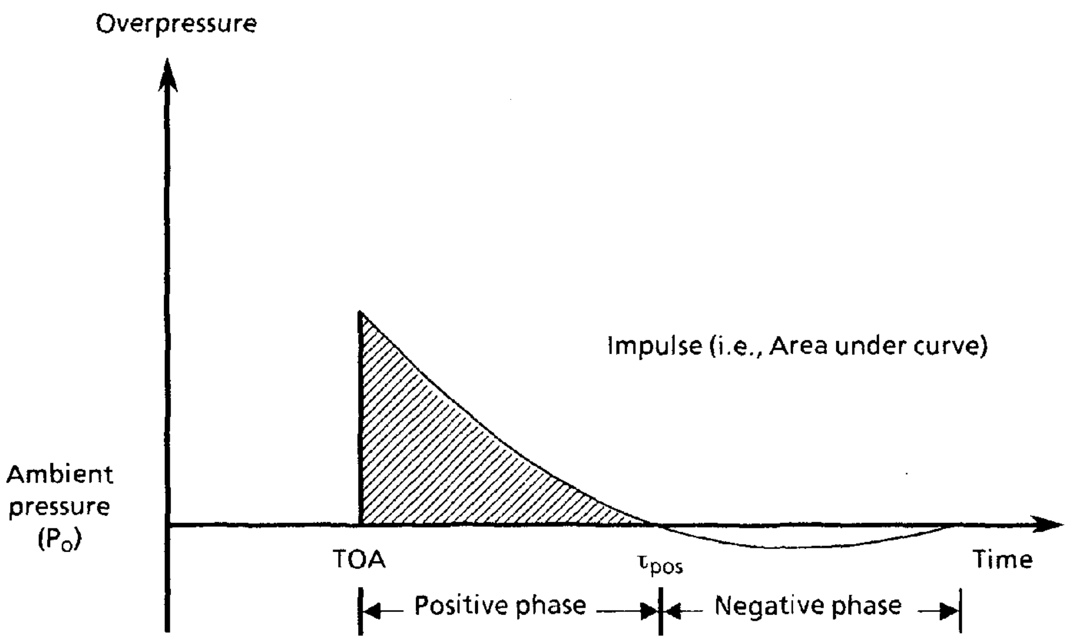
Scaling laws are used to calculate characteristic properties of blast waves from explosions. With the aid of such laws, it is possible to present characteristics of the blast wave, for any yield, in a simple form. This is presented below for the case of air at constant temperature and pressure.

Full-scale tests have shown that these relationships hold over a wide range of explosive weights (up to and including megatons). According to the scaling laws, if d_1 is the distance from a reference explosion of W_1 lb at which a specified hydrostatic overpressure or dynamic pressure is found, (Dynamic pressure $q = 1/2\, \rho v^2$, where ρ is air density and v is particle velocity), then for any explosion of W lb, these same pressures will occur at a distance, d, given by:

$$d/d_1 = (W/W_1)^{1/3} \qquad (5\text{-}1)$$

In other words, the pressures are functions of a unique variable ($d/W^{1/3}$) called the scaled-distance or k-factor.

Cube-root scaling can also be applied to the arrival time of the shock front, positive-phase duration and impulse; the distances concerned are also scaled according to the cube-root law (see Figure 5-1 for a definition of these terms). The relationships may be expressed in the form: $t/t_1 = i/i_1 = d/d_1 = (W/W_1)^{1/3}$, where t represents arrival time or positive-phase duration, i is the impulse and the subscript 1 denotes the reference explosion W_1.



(1) TOA (time-of-arrival) = The time required for the shock wave to transit the distance from the center of the explosion to the point at which the measurement is to be made.

(2) P (overpressure) = Peak pressure above ambient conditions.

(3) τ = Positive phase duration - the length of time (measured from the first pressure rise) necessary for the overpressure to return to the ambient pressure.

(4) Positive phase impulse = $\int_0^{\tau} P(t)\,dt$

FIGURE 5-1. DEFINITION OF SHOCK WAVE PARAMETERS (Ref. 1)

These relationships are well established and accepted in the literature. They form the basis of most explosion models, including that used in Chapter 10 of this report.

It should be noted that the above relationships are for blast waves in free field, under ideal conditions. In a real, stratified atmosphere, shock focusing may occur producing higher overpressures than in free field. Such effects have been taken into account in a computer model named BLAST based on acoustic wave propagation. The model was developed by WSMC and has been verified experimentally.[9]

5.2.2 TNT Equivalency Analysis

It is conventional to express the magnitude of an explosion of a given material (e.g., solid or liquid propellant) in terms of an equivalent weight of TNT (symmetrical tri-nitrotoluene, a conventional ordnance explosive) required to produce essentially the same blast wave parameters. The TNT equivalent weight was selected because of the large amount of experimental data available on blast waves and damage produced by TNT explosions. A given material may have several TNT equivalent weights depending on the selected blast wave parameter, i.e., it may have an equivalent weight based on peak overpressure, another based on positive impulse, (see Glossary, App. A, or Figure 5-1), etc. Peak overpressure is more commonly used, however, to define TNT equivalence. TNT yield refers to the TNT equivalent weight expressed as a percent of the weight of the propellant.

The TNT-equivalent analysis has a number of limitations that should be borne in mind to obtain valid comparisons. They are:

- Not all the accidentally-released material is involved in the explosion: part of it may disperse without reacting and part may react at a different time or location from the explosion. Accordingly, measured TNT yields of liquid propellants were found to depend on the degree of fuel/oxidizer mixing prior to explosion initiation. This degree of mixing depends, in turn, on the rate of mixing (a function of vehicle design, failure mode and accident conditions) and its duration (a function of when ignition occurs).
- Of the portion of released material that reacts in the explosion, part of it may detonate and part may deflagrate, with the latter contributing little energy to the blast. Predicting whether a detonation or deflagration (or any combination of them) will occur is a very complex subject, as discussed earlier. The outcome depends on the propellant properties and on the conditions of the accident. For example, with solid propellant fragments, an impact speed greater than 300 ft/sec is likely to have sufficient energy to initiate the detonation of that fragment upon impact.[7]
- Even for the portion of the released material that contributes directly to the blast energy, the blast characteristics are different from those of a TNT charge with an equivalent energy. Measured overpressure amplitudes are generally lower and

durations are longer because of a slower reaction rate for propellants than for TNT. This rate depends on accident-specific conditions such as: strength of initiating source, degree of confinement and shape of propellant.

Thus, the TNT yield of a material is not an absolute property such as density or molecular weight. Instead, it depends on the test conditions in which it is measured. Fortunately, the dependence of blast parameters on yield is low because of the cube-root exponent in the scaling law (Eq. 5-1). Hence, the prediction of a hazard distance (d) is not very sensitive to the employed yield (W). For example, if the yield is off by 50 percent, the distance (at which a particular overpressure is reached) is off by only 15 percent. Thus, the TNT method of analysis has been used effectively over many years despite the limitations mentioned above.

In 1978, NASA established an Explosive Equivalency Working Group to define potential failure scenarios which could lead to an explosion and to estimate the maximum credible explosive TNT equivalency for these explosions. The most complete documentation of the findings of this group is reportedly in a collection of briefing charts by W.A. Riehl et al.[10] The work performed by this group provided a basis for many subsequent studies,[11] many of which have quoted verbatim TNT equivalent values from Ref. 10. This is illustrated in Table 5-1, which is extracted from a study on shuttle safety.[11] A variety of failure modes and accident scenarios are identified for the external tank and the solid rocket motors; a maximum credible explosive equivalent (or TNT yield) is estimated for each case. Also, the range for these maximum credible TNT yields varies from:

- 5 to 50% for LH_2/LOX
- 18 to 100 % for the solid rocket motors

The lower bound for these yields is zero, since the propellants may react or burn without producing mechanical damage.

Although the STS is not being considered for commercial space transportation, Table 5-1 is very useful to illustrate that the yield of a propellant system can vary depending on the failure mode.

Recommended values for TNT equivalency of liquid propellants under selected worst case accident conditions are given in AFR 127-100.[4] Since AFR 127-100 addresses the circumstances in handling and storing propellants, it may not apply to launch operations. The values are presented in Table 5-2, where it should be noted that:

- TNT yields for the same propellant vary depending on the accident conditions. While this variation is consistent with the concept of TNT yield (as discussed above), it is important to select the appropriate value for each set of accident conditions since the yield varies by up to a factor of seven.

TABLE 5-1 ESTIMATED SHUTTLE MAXIMUM CREDIBLE EXPLOSIVES EQUIVALENCIES

External Tank

Failure Mode			%TNT Yield (by weight)
Destruct (Range Safety System)			
Current Designe	- Without Orbiter		0.5
	- With Orbiter		1.0
Redesign (Galileo Mission)			0.25
Direct Fall Back on Pad			Not Credible
High Velocity Ground Impact (Intact) (W/O Destruct)			50*
Over Pressurization	- LH_2 Tank		0.5
	- LOX Tank	- Flight	Not Credible
		- Ground	10
SSME (Boattail) Explosion			0.5
Fallover	- Both SRB's Fail to Ignite		38
Tipover	- One SRB Fails to Ignite		38
TPS Failure	- ET - LH2 Tank	- Barrel	1
		- Aft Dome	0.5
	- LOX Tank		0.5
	- Intertank		10
	SRB - Nose Cone/ Aft Skirt		0
	- Cable Tray (Destruct)		5-1
	- Sep'n Motors, Thermal Curtain, Attach Ring		0.5
SRB - TVC Hardover (Corkscrew - Destruct)			5-1
SRB - Case Rupture	- Adjacent to ET		0.5
	- Elsewhere	- Cartwheel	5
		- Separation	0.5
Recontact on Separation - SRB/ET		- Aft	0.5
		- Forward (I/T)	10
	- Orbiter/ET		0.5

Solid Rocket Motors

Failure Mode	%TNT Yield (by weight)
Aft Segment at Impact	18
High Velocity Ground Impact (W/O RSS)	20
Fallover - (Both SRB's Fail to Ignite)	20-50
Tipover - (One SRB Fails to Ignite)	20-100

Source: Briefing, Riehl, 1979 [Ref. 10]

*The yield is a function of impact velocity and can reach 150% for velocities in excess of about 500 feet per second.

- Significant equivalent TNT yields are estimated under the most severe scenarios. These worst case scenarios are very unlikely, however.

For illustration, the recommended TNT yield values are applied to three classes of ELV vehicles of interest: Atlas/Centaur, Delta and Titan. This is presented in Table 5-3, which shows the propellant composition, weight, TNT yield estimate and TNT equivalent weight for each vehicle. Note that :

TABLE 5-2. LIQUID PROPELLANT HIGH EXPLOSIVE (TNT) EQUIVALENT YIELDS (Source Ref. 4, AFR 127-100, Table 5-14)

Propellant Combination	Static Test Stands[3]	Range Launch Pads[3]
LO_2 - LH_2	60%	60%
LO_2 - LHx RP-1	Sum of 60% for LO_2 - LH_2 plus 10% for LO_2 - RP-1	Sum of 60% for LO_2 - LH_2 plus 20% for LO_2 - RP-1
LO_2 - RP-1 or LO_2 - NH_3	10%	20% up to 500,000 lbs plus 10% over 500,000 lbs
Inhibited Red Fuming Nitric Acid (IRFNA) - Aniline[2]	10%	10%
IRFNA-UDMH[2]	10%	10%
IRFNA-UDMH + JP-4[2]	10%	10%
N_2O_4-UDMH + N_2H_4[2]	5%	10%
N_2O_4-UDMH + N_2H_4-Solid[2]	5% plus the high explosives equivalent of the solid propellant	10% plus the high explosive equivalent of the solid propellant
Pentaborane + a fuel	10%	20% up to 500,000 lbs plus 10% over 500,000 lbs
Pentaborane + an oxidizer	60%	60%
Tetranitromethane (alone or in combination)	100%	100%
Nitromethane (alone or in combination)	100%	100%
Substitutions	Percentages given above continue to apply where any of the substitutions shown below are made in the basic combination.[4]	

NOTES:

1. Basis of the table: Developed by the Department of Defense Explosives Safety Board Work Group on Explosive Equivalents for Liquid Propellants. Tetranitromethane and nitromethane are known to be detonable. The net weight of all nonnuclear mass-detonating explosives involved in any configuration, including component of nuclear items, will be added to the above equivalencies, where applicable, in determining required separations. See paragraph 5-26a(5) in Ref. 4 concerning equivalents for hypergolic combinations.

2. These are hypergolic combinations. (Fuel and oxidizers that will ignite with each other.)

3. The percentage factors used for the explosive equivalencies of propellant mixtures at launch pads and static test stands were based on such propellants located above ground and unconfined except for their tankage. Other configurations will be considered on an individual basis to determine applicable equivalencies.

4. Substitutions, alcohols or other hydrocarbons substitute for RP-1; H_2O_2, F, BrF_5, CLF_3, OF_2, or O_3F_3 substituted for LO_2, Monomethylhydrazine substituted for hydrazine or UDMH, or ammonia substituted for any fuel where hypergolic combination results.

- for liquid propellants, the yield estimates are based on the recommended guidelines in AFR 127-100 which represent worst cases. Thus, they are inherently conservative.

- For solid propellants, the yield estimates are taken from a compilation of SRM impact detonation history.[1] A range of values (varying over a factor of five) is given to cover a number of accident scenarios.

TABLE 5-3. ESTIMATED UPPER BOUNDS ON TNT-EQUIVALENT WEIGHTS OF ELV PROPELLANTS

System	Propellants Composition	Propellants Weight, klb	TNT Yield %	Basis for TNT Yields	TNT Equivalent Weight, klb
Atlas	RP-1/LO_x	303	10 - 20	a	30 - 60
Centaur	LH_2/LO_x	30.7	60	a	18
					48 - 78
Delta					
Booster	Solid (Castor IV)	186	14 - 100 +	b	26 - 186 +
Stage 1	RP-1/LO_x	179	10 - 20	a	18 - 36
Stage 2	Aerozine 50/N_2O_4	13	5 - 10	a	0.7 - 1.3
Stage 3	Solid	2.3	14 - 100 +	b	0.3 - 2.3 +
					45 - 226 +
Titan III					
Stage 0	Solid (UTP-3001 B)	464	14 - 100 +	b	65 - 464 +
Stage 1	Aerozine 50/N_2O_4	294	5 - 10	a	15 - 29
Stage 2	Aerozine 50/N_2O_4	69	5 - 10	a	3.5 - 7
Transtage	Aerozine 50/N_2O_4	9	5 - 10	a	0.5 - 1
					84 - 501 +

Notes on Basis for TNT yield:

(a) Recommended values for liquid propellants in AFR 127-100 (Table 5-14 on pg. 72). It is recognized that these recommended values are based on worst case scenarios and are thus conservative.

(b) Based on data in CIPA Handbook (Ref. 1) for SRM Impact Detonation History (Table 2-1 on pg. 2-6).

Note that the range of TNT yields vary from a lower bound of zero (i.e., no blast) to the upper values given above.

TNT equivalent weights are obtained by multiplying each propellant weight by its yield. A range of TNT weights is obtained because of the uncertainties in the yields. Such uncertainties are expected in view of the previous discussion of the various factors that affect TNT yield. In reality, the ranges vary from a lower bound of 0 (i.e., no blast) to the upper values (i.e., worst cases) in Table 5-3. To estimate a reasonable value within this range requires an accident-specific analysis, which is not attempted in this generic report.

Finally, note that a hybrid propellant mix technology (liquid oxygen/solid polybutadiene fuel) proposed by AMROC, has been assigned a TNT equivalence of zero by the DOD Explosives Safety Board.

5.2.3 Damage Criteria

Blast waves from accidental explosions can cause damage to people and property (structures) by subjecting them to transient crushing pressures and winds (which cause drag pressures due to the sheer force of the wind). Even though the interactions of the waves with the objects involve very complex phenomena, relatively simple concepts have been used quite effectively to correlate blast wave properties with damage to a variety of targets. The concept states that damage is primarily a function of either the peak overpressure, the impulse or some combination of these two factors.

Guidelines for peak overpressures required to produce failures to structures such as shattering of glass windows and collapse of concrete walls are presented in Table 5-4.[1] Note that a very low pressure (force per unit area) is sufficient to cause damage, mainly due to the large area of such surfaces. Similar criteria are used in the hazard assessment model used in Vol. 3, Ch. 10 of this report.

Criteria for injury of personnel standing in the open are given in Table 5-5.[1] They cover ear drum rupture and lung hemorrhage caused by overpressure and personnel blowdown caused by the impulse imparted by the blast wave, with the concomitant potential of injury due to bruises, lacerations and bone fractures. These data are presented in graphic form in Figure 5-2 and Figure 5-3.[12] Note that:

- The overpressure required to cause damage decreases (as expected) with the increase in the duration of the positive phase of the blast wave.
- There is a significant variability in the susceptibility of people to such overpressure. Such variability can be accounted for statistically by raising overpressure thresholds to ensure higher levels of lethality. This should be done carefully to maintain a realistic approach to analysis.

Finally, blast wave characteristics (Section 5.2.1) can be combined with the present damage criteria in order to estimate the extent of the damage (in feet) as a function of various equivalent weights of TNT. Typical results are shown in Figure 5-3 for eardrum rupture, lung damage, etc. Similar data are used in the next section and in Ch. 10, Vol.3 to illustrate the assessment of both property damage and personnel injury over a range of accident conditions.

5.2.4 Variation of Explosion Hazards with Time from Liftoff

As noted, launch hazards decrease with time into the flight. This point is illustrated in this section for potential third party damage due to an accidental explosion of an ELV. The variations of other hazards with time are not discussed.

TABLE 5-4. CONDITIONS OF FAILURE OF PEAK OVERPRESSURE-SENSITIVE ELEMENTS (Ref. 1)

Structural Element	Failure	Approximate Incident Blast Overpressure kPA (psi)
Glass windows, large and small	Shattering usually, occasional frame failure	3.4-5.9 (0.5 - 1)
Corrugated asbestos siding	Shattering	5.9-13.8 (1 - 2)
Corrugated steel or aluminum paneling	Connection failure followed by buckling	6.9-13.8 (1 - 2)
Wood siding panels, standard house construction	Usually failure occurs at main connections allowing a whole panel to be blown in	6.9-13.8 (1 - 2)
Concrete or cinder-block wall 20.3 or 30.5 cm (8 or 12 in) thick (not reinforced)	Shattering of the wall	13.8-20.7 (2 - 3)
Self-framing steel panel building	Collapse	20.7-27.6 (3 - 4)
Oil storage tanks	Rupture	20.7-27.6 (3 - 4)
Wooden utility tanks	Snapping failure	34.5 (3 - 4)
Loaded rail cars	Overturning	48.3 (7)
Brick wall panel, 20.3 or 30.5 cm (8 or 12 in) thick (not reinforced)	Shearing and flexure failures	48.3-55.2 (7 - 8)

Data are used for a typical Delta ELV system flight profile and propellant consumption rate as a function of time elapsed after liftoff.(13) However, qualitatively, the discussion applies equally well to other ELV systems.

The outcome of an accident is usually determined by the specific circumstances present at the time and location of the accident. Usually, there are a number of variations for these circumstances which can lead to a number of outcomes. In this illustration, the analysis is simplified to focus on the effects of "time into flight."

The calculations presented below are also based on a number of assumptions selected to make the analysis workable. For example, for the sake of simplicity, it is assumed that all of the propellants

TABLE 5-5. AIR-BLAST CRITERIA FOR PERSONNEL STANDING IN THE OPEN (Ref. 1)

Criteria	Physical Parameters Dose	Remarks
	Direct Overpressure Effects:	
1% Eardrum Rupture	23 kPa (3.4 psi)	Not duration sensitive except possibly for durations of less than 1 msec. Not a serious lesion.
50% Eardrum Rupture	110 kPa (16.0 psi)	Some of the ear injuries would be of a severe form.
Threshold of Lung Hemorrhage	69 kPa (10.0 psi)	69 kPa (10 psi) applies to blasts of long duration, over 50 msec; 138-207 kPa (20-30 psi) required for 3-msec duration waves; not a serious lesion.
1% Mortality	186 kPa (27 psi)	186 kPa (27 psi) applies to blasts of long duration, over 50 msec; 414-483 kPa (60-70 psi) required for 3-msec duration waves. A high incidence of severe lung injuries.
	Displacement Effects:	
No Personnel Blowdown	8.62 kPa-msec (1.25 psi-msec)	At this dynamic-pressure impulse, man would attain a peak horizontal velocity of 0.09 m/s (0.3 fps)
50% Probability of Personnel Blowdown	57.2 kPa-msec (8.30 psi-msec)	At this dynamic-pressure impulse, man would attain a peak horizontal velocity of 0.61 m/s (2.0 fps).
1% Probability of Serious Injury from being Blown down	372 kPa-msec (54 psi-msec)	At this dynamic-pressure impulse, victim would attain a peak horizontal velocity of 4 m/s (13 fps); serious injury (bone facture or rupture of internal organs) could occur from impact with the ground; high probability of minor injuries such as bruises and lacerations.

remaining on board will explode instantly (this corresponds to a worst case calculable explosion scenario). In reality, the situation is more complicated:

- some of the propellant may explode initially, producing fragments that may explode later upon impact with the ground (secondary explosions);
- some of the propellant may burn in a fireball; and

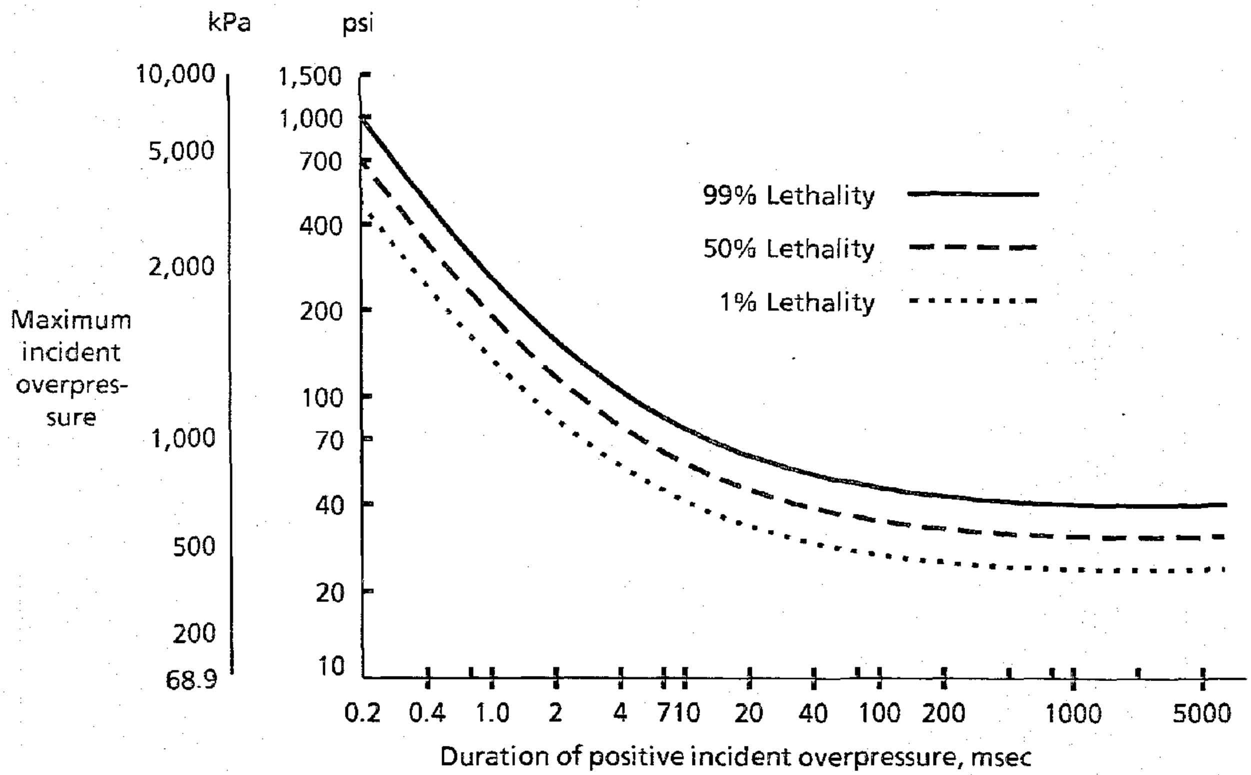


FIGURE 5-2. LETHALITY CURVES PREDICTED FOR 154 LB. PERSON IN FREE-STREAM SITUATIONS (Ref. 12)

- some of the hypergolic propellant may disperse in the environment without reacting, posing toxic risks or dispersing harmlessly.

Another example of a simplifying assumption is to represent different circumstances occurring at various times into flight by simply changing the TNT yield. The yield is increased when the circumstances (such as failure mode, mixing rate or impact speed) favor a stronger explosion (as described in more detail below).

Note that each scenario can be associated with a vehicle failure mode and is likely to occur with a particular probability value (Section 5.6). Thus, although the discussion below makes no explicit mention of probabilities, the predicted results are tied to a particular probability value.

Therefore, three key changes can be identified as time elapses from liftoff: the vehicle altitude (and down-range distance), the quantities of propellants remaining on board and the explosive potential of these propellants. These changes are illustrated in Figure 5-5 and are discussed below.

First, the vehicle altitude increases very rapidly with time into flight -- reaching roughly 20 nmi. in the first 2 minutes, as illustrated by curve A in Figure 5-5, which shows a typical flight profile for a Delta mission.[13] Furthermore, the location of launch sites and the direction of launch are usually selected

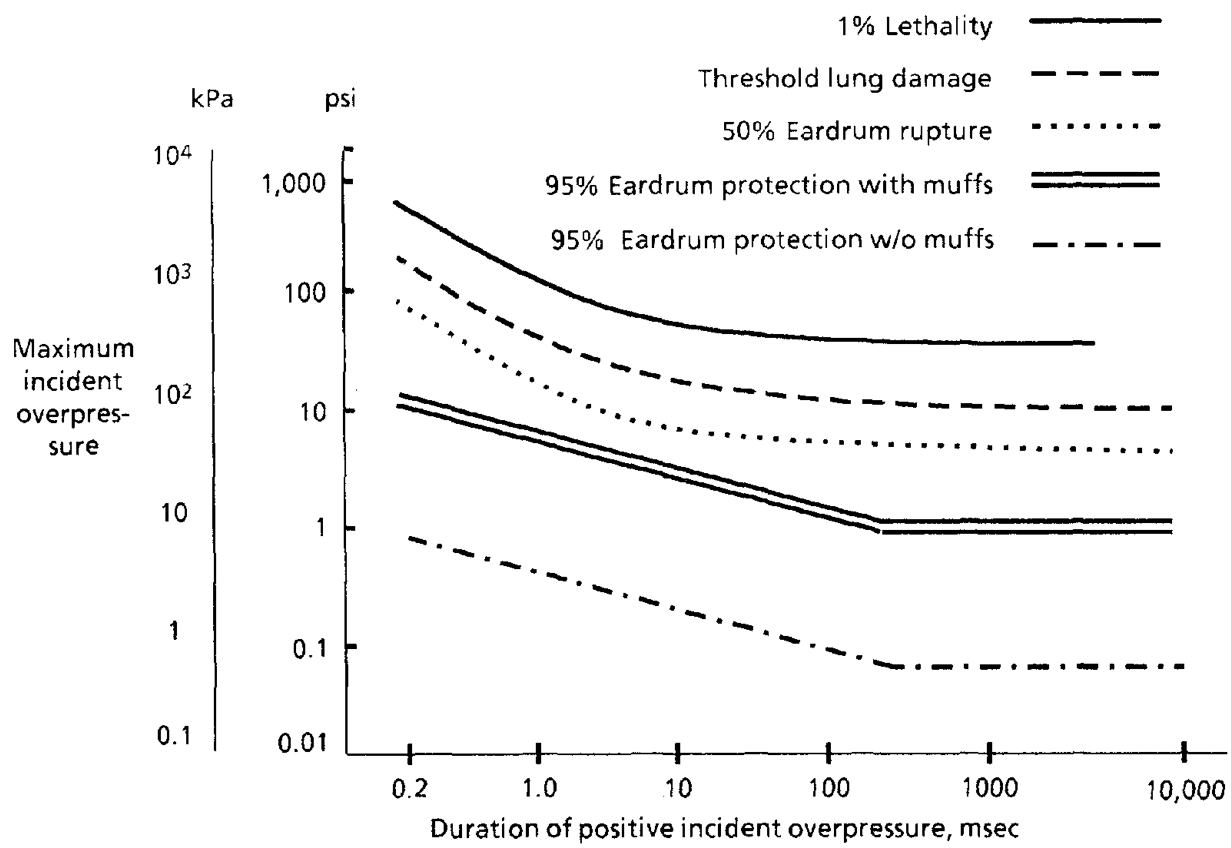


FIGURE 5-3. LETHALITY AND DAMAGE/INJURY CURVES PREDICTED FOR 154 LB. PERSON IN FREE-STREAM SITUATION (Ref. 12)

so the vehicle moves away from population centers. Thus, the "separation" distance between the vehicle and the communities potentially vulnerable, in case of a vehicle accident, increases with time.

Second, as time elapses from liftoff, the quantity of propellants remaining on board decreases very rapidly due to their rapid consumption by the rocket booster and other engines. The total weight of all propellants remaining on board is illustrated by Curve B in Figure 5-5. Note that the total remaining propellant weight decreases by about 50% within 2 minutes from liftoff.

Third, the explosive potential (or TNT yield) of a given quantity of propellant may change as time elapses from liftoff. As discussed earlier (Sec. 5.2.2), the TNT yield of a propellant in an accidental explosion depends on its properties, as well as on a variety of other factors, determined by the details of the accident scenario. Example of such factors include: the sizes of solid propellant fragments their impact speed, the rate and extent of mixing of liquid propellants, the degree of confinement, etc. In fact, the significance of TNT yields, how they are estimated and the pertinent ranges of values given in the published literature were discussed in Section 5.2.2.

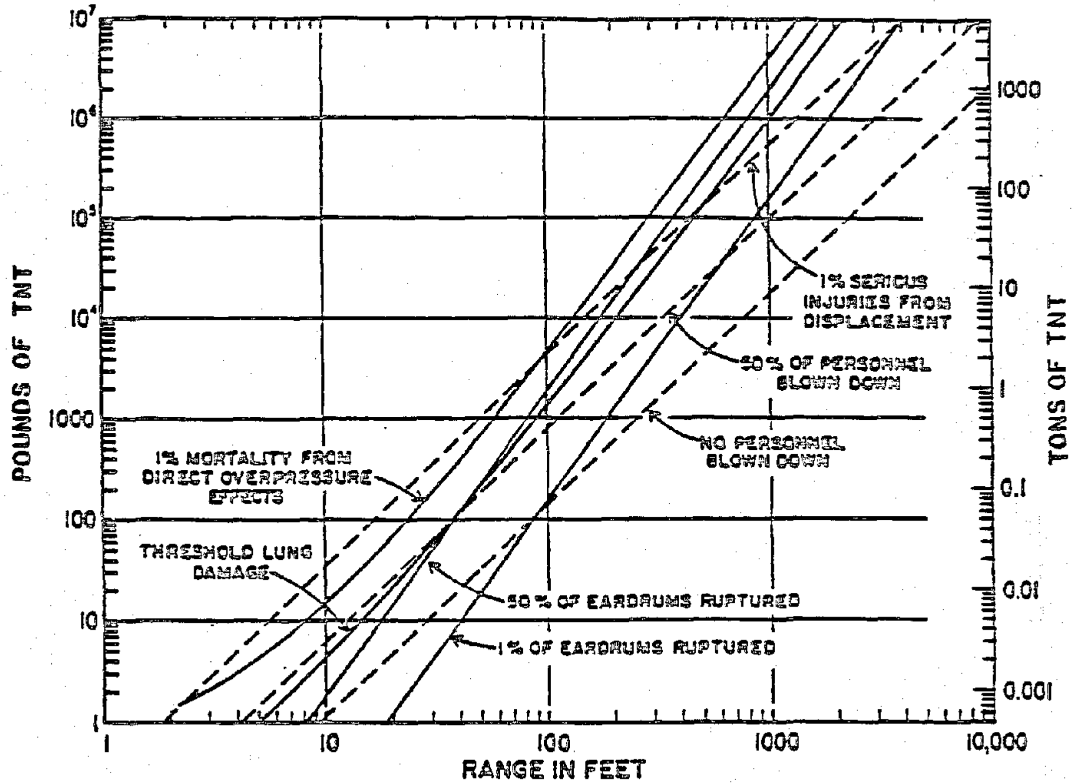


FIGURE 5-4. AIR-BLAST CRITERIA FOR PERSONNEL STANDING IN THE OPEN (Ref. 12)

Determination of TNT yield at various times after liftoff requires an extensive analysis. First, identify the type of failures and accident scenarios that are likely to occur and second, estimate the yield for each scenario and each propellant system based on historical accident data, test data, experience and engineering judgment. Such an analysis was done for the Space Shuttle system by the Explosive Equivalency Working Group established by NASA in 1978, as discussed in Section 5.2.2. Ideally, the same type of analysis for each ELV type is needed to establish pertinent explosive yields were the accident to occur at various times from liftoff. However, for simplicity, another approach which is not as rigorous, but may suffice, is used to illustrate the explosive yield dependence on time from liftoff.

Table 5-2 in Section 5.2.2 lists upper limits for TNT yields for ELV propellants reported in the literature. The lower bound for these yields is zero (%), since the propellants may react or burn without producing mechanical damage. The range of upper values for the Delta vehicle propellants are:

- 10 to 20% for RP-1/LOX (Stage I)
- 5 to 10% for Aerozine-50/N_2O_4 (Stage II)

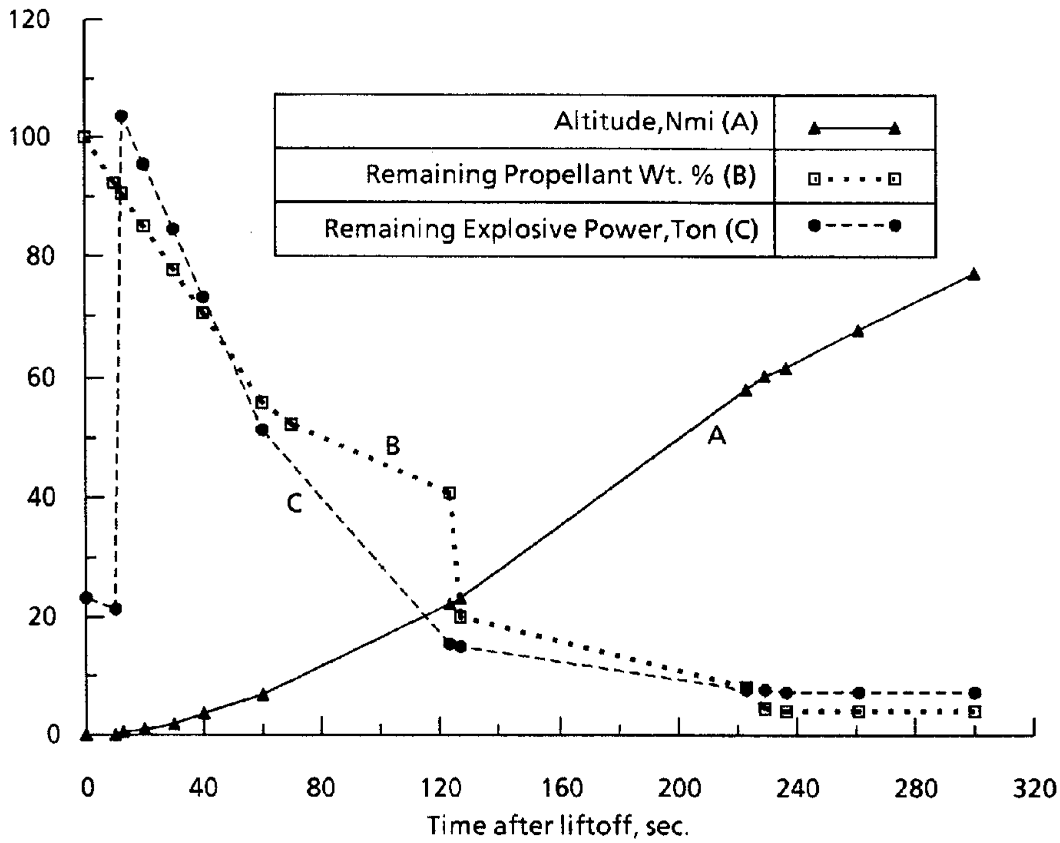


FIGURE 5-5 POTENTIAL EXPLOSION HAZARD AS A FUNCTION OF TIME (DELTA ELV)

- 14 to 100% for the solid rocket motors (Booster and Stage III)

Note that each point within these ranges can be associated with a particular accident scenario which, in turn, may be associated with a specific time from liftoff. For example, when a vehicle (or its fragments) falls back on the pad soon after liftoff, the speed at ground impact is a key factor in determining the likelihood of detonating the solid propellants. It is known that an impact speed of 300 ft/sec is required to detonate solid propellants and produce significant yields. In order to reach such terminal speeds in free-fall, a vehicle would have to start at an altitude of approximately 1400 ft (assuming no drag). This altitude would be reached in about 12 seconds after liftoff. Thus, if the vehicle falls back onto the pad in the first 12 seconds (or so), a low yield is anticipated, while if it falls back at a later time, a higher yield is anticipated. Following this reasoning, the yields corresponding to these two situations are assumed (for simplicity) to be the upper and lower values of the ranges listed above for the three propellant types in the Delta vehicle. Thus, the yields would be:

- 10, 5 and 14% (respectively for the 3 types of propellants) in the first 0 to 12 sec after launch;
- 20, 10 and 100%, respectively, at later times into flight.

By multiplying these yields with the amount of propellants remaining on board, the potential explosive energy (in terms of equivalent pounds of TNT) is estimated as a function of time from liftoff as illustrated by Curve C in Figure 5-5. Note that the explosive potential starts at a low value (because of the low yield); then increases because of the increase in yield corresponding to higher impact speed; finally it decreases because of the decrease in the quantity of propellant remaining on board.

Using the potential explosive energy determined above, the overpressure field around the explosion point was estimated following the analysis outlined in Section 5.2.1. It was assumed that the entire vehicle will explode at altitude and as one mass (a more realistic assumption is a smaller explosion in flight, breaking up the vehicle in fragments that will explode upon ground impact). It was also assumed that any reflection or focusing of the shock wave would have a negligible effect on the overpressure field.

For these assumed explosion conditions, the "hazard" distances at which critical overpressures are reached are shown as a function of time in Figure 5-6. Three overpressure levels are used:

- 1.5 psi, for collapse of light weight structures (Curve B)
- 0.35 psi, for window breakage with a probability of 50% (Curve C)
- 0.20 psi, for window breakage with a probability of 10% (Curve D)

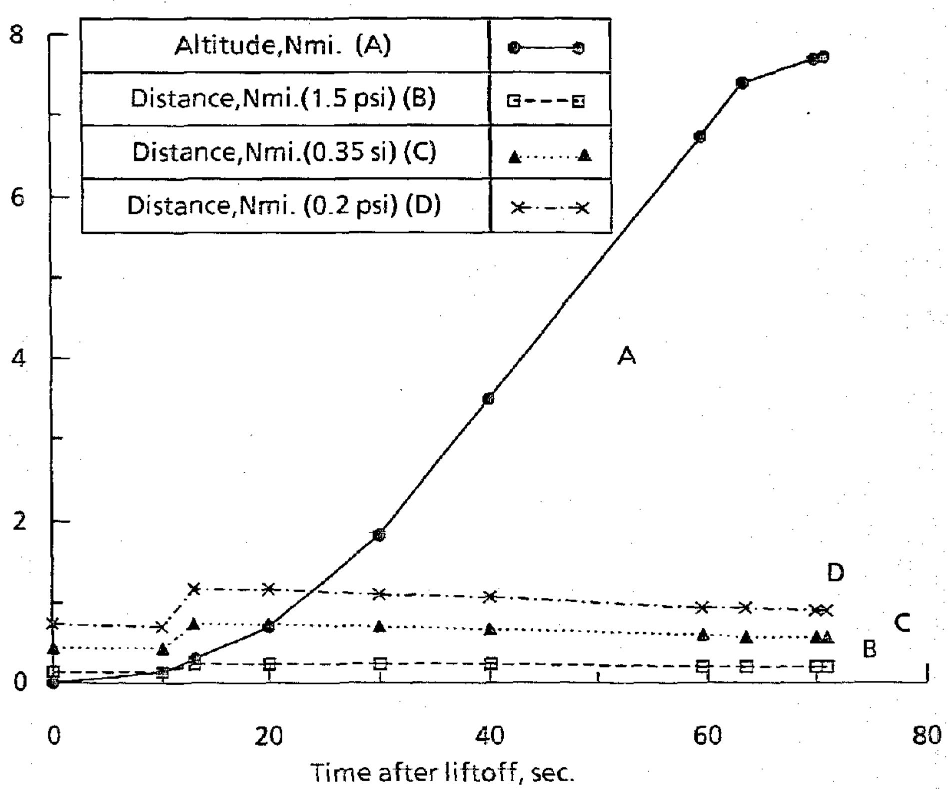


FIGURE 5-6 OVERPRESSURE AS A FUNCTION OF TIME (DELTA ELV)

The vehicle altitude from Figure 5-5 is also shown as Curve A in Figure 5-6 for reference.

In Fig. 5-6, the hazard distances first increase with time, and then decrease -- following the behavior of the potential explosive energy profile which is shown in Fig. 5-5. Furthermore, Fig. 5-6 can be interpreted as follows:

- in approximately the first 25 seconds, damage such as window breakage is possible in a distance of approximately 1 nmi. from the launch pad (or the location of vehicle impact with ground).
- at later times, key scenarios are:
  - a- all the propellant explodes at the vehicle altitude. The potential mechanical damage at ground level is negligible (even if maximum yield is assumed) because of the high altitude of the vehicle and its the large separation from ground
  - b- the vehicle falls back to Earth as one piece and explodes. This is a very unlikely scenario since the vehicle will breakup under the aerodynamic forces produced by the fall. Even in such a worst case scenario, Figure 5-6 suggests that the maximum overpressure distance will be less than 1 nmi. in the first 25 to 60 sec time frame; much smaller yet at later times because of the rapid consumption of propellants with time of flight. The location of the impact point will be governed by vehicle trajectory during the fall, which in turn depends on a number of factors as discussed in Section 5.3.
  - c- the vehicle breaks up at altitude, producing fragments, some of which may detonate as they impact ground. The hazard of item b above is now distributed over a broader region determined by the impact points of the fragments. The overpressure hazard distances around each impact point will be smaller than in b above. They will depend on additional factor such as number and size of fragments and their rates of consumption during their fall. This is further discussed in Section 5.3.

Off-range damage in any of the above cases will depend on the presence of population centers within a radius (of the explosion center) equal to the above distances (see Sec. 5.6).

Generally, the hazard from propellant explosion decreases rapidly with time into flight, except for the first 10 to 25 seconds. Activation of the Flight Termination System is likely to further reduce such explosion hazards by dispersing the propellant. Typically, the FTS is not activated during the first 8-12 seconds (depending on ELV, mission and site) in order to avoid damage to the pad facilities. This subject is discussed in more detail in Ch. 3, Vol. 1 and Ch. 10, Vol. 3.).

5.3 DEBRIS HAZARDS

A debris hazard exists even for a normal successful launch, primarily from jettisoned stages, shrouds and other components. These can be expected to impact within the impact limit boundaries of the flight corridor. The flight corridor is specified by applying safety considerations to the mission flight requirements, as discussed in Ch. 2, Vol. 1. Thus, hazards which cannot be eliminated are controlled. Since the launch facilities are located so that the vehicles will fly over largely uninhabited areas and oceans, the risks to third parties in normal operational situations are very low .

A debris hazard also exists due to failure modes such as malfunction turns (from gradual to tumbling turns) and premature thrust termination (from an accidental subsystem failure, commanded thrust termination or commanded vehicle destruction). Debris may be created either from breakup of the vehicle due to excessive aerodynamic pressure or explosion (accidental or commanded destruct). Major issues in assessing debris hazards include: what is the number, weight and shape of fragments? Where will they land? What is their impact force upon landing? What is their impact in terms of structure penetration and lethality?

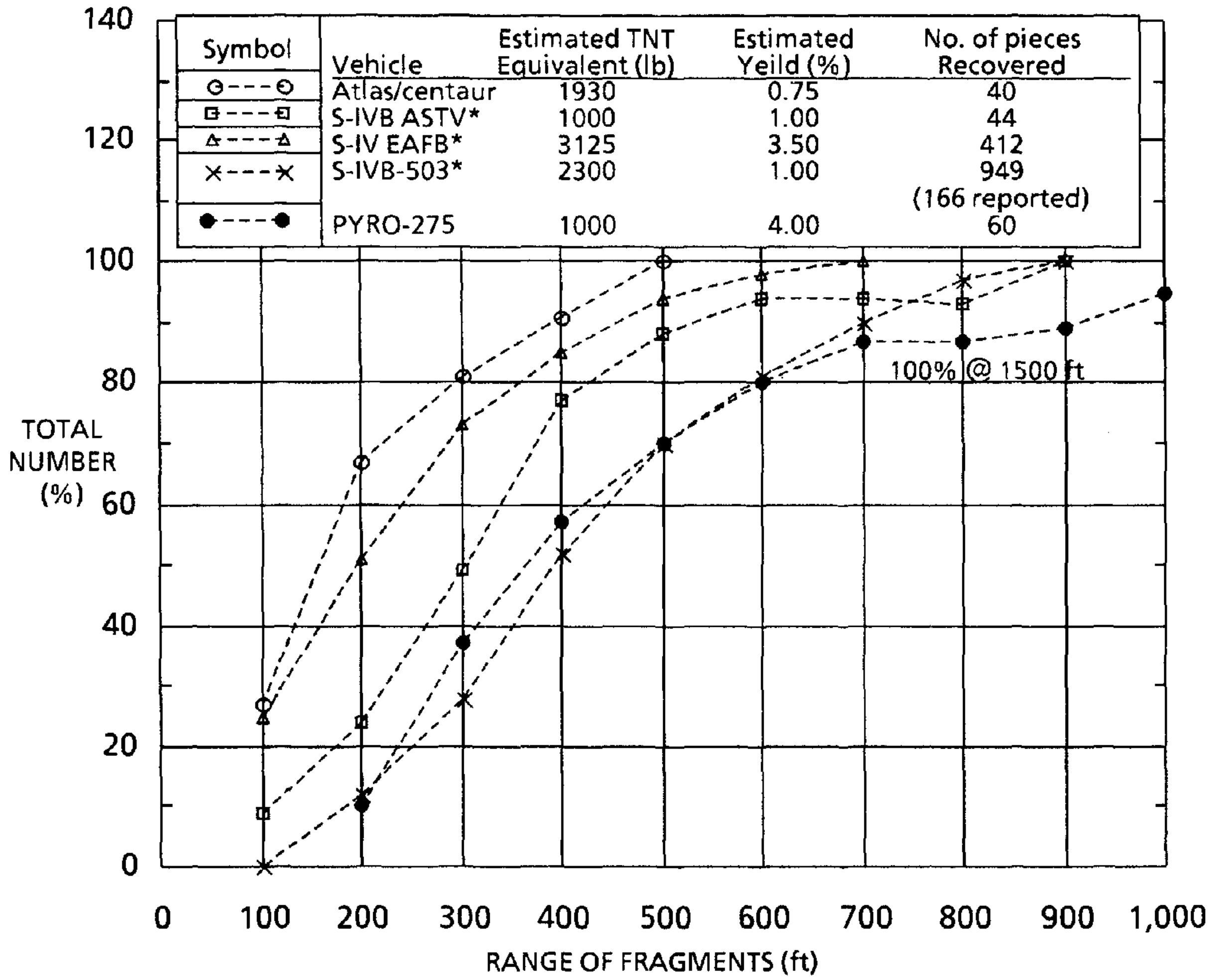
Illustrative examples of debris data from selected space vehicle explosions and test data (occurring at or near ground level) are shown in Figure 5-7 and Figure 5-8. These figures show the total number and weight distributions of fragments (respectively) as a function of range (i.e., distance). These distances were determined by the forces of the explosions.

Clearly, when a vehicle is in flight at significant altitude, the debris will land over a much larger area than in Figures 5-7 and 5-8. The distribution of debris impacts is dependent upon the forces acting on the fragments. Initially, the velocity vector of the vehicle is of primary importance and this contribution is affected by the velocity vectors resulting from the turns, tumbling and/or explosions. Thereafter, the effects of the atmosphere on the fragments during free fall (which depend on wind and the fragment size, shape and mass) become important. These issues lead to uncertainties in the fragment impact distribution which can be attributed to four basic sources:

(1) uncertainty in the vehicle state vector at vehicle breakup or destruct;
(2) uncertainty in any destruct velocity imparted to the fragment by a destruct system (or explosive failure);
(3) uncertainty in the atmospheric environment during free fall; and
(4) uncertainty in the fragment size, aerodynamic lift and drag.

Furthermore, impacting launch vehicle fragments can be divided into four categories:

(1) inert pieces of vehicle structure;
(2) pieces of solid propellant (some of which may burn up during free fall);
(3) vehicle structures which contain propellant (solid or liquid) that may continue to burn after landing (but are non-explosive). They pose the risk of starting secondary fires at the impact points; and



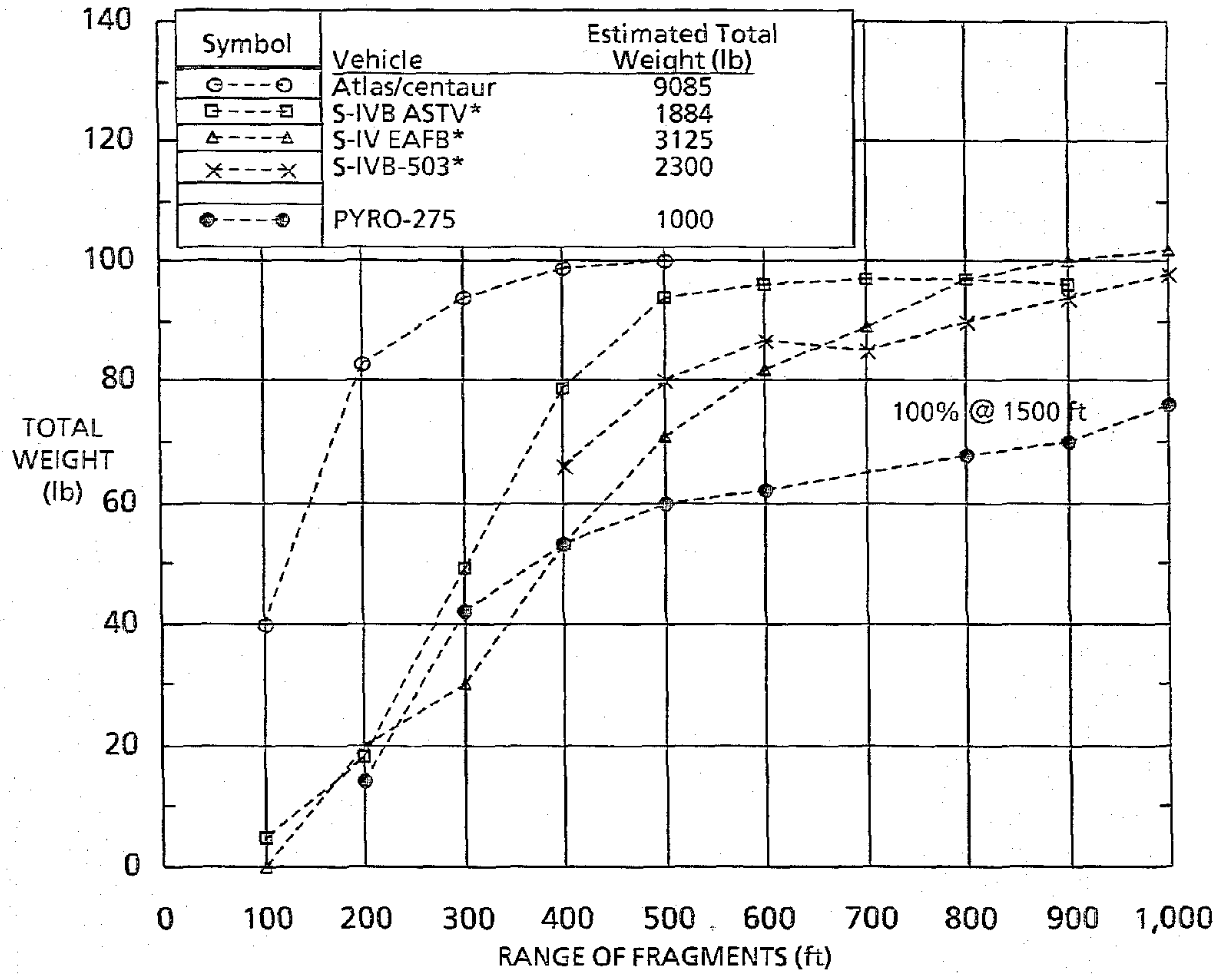
* Solid propellant used in Project Sophy

FIGURE 5-7. PERCENTAGE OF TOTAL NUMBER OF VEHICLE FRAGMENTS WITHIN RANGE INDICATED (Ref. 1)

(4) fragments which contain propellant and which can explode upon impact (if their impact velocity is greater than roughly 300 ft/s).

The casualty area of an impacting fragment is the area about the fragment impact point within which a person would become a casualty. Casualties may result from a direct hit, from a bouncing fragment, from a collapsing structure resulting from an impact on a building or other shelter, from the overpressure pulse created by an explosive fragment, from a fire or toxic cloud produced by the fragment or some combination thereof. The hazard area is increased if a fragment has any significant horizontal velocity component at impact which could result in bouncing or other horizontal motion near ground level.

Casualty area is also affected by the sheltering of people by structures. Structures may be divided into classes (for computational purposes) depending on the degree of protection they can afford.



* Solid propellant used in Project Sophy

FIGURE 5-8. PERCENTAGE OF TOTAL WEIGHT OF VEHICLE FRAGMENTS WITHIN RANGE INDICATED (Ref. 1)

Clearly, estimating a casualty expectation is a complex computational problem. Different Ranges approach the problem in different ways depending on the needs of the Range. Computer models may be used, but the sophistication varies greatly from Range to Range. A computer model called LARA (Ref. 9) treats casualty areas analytically and is presented in other chapters (Vol. 2, Ch. 4, and Vol. 3, Ch. 10).

5.4 FIRE HAZARDS

The fire hazards of accidentally released solid and liquid propellants depend on the details of the accident scenario including: the thermodynamic state of the propellant, the amount of the release, vehicle location and speed (on launch pad versus in flight), the presence of confining surfaces and ignition sources, etc. The major types of fires that can develop are:

- Fireball: where burning occurs in a ball of fire that expands and rises in the air (due to buoyancy forces) until the propellant is consumed.
- Pool fire: where a film of propellant is formed on the ground and burns with a flame attached to the film.
- Vapor cloud fire: where ignition is delayed and vapors are carried away by prevailing winds, thus forming a flammable cloud that may ignite at a later time.
- Various combinations of the above fires.

These fires are discussed below.

5.4.1 Fireballs

Fireballs are produced when the propellant is quickly vaporized or atomized. These conditions include flash vaporization of pressurized liquids and releases during flight at high speed. The vapors or fine droplets can then rise under the effects of buoyancy as they burn in the fireball.

The main damage mechanism is thermal radiation to people and property. Another damage mechanism is firebrands from burning solid propellants and hot debris which might start secondary fires where they land. A third damage mechanism is impact damage by vessel fragments which have been reported to travel large distances. Overpressure may also develop due to the initial high rate of energy release associated with vessel failure, but it is usually insignificant.

The damage potential depends on key fireball parameters such as diameter, rise rate, duration and temperature or emissive power. These parameters have been quantified in several experimental and analytical studies.(1) In fact, the ball diameter was found to scale roughly with the 1/3 power of the weight of released propellant.

The chemical composition of the products of combustion depend on the chemical composition of the propellants. The combustion products contain mainly water vapors and oxides of carbon and nitrogen. Thermal radiation emitted in the form of water vapor will be (partly) reduced by moisture absorption in the atmosphere. The transmitted radiation can impact people and structures. Table 5-6 shows critical radiation fluxes required to cause burn injury and start secondary fires (such as by igniting fuels placed inside and outside buildings). Note that as the exposure time increases, the required radiant flux decreases, as expected.

5.4.2 Pool Fires

Pool fires are produced when liquid propellants are accidentally spilled on the ground such as:

TABLE 5-6. MINIMUM CRITICAL RADIANT EXPOSURES NECESSARY TO IGNITE OR DAMAGE VARIOUS TARGETS (Source: Ref. 1)

CRITICAL IRRADIANCE, Btu/ft^{2}sec (cal/cm^{2}sec)

Exposure (seconds)	Int. Building Fuel	Ext. Building Fuel	Open Stacks of propellants	Human Beings	Aircraft
10	5.89 (1.60)	4.64 (1.26)	2.77 (0.75)	8.33 (2.26)	4.79 (1.30
20	3.91 (1.06)	3.87 (1.05)	1.99 (0.54)	5.77 (1.51)	2.39 (0.65)
30	3.02 (0.82)	3.50 (0.95)	1.62 (0.44)	4.39 (1.19)	1.59 (0.43)
40	2.62 (0.71)	3.28 (0.89)	1.40 (0.38)	3.72 (1.01)	1.40 (0.38)
50	2.32 (0.63)	3.13 (0.85)	1.29 (0.35)	3.32 (0.90)	1.22 (0.33)
60	2.21 (0.60)	3.02 (0.82)	1.18 (0.32)	3.09 (0.84)	1.03 (0.28)
70	2.18 (0.59)	2.88 (0.78)	1.11 (0.30)	2.91 (0.79)	0.94 (0.255)
80	2.14 (0.58)	2.80 (0.76)	1.03 (0.28)	2.80 (0.76)	0.856 (0.235)
90	2.10 (0.57)	2.69 (0.73)	0.92 (0.25)	2.69 (0.73)	0.81 (0.22)
100	2.06 (0.56)	2.65 (0.72)	0.88 (0.24)	2.58 (0.70)	0.77 (0.21)
110	2.03 (0.55)	2.62 (0.71)	0.85 (0.23)	2.47 (0.67)	0.756 (0.205)
120	1.99 (0.54)	2.58 (0.70)	0.81 (0.22)	2.36 (0.64)	9.74 (0.20)
130	1.95 (0.53)	2.54 (0.69)	0.79 0.215)	2.29 (0.62)	0.726 (0.197)
140	1.92 (0.52)	2.51 (0.68)	0.77 (0.21)	2.21 (0.60)	0.719 (0.195)
150	1.88 (0.51)	2.49 (0.675)	0.756 (0.205)	2.14 (0.58)	0.704 (0.191)
160	1.84 (0.50)	2.47 (0.67)	0.74 (0.20)	2.06 (0.56)	0.697 (0.189
170	1.84 (0.50)	2.45 (0.665)	0.719 (0.195)	1.99 (0.54)	0.693 (0.188)
180	1.84 (0.50)	2.43 (0.66)	0.712 (0.193)	1.95 (0.53)	0.689 (0.187)
190	1.84 (0.50)	2.41 (0.655)	0.708 (0.192)	1.92 (0.52)	0.686 (0.186)
200	1.84 (0.50)	2.40 (0.65)	0.70 (0.190)	1.92 (0.52)	0.682 (0.185)
300	1.84 (0.50)	2.33 (0.631)	0.659 (0.179)	1.92 (0.52)	0.667 (0.181)
600	1.84 (0.50)	2.29 (0.621)	0.641 (0.174)	1.92 (0.52)	0.645 (0.175)

- from a vehicle in pre-launch phase: this scenario is outside the scope of this study since its impact is not likely to extend outside the Range boundaries.
- from ground operations such as propellant transport to the Range and storage, handling and transfer within the Range. In this case, the impact may occur outside the Range boundary.

A spilled liquid will spread on the ground under the effect of gravity, filling small-scale crevices in a ground with surface roughness or large-scale depressions in an undulating terrain. While spreading, cryogenic propellants (such as liquid hydrogen and oxygen) will boil violently due to heat transfer from the relatively warm ground. A propellant at ambient temperature (such as RP-1) will evaporate more slowly. Some flash vaporization of cryogenic liquids will also occur because their vessels are usually maintained at slightly above atmospheric pressure.

Ignition produces a pool fire with a flame base which spreads along with the liquid film and a flame height determined by the rate of evaporation and the rate of mixing of fuel and oxidizer. The overall character of such a pool fire is essentially a turbulent diffusion flame which may continue to expand on flat ground (or remains stationary if the liquid has accumulated in a depression area) until it runs out of fuel.

The danger of pool fires consist of thermal radiation to people and property (as in the case of fireballs) and direct flame impingement on structures near the fire.

5.4.3 Vapor Cloud Fires

In the pool fire scenario described above, if:

- the liquid pool does not ignite immediately after the release, because of lack of an ignition source; and
- the released propellant has a high vapor pressure such as liquid hydrogen, oxygen, nitrogen, air or methane which boil due to heat transfer from the environment and not from a fire;

then, a large amount of vapor will be produced and transported by prevailing winds to form a vapor cloud. In this scenario, the resulting cloud is elongated in shape and is called a "plume". Its leading edge advances with the wind and its trailing edge is formed at the evaporating pool (the source of the vapors). As the leading edge moves further downwind, ambient air is entrained in the cloud, thus increasing its volume and decreasing the vapor concentration. This process is called atmospheric dispersion and is discussed further in the next section.

If a flammable cloud encounters an ignition source, a fire will spread through the cloud, engulfing in flames whatever is contained in the cloud. This is referred to as a vapor cloud fire. Under some conditions (particularly the presence of obstructions or confinement in the cloud) overpressure can be produced, posing the added risk of mechanical damage.

Alternatively, as the cloud disperses, the vapor concentration may drop below the flammable limit prior to encountering an ignition source. Thus, the hazard is dissipated without any adverse impact.

5.5 TOXIC VAPOR CLOUDS

The evaluation of the toxicity of any material is a very complex subject. Toxicity data are very sparse and questionable except for the common toxins. When available, they are usually for continuous exposures as one would find in a factory environment and not for the short exposures characteristic of launch operations.

Still, the issue is of great interest because toxic materials may be released during ELV launches as combustion products, or in the event of an accident, as uncombusted propellants. The most notorious ones are hypergolic liquid propellants such as monomethyl hydrazine, Aerozine-50 and nitrogen tetroxide. Their chemical properties and toxic Threshold Limit Values (TLV) are listed in Appendix B along with other characteristics of interest. If such materials are released in the environment, they may be carried by the wind and travel windward as they disperse. This atmospheric dispersion is described below.

5.5.1 Atmospheric Dispersion

Over the years, the subject of atmospheric dispersion has been studied extensively in connection with air pollution studies from power plants and automotive vehicles. These studies addressed the case of continuous releases from normal operations where pollutant concentrations were monitored over long periods of time.

In this study, the interest is mainly in larger uncontrolled "instantaneous" releases (as would occur in an accident). Then, a large amount of potentially noxious vapor may be produced and transported by prevailing winds to form a vapor cloud. There are two main types of vapor clouds:

- a "plume": an elongated cloud whose the leading edge travels with the wind, while the trailing edge remains stationary at the source of the vapors. Conditions which produce a plume are described in the preceding section;
- a "puff": a more or less spherical cloud where both leading and trailing edges move together downwind.

In reality, a combination of these two cloud geometries may occur, depending on accident conditions.

As the cloud travels downwind, ambient air is entrained in the cloud; this increases its volume and decreases the vapor concentration. The process can be further complicated by chemical interactions among hypergolic vapors and between vapors and entrained air.

Such cases of large "instantaneous" releases have also been studied experimentally. Large scale tests involving the spillage of large quantities of chemicals were carried out and concentrations were measured downwind. The most notable tests, carried out as part of national and international programs include:[21]

(1) the liquefied natural gas (LNG) dispersion tests at the Naval Weapons Center, China Lake, California, for the US Department of Energy;

(2) the ammonia spill tests at the above location for the Fertilizer Institute and the US Coast Guard;

(3) the Porton Down tests in England involving the instantaneous release of Freon;

(4) the heavy gas dispersion trials on behalf of the Health and Safety Executive of the British Government and other participants; and

(5) the LNG spill tests conducted by Shell UK Ltd. at Maplin Sands, England.

Based on such tests, it is recognized that cloud dispersion depends mainly on:

- ambient conditions such as wind, atmospheric stability and local terrain.
- the buoyancy of the vapor cloud. It is important to determine whether the cloud is lighter or heavier than air because the former will disperse much faster than the latter. The presence of aerosols (fine droplets sprayed from the spilled liquid) increases the effective density of the cloud and modifies its dispersion characteristics. Also, cloud density may vary in space and time so that some portions may be lighter than air and others heavier.
- the size and location of the release, i.e., whether it is on ground level (from an accident on the launch pad) or from an elevated altitude (from an accident in flight).

There are several models in the literature describing the dispersion behavior of heavier-than-air gases under a wide range of conditions.[14a, b, c] Models which discuss the dispersion of vapors released passively (as from a boiling pool of liquid) include Van Ulden,[15] Britter,[16] and Colenbrander.[17] There are also models in the air pollution literature dealing with release of neutral and positively buoyant vapors from stacks.

In general, the dispersion of vapors in the far-field (after sufficient dilution) can be predicted with reasonable accuracy by the standard Gaussian models of Pasquill[18] and Gifford.[19] However, in the near-field, these models have to be modified to take into account the effects of initial gravitational spreading, jet mixing or the effects of aerosol evaporation.[20]

5.5.2 Rocket Exhaust Products

Most of the combustion products from rocket engines are harmless or unlikely to exist in concentrations which would affect the health and safety of third parties. These combustion products may include:

- water and water vapors
- nitrogen
- hydrogen

- carbon monoxide and dioxide
- hydrogen chloride
- aluminum oxide

Of these combustion products, carbon monoxide and hydrogen chloride may be considered hazardous. Aluminum oxide is not toxic, but may contribute to certain lung diseases if exposure persists over time. The remaining combustion products are not dangerous unless present in sufficient concentration to cause asphyxiation, which is not the case. Threshold Limit Values (TLV) for major combustion products are given in Table 5-7 for various exposure durations for both controlled (Range personnel) and uncontrolled (third party) populations.

For illustration, Figure 5-9 shows results from a model using the NASA/MSFC (buoyant-rise, multilayer dispersion model of exhaust products) to compute peak instantaneous concentrations of hydrogen chloride as a function of downwind distances from the launch pad for sea breeze meteorological conditions and certain vehicle configurations. Also, Figure 5-9 shows the exposure criteria limit (as given in Table 5-7) for 10 minute-exposure of uncontrolled populations (third parties). Note that this limit is not exceeded at downwind distances of interest. In 1985, the Committee on Toxicology, Board on Toxicology and Environmental Health Hazards, Commision on Life Sciences, National Research Council published a document entitled "Emergency and Continuous Exposure Levels for Selected Airborne Contaiminants," Volume V.[20a] This document updates recommendations for public exposure to the hydrazines and creates a new category, Short-term Public Emergency Guidance Levels (SPEGL's) for up to 24 hours for hydrazine propellants. The data in this document affects values for the uncontrolled population exposure to hydrazine shown in Table 5-7.

5.5.3 Releases During Accident Conditions

In the case of a near-pad explosion, all of the propellant is unlikely to be combusted. Thus, a vapor cloud containing vapors and aerosols of hydrazine, nitrogen tetroxide and hydrocarbon fuels might result. Other chemicals such as fuel additives and contaminants may also be present. These materials are toxic (see TLV values listed in Appendix B) and in high concentrations may cause adverse health effects, particularly if meteorological conditions at the time of the accident do not favor rapid dispersion to below toxic levels.

The Titan 34 D explosion at WSMC of April 18, 1986, produced a vapor cloud containing toxic Aerozine-50 (Unsymmetric dimethylhydrazine and hydrazine blend) and nitrogen dioxide. There was no verified exposure of third parties to toxic concentrations exceeding established limits.

TABLE 5-7. EXPOSURE CRITERIA FOR SOME COMBUSTION PRODUCTS AND PROPELLANTS

Substance	Controlled Populations [a]				Uncontrolled Populations [b]					
	TLV. [c] ppm	Short-Term Emergency limits, ppm			Exposure From Ordinary operations, ppm			Emergency Exposure, ppm		
		10 min.	30 min.	60 min.	10 min.	30 min.	60 min.	10 min.	30 min.	60 min.
HC1	3	30	20	10	4 (d)	2 (d)	2 (d)	7 (d)	3 (d)	3 (d)
CO	50			200			30 (e)		--	125 (g)
mg/m^3	10	50 (g)	25 (g)			--			--	
NO_2 (N_2O_4)	3	30	20	10	1 (f)	1 (f)	1 (f)	5 (f)	3 (f)	2 (f)
Hydrazine	0.1	100	20	10	1 (g)				--	
UDMH	0.5	100	50	30	0.5 (g)				--	
$AlCl_3$ (mg/m^3)	10 (h)		--			--			--	

(a) Controlled populations consist of persons with known medical histories, subject to periodic health checks, and generally under the control of the responsible agency. Such persons are normally employees with jobs that will potentially result in exposure to known contaminants.

(b) Uncontrolled populations consist of persons with unknown medical histories, not subject to periodic health checks, and not generally controlled by the responsible agency. The general public is included in this classification.

(c) No short duration exposure criteria for controlled populations appear applicable for ordinary launch operations. Threshold Limit Values (TLV) are time-weighted concentrations for 7 or 8 hour work days and a 40-hour work week, except that the values for HCl and NO_2 are also considered ceiling values not to be exceeded. TLV's are thought to be conservative for short duration exposures of controlled populations for relatively infrequent normal operations.

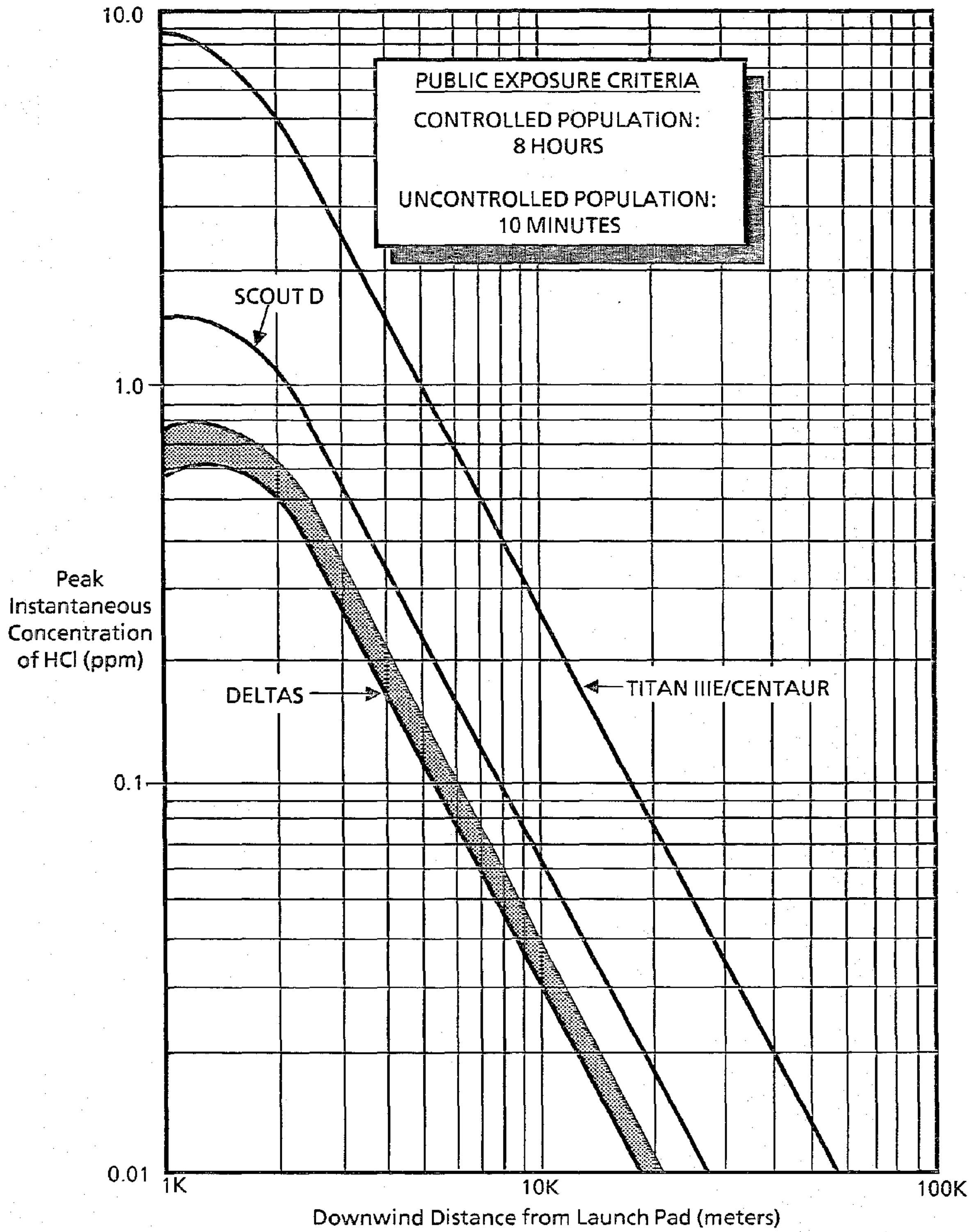
(d) While there are no criteria for short-term exposure of uncontrolled populations to HCl which have official standing, the values quoted here have been proposed by a responsible organization after careful study of the problem.

(e) Based on 1.5% Carboxyhemoglobin in 1 hour exposure.

(f) There are no officially accepted criteria for short-term exposure of uncontrolled populations to nitrogen oxides. The criteria given here have been proposed by a responsible organization after careful study.

(g) Ordinarily set equal to the 8 hour industrial TLV: i.e., 1/48 of the acceptable industrial dose.

(h) Based on hydrolysis to HCl. In subsequent discussion, $AlCl_3$ is considered only in terms of its contribution to overall HCl levels.



NOTES:

1. The concentrations for for the 3, 6 & 9 Castor Deltas fall within the shaded area.
2. To convert meters to statute miles, multiply by 6.2×10^{-4}.

FIGURE 5-9. ESTIMATED PEAK HCl CONCENTRATIONS DOWNWIND OF LAUNCHES (SEA BREEZE METEROLOGICAL CONDITIONS) Ref. 14

However, reports indicate that doctors examined 74 people for possible exposure to the clouds and two were kept in the hospital for observation (see also Ch. 10).

Depending on their chemical properties (see Appendix B), accidentally released vapors may only be flammable (e.g., hydrogen) or also toxic (e.g., hydrazine and nitrogen tetroxide). The Threshold Limit Values (TLV) for exposure to various toxic propellants or their combustion products, shown in Table 5-7 and Appendix B, are on the order of 0.1-100 ppm, while typical flammability limits are on the order of 1-10% (i.e., 10,000-100,000 ppm). Because the minimum vapor concentrations with toxic impacts are much below those required to sustain a flame, the potential size of a toxic cloud is much greater than that of a flammable cloud. Accordingly, for equal amounts of released propellants, the potential for toxic impacts is of greater concern than for fire damage.

5.6 OFF-RANGE IMPACTS ASSOCIATED WITH ELV OPERATIONS

This section presents a summary discussion of the potential off-range impacts associated with ELV operations (See Table 5-8). Potential ELV hazards were discussed in this chapter with no explicit mention of the associated probabilities. However, each hazard is tied to a particular probability value -- that of the occurrence of the enabling conditions. This fact should be remembered in assessing the significance of potential off-range impacts. The subject of assessing impacts from the perspective of both their magnitude and probability is referred to as Risk Analysis, and it, along with the various methods used to quantify risks, is discussed in detail in Chs. 8 and 9, Vol. 3.

Illustrative examples of the application of Failure Analysis methods to space systems are given in Ch. 9, Vol 3. They are typically focused on a specific phase of launch operations and are rarely integrated, as is attempted (qualitatively) below.

Examples of the results from such a preliminary hazard analysis are given in Table 5-8 for the main phases of ELV operations: pre-launch, launch and pre-orbital. As usually done, the failure types are classified in a manner compatible with the availability of data. For example, in Table 5-8, all failures leading to vehicle break up in flight,are lumped into one category for which a failure rate may be estimated based on historical data for each ELV.

Hazard Analysis is then used to analyze the consequences of the types of accidents identified in Failure Analysis. These consequences include explosion, fire, toxic vapor clouds and inert debris. The principles of physics and chemistry are used, along with data from historical experience, testing and engineering judgment, to describe the hazards and potential impact severity. For example, the strength of an explosion or fire may be described and associated with potential damages (by overpressure or heat) to people and/or property. Estimates of the magnitude of the potential

TABLE 5-8 ADVERSE OFF-RANGE ACCIDENT IMPACTS FOR VARIOUS PHASES OF SPACE LAUNCH OPERATIONS

Phase/Time, sec. after lift-off	Selected ELV Failure Types	Probability of Failures	Potential Off-Range Impacts*			
			Explosion	Fire	Toxicity@	Inert Debris
Pre-launch						
	Storage tanks	Improbable	(a)	(a)	(b)	N/A
	Small Leaks	Occasional	(a)	(a)	(a)	N/A
Launch						
0-12	Fallback/Tipover	0.04 - 0.1 (j)	(c)	(d)	(b)	(d)
25-70	Thrust/Guidance Failure and Break-up in flight	0.04 - 0.1 (j)	(e,f)	(e,f)	(e,f)	Ec = 2.3XE-8 with FTS (Near ESMC, h)
70-400	Thrust/Guidance Failure and Break-up in flight	0.04 - 0.1 (j)	(g)	(g)	(g)	Ec = 4.0XE-3 without FTS (near ESMC, i)
Pre-orbital						
	Thrust/Guidance Failure and Break-up in flight		(g)	(g)	(g)	Ec = 8.0XE-7 regardless of FTS (Over Africa, l)

NOTES:

* Risk can be described by number of casualties (or dollar loss) weighted by its probability, or by an expected casualty Ec. The probability depends on the failure mode and accident rates and other accident circumstances.

@ For hypergolic propellants only

a. Large separation distances in Range siting and propellant storage preclude such hazards.

b. Possible with very large releases and adverse meteorological conditions.

c. Depends significantly on yield which in turn depends on accident scenario; window breakage is possible.

d. No likely impacts off-range.

e. Remaining fuel and hazard decrease rapidly as time ellapses after launch.

f. Hazard depends on number of fragments, size and impact points.

g. Remaining propellant (if any) is likely to dissipate in flight.

h. Ec = expected casualties per launch Ref. 22

i. Ec = expected casualties per launch (see Sec. 9.1)

j. Based on historical failure rates of all ELVs as given in Ch. 3.

l. Ec from Table 9-1, Ch. 9.

N/A = Not Applicable

damage may be expressed in terms of an impact area (or footprint) surrounding the location of the accident.

To do so, a range of possible accident circumstances have to be specified to allow a quantitative estimation. A further break down of the hazards in various ways may be needed to make the analysis tractable. For example, in Table 5-8, the hazards are divided into those (explosion, fireball and toxic releases) that may occur while the vehicle is in flight, versus those occurring when the vehicle or its fragments impact the ground. The break-down of consequences in Table 5-8 varies with time during the launch phase. As time elapses after liftoff, the quantities of propellants on-board will decrease, thereby affecting their potential hazards. This was discussed in detail in Section 5.2.4 for explosion hazards.

Risk Analysis is finally used to describe (for a particular activity) both the probabilities of accidents and the possible damages or losses associated with them, accounting for uncertainties in the occurrence of the accidents and in the circumstances surrounding them. For example, there are uncertainties as to what accident is likely to occur at a particular location and how many people would be present at that location at the time of the accident. A set of circumstances is defined (scenario) and their probability is estimated. For each set, the results of the Failure Analysis (frequency of an accident) and Hazard Analysis (area of damage) are combined to estimate an expected damage (e.g., a number of people affected with a particular frequency per year or per event). The overall outcome of the analysis is a probability distribution function (PDF) for the potential damages that can be associated with a particular hazardous activity. An expected value for potential damage (e.g., casualty expectation, E_c) is often calculated from that probability distribution.

Such expected casualty values have been estimated in an approximate manner for ELV-type vehicles, but only for a few specific scenarios involving inert debris hazards as shown in Table 5-8, namely:

- inert debris risks during the first 10-70 sec of launch, with and without a Flight Termination System.[22]
- inert debris risks during pre-orbital operation, with and without a Flight Termination System.

In Table 5-7, note that for the scenarios involving explosion, fire and toxic hazards, only a qualitative description of the potential off-range impacts is given because either their probabilities or magnitudes have not been quantified. These descriptions are given as footnotes in Table 5-8, to summarize key considerations in understanding these impacts and of their determining factors.

5.7 PERSPECTIVES ON THE MAGNITUDE OF THE HAZARDS ASSOCIATED WITH ELV PROPELLANTS

In the previous sections, the major hazards associated with ELV propellants were discussed. There are a number of hazards (explosions, debris, fires, toxic vapor clouds) each of which depend on a number of parameters such as propellant properties, quantity, mode of release, etc. Clearly, these hazards are very complex and multi-dimensional. In this section, a few reference points are provided to place these hazards in perspective compared to more familiar hazards. Only a partial perspective is provided because:

(a) the focus here is on the magnitudes of these hazards and not on their probabilities or likelihood of occurrence. This is addressed in Chs. 9, 10, Vol. 3, where a more complete discussion of public risk perspectives is provided.

(b) the comparison with other hazards is presented in a very simplified fashion, focusing only on selected dimensions of the hazards.

In simple terms, concern with ELV propellant hazards can be attributed to the following factors:

(1) rocket propellants are highly energetic fuels and most are inherently hazardous;

(2) large quantities of propellants are involved in space launch operations; and

(3) launch operations are inherently complex and have many potential failure modes.

The following discussion places these concerns in their proper perspective.

First, propellants such as liquid hydrogen, liquid oxygen and RP-1 have been used extensively in the chemical industry. They have been processed, transported and stored for several decades with a remarkable safety record. Also, the chemical industry uses (on a daily basis) chemicals which are even more hazardous than ELV propellants, such as acetylene and ethylene oxide (which are extremely explosive) and hydrogen chloride and hydrogen cyanide (which are extremely toxic).

Selected key properties which affect the hazard potential of such chemicals are listed in Appendix B and in Table 5-9. Note that the range of propellant properties are sometimes exceeded by other chemicals. For example, the flammability limits of acetylene and ethylene oxide are wider than those of hydrogen. In addition, these two chemicals can react autocatalytically without the need for an oxidizer, if initiated by heat, pressure or shock. On the other hand, hydrogen requires oxygen to react. Generally, the broader the flammability range, the easier it is to create a fire or an explosion. Thus, these two chemicals are more likely to ignite than hydrogen.

Second, the quantities of chemicals used in industry are often greater than those of propellants in ELV operations. This is illustrated in Table 5-10 which provides data for various space vehicles and for the storage and transportation of fairly common fuels such as LNG, LPG and gasoline. For each case,

TABLE 5-9. SELECTED PROPERTIES AFFECTING THE HAZARDOUS BEHAVIORS OF LIQUID PROPELLANTS AND CHEMICALS

Property	LH_2	LO_2	RP-1*	Hydrazine	N_2O_4	LPG**	LNG+	Gasoline	Acetylene	Hydrogen Cyanide	Ethylene Oxide
Boiling Point or Distillation Range (at 1 atmosphere), °K	20.3	90	440-539	387	294.3	227	111.6	310-478	189	299	284
Vapor Pressure, at 311° K, kPa	gas	gas	1.4	5	273	gas	gas	46	gas		233
Flash point at 1 atmosphere, °K	gas	NA	~330	310-525	NA	gas	gas	~230	gas		
Net heat of combustion, kJ/kg	119,900	NA	43,200	49,300	NA	46,400	50,000	44,500	48,200	13,100	27,700
Btu/lb	51,630	NA	18,600	21,300	NA	20,000	21,500	19,200	20,800	5,600	12,000
Flammability limits in air (volume %)											
Lower	4	NA	0.6	4.7	NA	2.2	5.3	1.7	2.5		2
Upper	75	NA	4.7	100	NA	9.6	15	7.6	81		100
Autoignition temperature at 1 atm, °K	858	NA	511	497-543	NA	766-877	813	501-744	578		
Minimum Electric Spark Ignition Energy, mJ	0.02	NA	0.2			0.29	0.29	0.24			
Approximate laminar flame speed, m/s	2.7-3.3	NA	0.3-0.6				0.37-0.45		0.37-0.43		
Comments		Intensifies combustion with most materials			Very strong oxidizer				Unstable under heat or shock	Extremely toxic	Unstable in presence of heat and may undergo autocatalytic polyminization and decomposition

Source: Data are taken from a number of handbooks and reports and are approximate.

Notes:
NA = Not applicable because it is an oxidizer, not a fuel
* taken as mainly kerosene
** taken as mainly propane
\+ taken as mainly methane

TABLE 5-10. COMPARISON OF CHEMICAL ENERGY CONTENTS OF SPACE VEHICLES AND OTHER INDUSTRIAL OPERATIONS

System	Propellant/fuel	Weight Klb	Heat of Combustion Btu/lb*	Total Chemical Energy Btu
Atlas	RP-1/LO_2	303	5,850	1.8×10^9
Centaur	LH_2/LO_2	30.7	7,200	0.2×10^9
				2×10^9
Delta				
Booster	Solid (Castor (IV)	186	1,950**	0.36×10^9
Stage 1	RP-1/LO_2	179	5,850	1.0×10^9
Stage 2	Aerozine-50/N_2O_4	13	7,200	0.09×10^9
Stage 3	Solid	2.3	1,950**	0.004×10^9
				1.5×10^9
Titan III				
Stage 0	Solid (UTP-300 1B)	464	1,950**	0.9×10^9
Stage 1	Aerozine-50/N_2O_4	294	2,000	0.59×10^9
Stage 2	Aerozine-50/N_2O_4	69	2,000	0.14×10^9
Transtage	Aerozine-50/N_2O_4	9	2,000	0.02×10^9
				1.7×10^9
LNG tanker (Cryogenic)				
1 tank: 25,000 m^3	LNG	23,000	21,500	500×10^9
entire ship: 5 tanks	LNG	115,000	21,500	$2,500 \times 10^9$
Cryogenic Storage Tank (100,000 bbl)	LPG	20,000	20,000	400×10^9
Pressurized Spheres (1,200 m^3)	Propane	950	20,000	19×10^9
Jumbo jet fully loaded (50,000 gallons)	Jet A	390	18,600	7.3×10^9
Rail tank car	Propane	139	20,000	2.8×10^9
Tank Trucks (10,000-20,000 gal.)	Gasoline	60-120	18,000	1.1 to 2.2×10^9

* The heat of combustion is given in Btu per lb of fuel/oxidizer mixtures for propellants and per lb of fuel for non-space applications.

** Assumed to be same as TNT.

the table gives the total weight, heat of combustion per unit mass, and the total chemical energy. It also would have been desirable to provide the explosive (TNT) yield for each case. However, this would require the definition of a pertinent accident scenario for each (as was done in Sec. 5.2.4) and the estimation of a reasonable yield.

In view of the lack of such data, instead the total chemical energy is used as a rough indication of the magnitude of the potential hazard which is reasonable for propellants and fuels. In terms of total chemical energy alone, three typical launch vehicles are approximately:

- equivalent in order of magnitude to a gasoline truck or a rail tank car of LPG.
- one order of magnitude smaller than a pressurized LPG sphere.
- two orders of magnitude smaller than standard cryogenic tanks of LNG and LPG.
- three orders of magnitude smaller than an LNG ship.

Third, although ELV launch operations are inherently more intricate and complex than conventional chemical and transport operations, the safety precautions for ELV operations are far greater than those for other more common activities. For example, launch sites are separated significantly from population centers while chemical plants and fuel tank farms are located within cities.

An additional perspective on the magnitude of the hazards of ELV propellants relative to other fuels and chemicals can be obtained by comparing their respective past accident data. This is presented below for explosion accidents.

Data summarized in Table 5-11 involve major chemical process and transportation activities where the explosive yield was 40,000 pounds of TNT or greater. The table provides a brief description of each accident, identifies the chemical involved, the approximate quantity released (pounds) and the TNT equivalent weight (reported by the accident investigators based on the observed damage at the location of the accident). The TNT equivalent weights ranged from 40,000 to 125,000 pounds, which is roughly the same order of magnitude as that estimated conservatively for worst case propellant accidents in Table 5-3.

Unfortunately, similar historical data on space vehicle accidents may be restricted or classified and are not readily available in the open literature. The data found in the open literature are shown in Table 5-12 for large SRM explosions. No comparable data were found for liquid propellants. The reported TNT equivalent weights range from 9 to 42,000 pounds, a range lower than yields from industrial/transportation accidents and lower than the estimates for worst case propellant accidents in Table 5-3.

Although the historical data and comparisons presented above are limited in scope and depth, they still suggest that the hazards anticipated from ELV propellants can be considered to be qualitatively similar in type and magnitude to those associated with comparable chemicals and fuels commonly used in chemical processing and transportation activities.

TABLE 5-11. EXAMPLE OF MAJOR ACCIDENTAL EXPLOSIONS OF FUELS AND CHEMICALS AND THEIR TNT EQUIVALENT WEIGHTS (ESTIMATED FROM ACTUAL DAMAGES) (Ref. 20)

Date	Location	Chemical	Quantity Released klb	TNT Equivalent Weight klb	Accident Description
	CHEMICAL PROCESS INDUSTRY				
1970	N.J.	Heavy Hydro-carbon and hydrogen	251	110	Failure of high pressure reactor due to localized overheating. Blast was highly directional. Peripheral damage was used to indicate yield.
12/9/70	Port Hudson, MO	Propane	150	110	Spill from a pipeline rupture produced cloud 460 m long, and 3-6 m high.
1/20/68	Parnis, Holland	Light Hydrocarbons	110-220	44	Breaking of water-oil emulsion in stop oil tank caused cloud.
6/1/74	Flixborough, England	Cycloxane	79	40	Poorly installed 500 cm pipe failed. Ignition in 25-35 sec.
4/17/62	Doe Run, KY	Ethylene Oxide	48	40	Tank containing ethylene oxide became contaminated with ammonia. Tank ruptured, dispersed ethylene oxide into air. Ignition was immediate.
	TRANSPORTATION				
9/21/74	Houston, TX	Butadiene	176	44-125	Accident in rail yard punctured rail tank car. Amount of spill in 2-3 minutes not known. Ignited by locomotive 180 m away. Estimate 14-21 kPa over-pressure at 300 m from point of rupture.
7/19/74	Decatur, IL	Propane	139	44-88	Accident created 56 cm x 65 cm hole in end of rail car, releasing contents.

TABLE 5-12. HISTORY OF LARGE SRM EXPLOSIONS AND THEIR TNT EQUIVALENT WEIGHTS

MOTOR TYPE	SIZE LGTH. ft.	SIZE DIAM. ft.	PROPELLANT TYPE	HAZ. CLASS	INIT. WT. klb.	TNT EQUIV WT. klb.	INCIDENT DESCRIPTION	IMPACT SPEED ft/s	IMPACT SURFACE
TRIDENT F/S	15.5	6.2	CXDB	1.1	42	42 +		<800	WATER
STS Seg.		13.0	TP-H1123	1.3	280	28			
TRIDENT S/S	8.3	6.2	CXDB	1.1	19	19 +		<800	WATER
M^2 Stage II	13.5	4.3	PBAA	1.3	10.5	10.5		>600	SAND
M^2 Stage I	24.5	5.4	PBAA	1.3	45.8	6.4-8.2		380-490	SAND
TITAN III Seg.	12.0	10.0	PBAA	1.3	82	6.6-11.5		670	CONCRT.
M^2 Stage III	5.2	3.1	DDP77	1.1	3.7	3.7 +		<600	SAND
POLARIS F/S	15.1	4.5	ANP2655	1.3	15.2	2.6		>490	SAND
POLARIS S/S	7.0	4.5	ANP2655	1.3	7.3	1.81		>575	SAND
M^2 Stage II	13.5	4.3	PBAA	1.3	10.5	0.105		40	STEEL
MX Stage I		7.7	HTPB	1.3	95.1	0.06	HOT DESTRUCT, LSC = 325 g/ft		
MX Stage III		7.7	HTPB	1.3	15.5	0.009	HOT DESTRUCT, LSC = 40 g/ft		

Source: Ref.7

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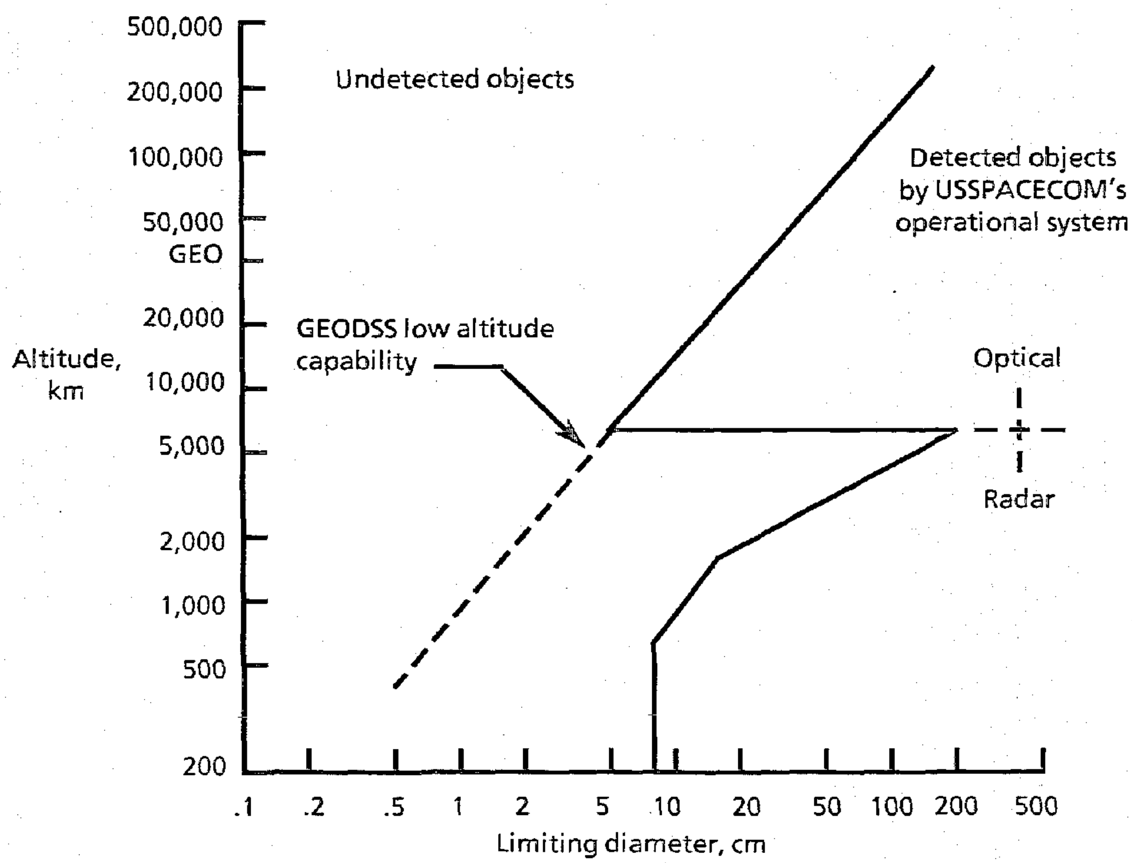
6. ORBITAL COLLISION HAZARDS

6.1 ORBITING SPACE OBJECTS

It is important to estimate the hazards of on-orbit collisions between space objects because the US may be liable for any damage to a foreign country, or satellite caused by a US spacecraft. The latest NASA Satellite Situation Report lists 1,702 spacecraft in orbit and 5,130 large debris such as spent rocket stages and payload shrouds.[4] Expanding the count to include trackable debris, the tally was 18,145 cataloged space objects as of June 30, 1987. Of these, 5,763 are from the US and 11,603 from the USSR. Of the total, approximately 7,000 are still in orbit (the rest have decayed and re-entered). Radar-trackable objects in space (i.e., larger than about 10 cm across) are monitored and cataloged by both the US Space Command (USSPACECOM). Considerably more objects and debris too small to be trackable are in orbit, as indicated in Figure 6-1.[1] Measurements using the USSPACECOM's



Source; D. J. Kessler presentation to USAF, Jan. 1987 (Ref. 1)

FIGURE 6-1. GEODSS CAPABILITY TO DETECT OBJECTS AT LOWER ALTITUDES

Perimeter Acquisition Radar Attack Characterization System (PARCS), which is sensitive to objects of about 1 cm in size, yields the debris population shown in Figures 6-2 and 6-4. The tracked population has increased steadily since the early 1970's, as shown by a comparison of the number of cataloged space objects between 1976 and 1986. During this period the tracked population has increased from 4100 to 4700 objects, compared with an increase of 25 percent in launch activity over the same period. This reflects the dynamic nature existing between new and decaying objects in space. (see Ch.7)

The 1986 Satellite Catalog (SATCAT) listed 16,660 entries, including all satellites launched in the last 30 years, their stages and trackable debris. However, only about 6000 of these objects are still in orbit, and about 44 percent of them originated from major on-orbit break ups (see Sec. 6.3.2).[4b, c]

Satellites are currently being launched into space at a rate of approximately 150-200 per year.[5] Eight countries presently possess space launch capability and over 100 nation-states participate in international satellite communication programs.[5-6] The rate of new objects cataloged is higher than the number of payloads because it includes debris. There were 983, 843 and 458 new objects cataloged during 1985, 1986 and 1987, respectively.

More than 3,600 payloads have been launched into space since 1957, but only 342 satellites were operational as of Sept., 1987, of which US operates 133, the USSR 148 and 13 other countries and international organizations, 61. Nearly half of this total are military satellites. By aggregate satellite mass, the Soviets account for 2/3 of the total.[4b, 33] The total mass now in Earth orbit exceeds 500 tons; each year about 800 additional tons are launched.[2] Active payloads comprise only 5 percent of all objects in space. The other 95 percent, including dead payloads, expendable launch stages and debris fragments are also monitored in case they pose re-entry hazards (Ch. 7). The mass/number balance of space objects decaying and re-entering Earth's atmosphere vs. those in long lived "deep space" orbits (periods longer than 225 min) and the projected annual influx of decaying space objects will also be discussed in Chapter 7.[2]

The orbital collision hazards are under active consideration by several national agencies (NASA, DOD, DOS, DOT, DOC) and international organizations. The "Unispace 82" conference acknowledged the growing threat to space activities posed by accidental collisions in orbit. The magnitude of the current and projected collision hazards for low-Earth orbit (LEO) and geosynchronous orbits (GEO) is shown in Figures 6-2 and 6-3.[1-3]

Several international agreements have been proposed, and are being considered to govern the orbital operation of satellites, disposal of inactive spacecraft and management of space debris. These agreements are limited primarily to the control of commercial communications satellites in geostationary orbits (GEO). Such agreements are motivated primarily by the need to prevent radio frequency interference between neighboring satellites, rather than to insure that collisions between satellites will not occur, given their relatively low spatial density. Depending on their orbital altitude and other parameters (inclination, eccentricity), mean orbital collision times for satellites range from

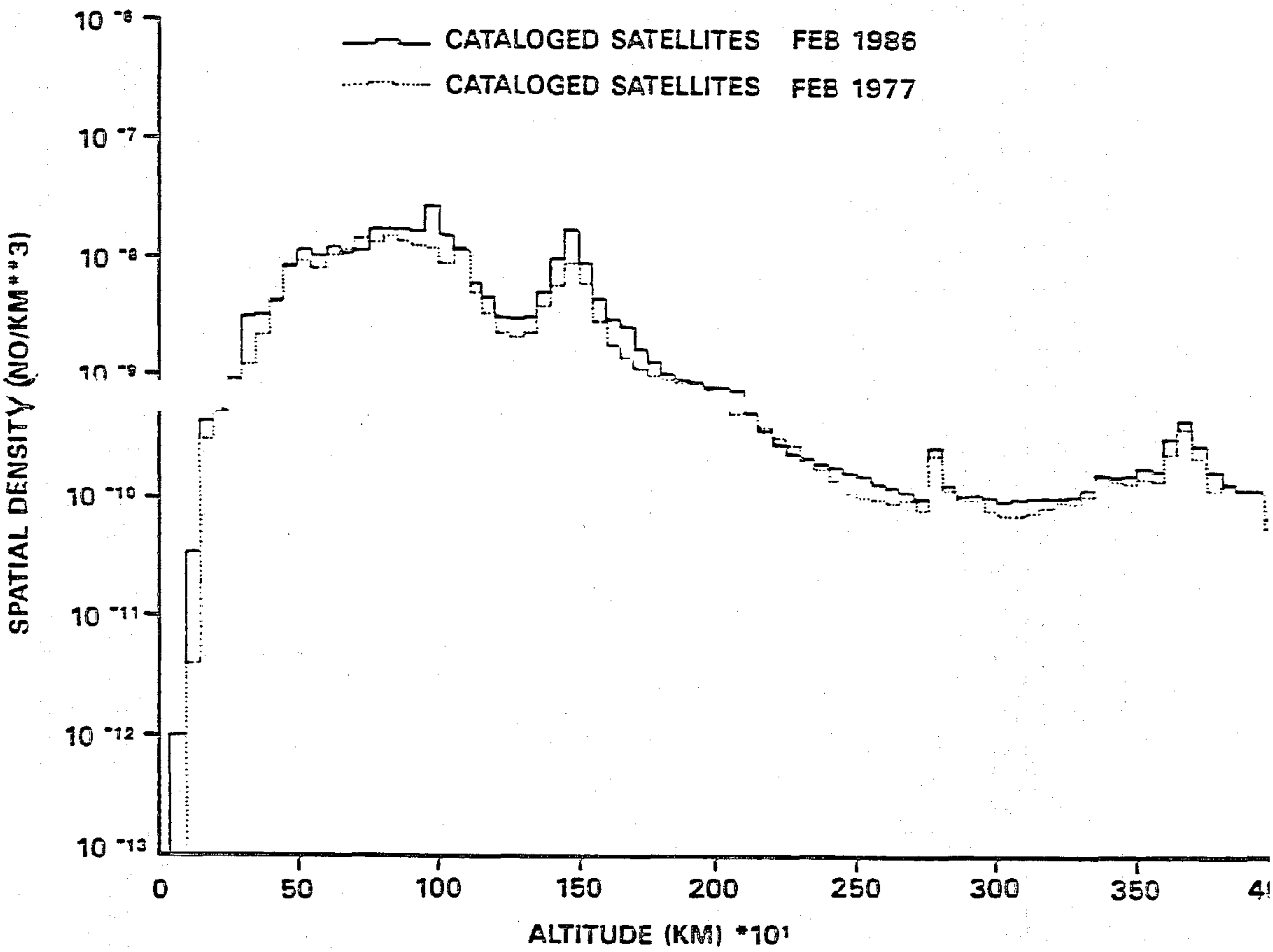


FIGURE 6-2. SPATIAL DENSITY OF SPACE OBJECTS VS. ALTITUDE (REF. 2)

a few years to as long as 1000 years. However, since the population of space objects is increasing rapidly in LEO and GEO orbits of interest, and since on-orbit debris increase even more rapidly, crisis proportions could be reached after the year 2000 unless debris management policies and procedures are adopted soon. Already, in 1979, the Japanese satellite ECS-1 was lost by a collision in space with the third stage of its own launch vehicle, causing a multimillion dollar loss.

Recent measurements and observations of satellite debris have indicated that the untracked man-made debris population in near-Earth and deep space orbits (of 1cm sizes in near-Earth and up to 20 cm in deep-space and GEO orbits) far exceeds the number of USSPACECOM-tracked fragments. These would augment the near-Earth amount of tracked debris by a factor of 10 and the debris orbiting in deep space by 25-50 percent. The collision hazards increase proportionately.[23] (see Secs 6.2 and 6.3) Although the tracked population of debris is increasing linearly (by ~ 250-300 objects per year), not exponentially as previously predicted, it already has exceeded the natural meteoroid background (Fig. 6-4).[1-3] Untracked smaller debris appear to dominate collision encounters. Little data on the man-made debris flux are available on debris less than 4 centimeters in size (Fig.6-1).

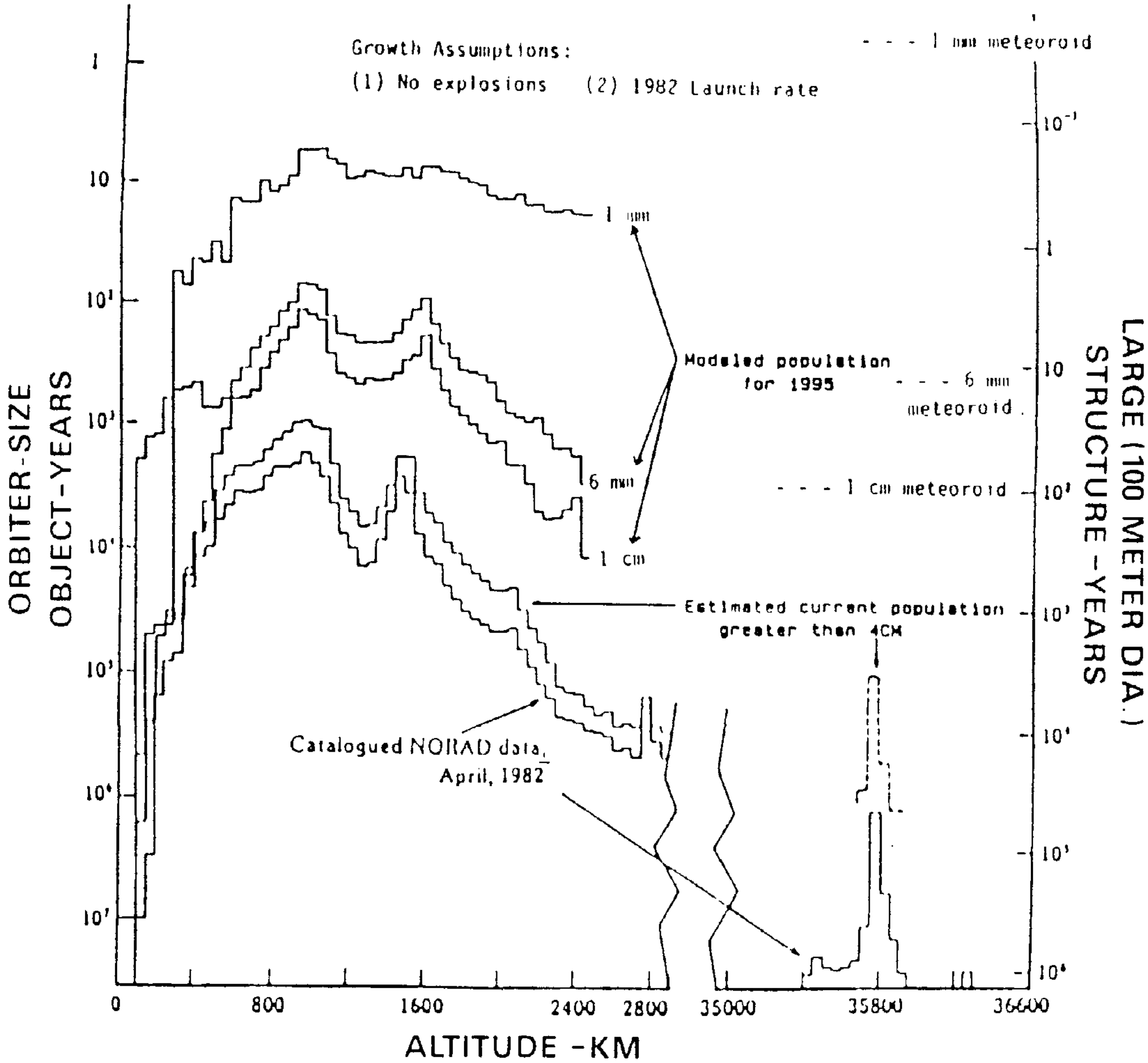


FIGURE 6-3. AVERAGE TIME BETWEEN COLLISIONS, MODELED POPULATION FOR 1995

'Objects below this size cannot be detected by Space Command's deep space tracking detection systems. GEODDS (The Ground-Based Electro-Optical Deep Space Surveillance System) however, is an expanding global network of tracking sensors which is continually being upgraded to aid in monitoring space assets.(13)

Space hazards of interest to this analysis include:

- Low Earth Orbit (LEO) Collisions (Secs. 6.4.1 and 6.4.2):

  - Collisions between two active spacecraft in LEO between 200 km and 4000 km (120 miles and 2400 miles).

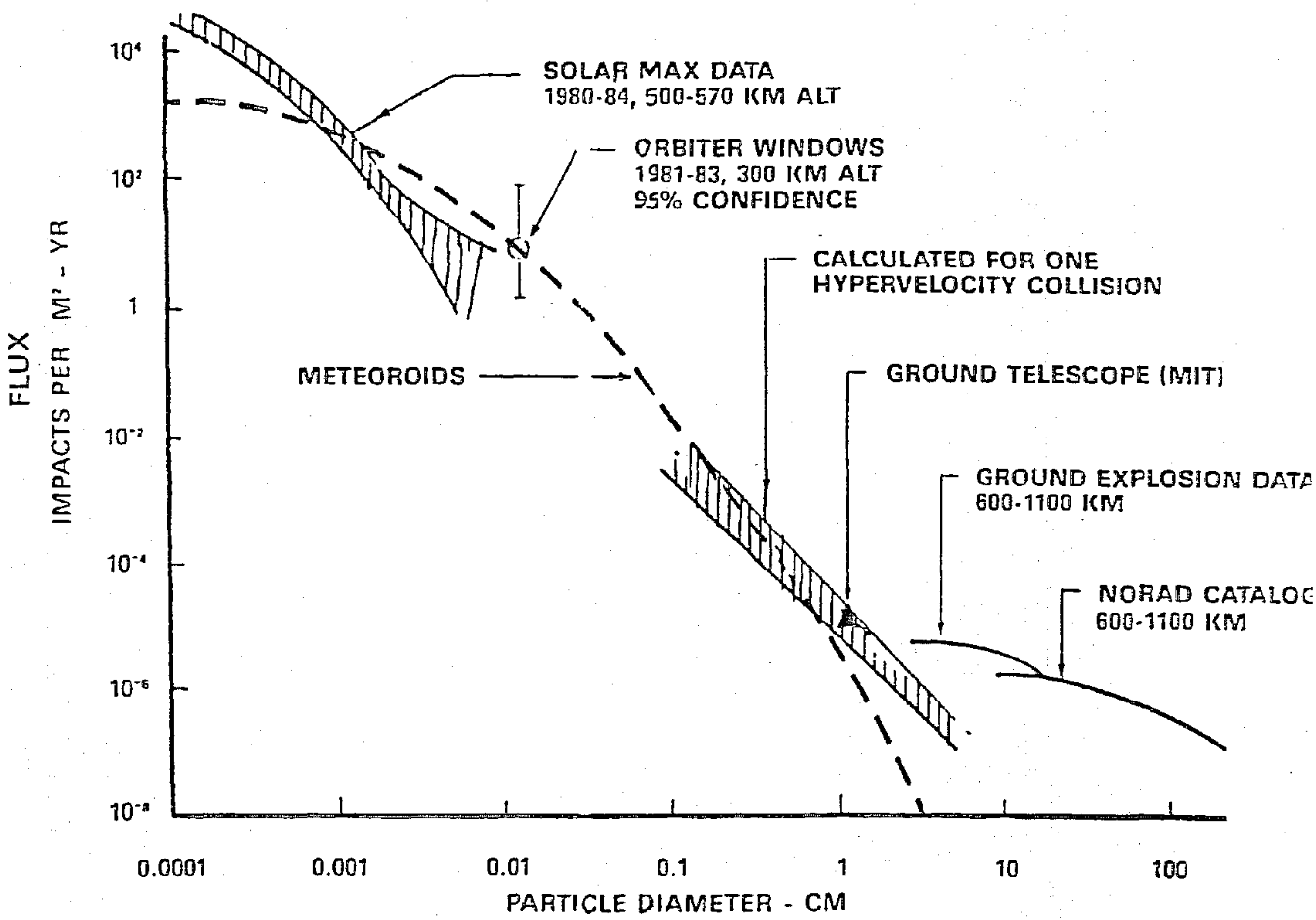


FIGURE 6-4. RECENT ORBITAL DEBRIS MEASUREMENTS COMPARED TO METEOROID FLUX (Refs. 1, 3)

- Collisions with both man-made and natural (meteoroids) objects in the near-Earth orbits. The hazard from man-made debris increases with time while the debris of the natural environment remains at a near constant level (Figures 6-2, 6-4).

- Geosynchronous Earth Orbit (GEO) Collisions (Secs. 6.4.3 to 6.4.5):

- Collisions between active spacecraft and inactive spacecraft remaining in a geosynchronous orbit. This GEO "ring" is narrow in latitude and altitude bands, but spread over 360° in longitude (Fig. 6-5). The collisions may result from the accumulation of inactive spacecraft in the most desirable GEO orbits for communication satellites.

- Collisions between two active spacecraft in geostationary orbit. These collisions can be prevented if collision avoidance procedures are invoked by ground control or by judicious orbital slot allocation.

- Collisions between active spacecraft and spent orbital transfer stages in GTO or other debris in GTO and GEO. The probability of collision with objects in geo-transfer orbit (GTO) is

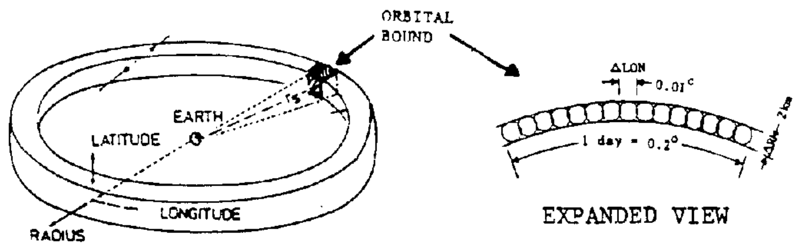


FIGURE 6-5. GEOSTATIONARY RING SHOWING SATELLITE ORBITAL BOUNDS

relatively small due to the short dwell and transit time of geo-transfer objects in the geosynchronous band (about 3% of their period).

When considering objects large enough to damage most spacecraft, artificial debris, whose sources are discussed in Sec. 6.3, constitute the dominant threat.[2,3] Collisions involving artificial and meteoritic debris possess these differing characteristics:

1) Collision hazards are proportional to the debris population densities, relative orbital velocities between colliding objects and the cross sectional area of the orbiting spacecraft.

2) Large debris consist primarily of artificial objects, while small debris are dominated by natural meteoroids.

3) Meteoritic debris remain at a relatively constant level, while the spatial density of man-made debris is increasing with time.

4) Artificial debris populate circular orbits with rather low relative velocities, while meteoritic debris orbits are elliptical with larger relative velocities at collision. The average velocity of meteorites relative to spacecraft is roughly twice as large as that of man-made objects, namely 14 km/s vs. 7 km/sec. However, cometary debris move in elliptical and sometimes retrograde orbits and can therefore reach 40-70 km/sec. relative impact velocities.

6.2 SPACE LAW AND SPACE DEBRIS ISSUES

6.2.1 The Regulatory Framework for Orbit Allocation and Space Debris

Major international agencies that establish and implement space law, as it applies to communication and remote sensing satellites, include:

- United Nations Committee on the Peaceful Uses of Outer Space (COPUOS)
- International Telecommunication Union (ITU)
- International Telecommunications Satellite Organization (INTELSAT)
- International Maritime Satellite Organization (INMARSAT)

COPUOS is the foremost entity of these agencies since the major space treaties in effect today have been negotiated under its auspices. The ITU is the principal agency that deals with regulatory matters pertaining to satellite communications. It receives support from several other organizations, namely:

- The International Radio Consultative Committee (IRCC)
- The International Frequency Registration Board (FRB)
- The International Telecommunications Satellite Organization (INTELSAT)

Of these organizations, the IRCC is the most likely to be involved with the problem of satellite collisions. Specific groups have been established within the IRCC to study special subjects, primarily in the areas of space communications and interference problems. INTELSAT is dedicated to the construction, deployment and operation of commercial telecommunication satellites.

A majority of nation-states must first endorse international treaties and regulations, in order for them to become effective. The implementation of such treaties requires all member states to abide by the dictates of the majority. Therefore, any proposal pertaining to on-orbit collision risk reduction and orbital debris management would require several years for discussion, consideration and ratification in an international forum.

Presently, only communication satellites are assigned orbital and frequency windows through international agreements. Other commercial, research and military missions go through a process of orbital parameter optimization prior to mission approval to avoid collisions during their useful life. These are simply registered with the UN by the launching state. USSPACECOM can identify space object fragmentation events and infer their probable cause: for example, if orbiting satellites cross in space and time disappear and the crossover point becomes strewn with debris, a mutual collision can be inferred. It is difficult to assign liability and to determine whether a collision encounter on-orbit was accidental or intentional. The National Ranges, as well as NASA and the Satellite Surveillance Center (SSC) within USSPACECOM, usually perform COLA (COLlision Avoidance at launch) to determine safe launch windows and COMBO (COmputation of Miss Between Orbits) screening runs for proposed missions to check the proposed orbits against cataloged orbits. A "point of closest approach" (PCA) is computed. If a risk exists, orbital maneuvering capability or orbital parameter changes are provided. Hence, preplanning of missions avoids collisions with known and tracked space objects. While COLA is run routinely prior to launch, COMBO runs are complex and costly, so that orbital safety screening has been done only for select US Government missions. Smaller debris which cannot be radar tracked pose unpredictable hazards. "Rules of the road" for satellite close approaches are currently being considered to avoid international conflicts in space.[28-30]

6.2.2 Orbital Debris Issues

An assessment of collision hazards in space requires a study of collision probabilities between all objects in space including those of natural origin (i.e., meteoroids) as well as man-made objects (satellite and space debris). Orbital debris consist of: spent spacecraft, used rocket stages, separation devices, shrouds and fragments from accidental or deliberate explosions and collisions.[1-3] A major concern for future space activities is the possibility of generating a debris belt as a result of cumulative collisions between orbiting objects.[1-14] Several models, discussed below, have been developed to estimate quantitative collision hazards for spacecraft in both low earth orbit (LEO) and geostationary orbit (GEO) regimes.[15-20] Each of these models relates the collision hazard to the orbital population density and to the relative object velocity. Estimates of collision probabilities between spacecraft and debris in LEO and GEO show that, at present, this hazard is still small (1 in 1000 and 1 in 100,000 per year in orbit, respectively), but increasing rapidly (Figs. 6-2, 6-3). The threat of losing on-orbit satellites through collisions with other inactive satellites or orbiting debris is not yet critical, but is becoming increasingly serious. The more crowded regions of space which are optimal for man-rated systems (like the Space Station), larger satellites or those used for communications, remote sensing, navigation and surveillance missions are of most concern.

Proposed space debris management options under consideration include the following:[4,13,24,31]

- provide impact hardened shielding to new satellites, as well as added orbital maneuvering capability to avoid collisions;
- require that extra fuel be provided to satellites inserted into more crowded space orbits to enable their transfer into either higher and longer lived "parking" orbits, or into lower decaying "disposal" orbits at the end of their life. International cooperation and agreement is needed to define such parking and disposal orbits;
- undertake "space salvaging" operations to retrieve and remove dead payloads from more crowded orbits. This "celestial trash can" could be ejected from the Solar System, injected into a Sun bound orbit or fitted with rockets for controlled re-entry to Earth. The latter would allow "disposal" by atmospheric burn-up, but would increase re-entry hazards (Ch. 7).

6.3 ORIGIN OF ORBITING DEBRIS

6.3.1 Hypervelocity Collisions

Hypervelocity collisions in orbit can generate a significant number of debris particles which are too small to be observed, yet sufficiently large to inflict damage to any unhardened spacecraft. Uncertainty about the population of unobserved debris particles is the most important factor limiting an accurate assessment of space collision hazards (Figures 6-3, 6-4). Ground based tests of hypervelocity impacts indicate that a single high speed collision in space could produce between 10,000 and 1,000,000 pieces of debris. Table 6-1 provides estimates of the number of debris objects which could result from collisions between different size objects [7]. Verification of the results of high speed collisions in space is hampered by the difficulty in observing the small particles. Given the present tracking capability, it is difficult to differentiate between a fragmentation caused by a hypervelocity collision or an explosion.[4] There have been no confirmed instances of satellite damage due to high speed collisions with debris in space to date.[4] The subject of collision by-products is closely tied to the generation of the so-called "debris belt" which could result from cumulative collisions. While such a catastrophe would cause severe problems for future space ventures, it is not considered a likely consequence for many years to come.

TABLE 6-1. FRAGMENTS GENERATED IN HYPERVELOCITY COLLISIONS[7]

Colliding Objects	Debris Generated K	G	M
K/K	100	4000	40,000
K/G	-	50	2,000
K/M	-	-	50
G/G	-	50	4,000
G/M	-	-	50
M/M	-	-	50

K: Objects larger than 1 kilogram
G: Objects in the gram to kilogram size range
M: Objects in the milligram to gram size range

6.3.2 Explosions and Spacecraft Breakups

Explosions and breakups of spent propulsion stages and spacecraft on-orbit (either spontaneous or collisional) are a major source of space debris (Figs. 6-6, 6-7 and 6-8). More than 90 known break ups have occurred in orbit, as of January, 1986.[2,3,7,13,14,21,22] For the 39 satellites known to have fragmented in orbit, 15% of the events are propulsion related, 40% were deliberate and the rest are

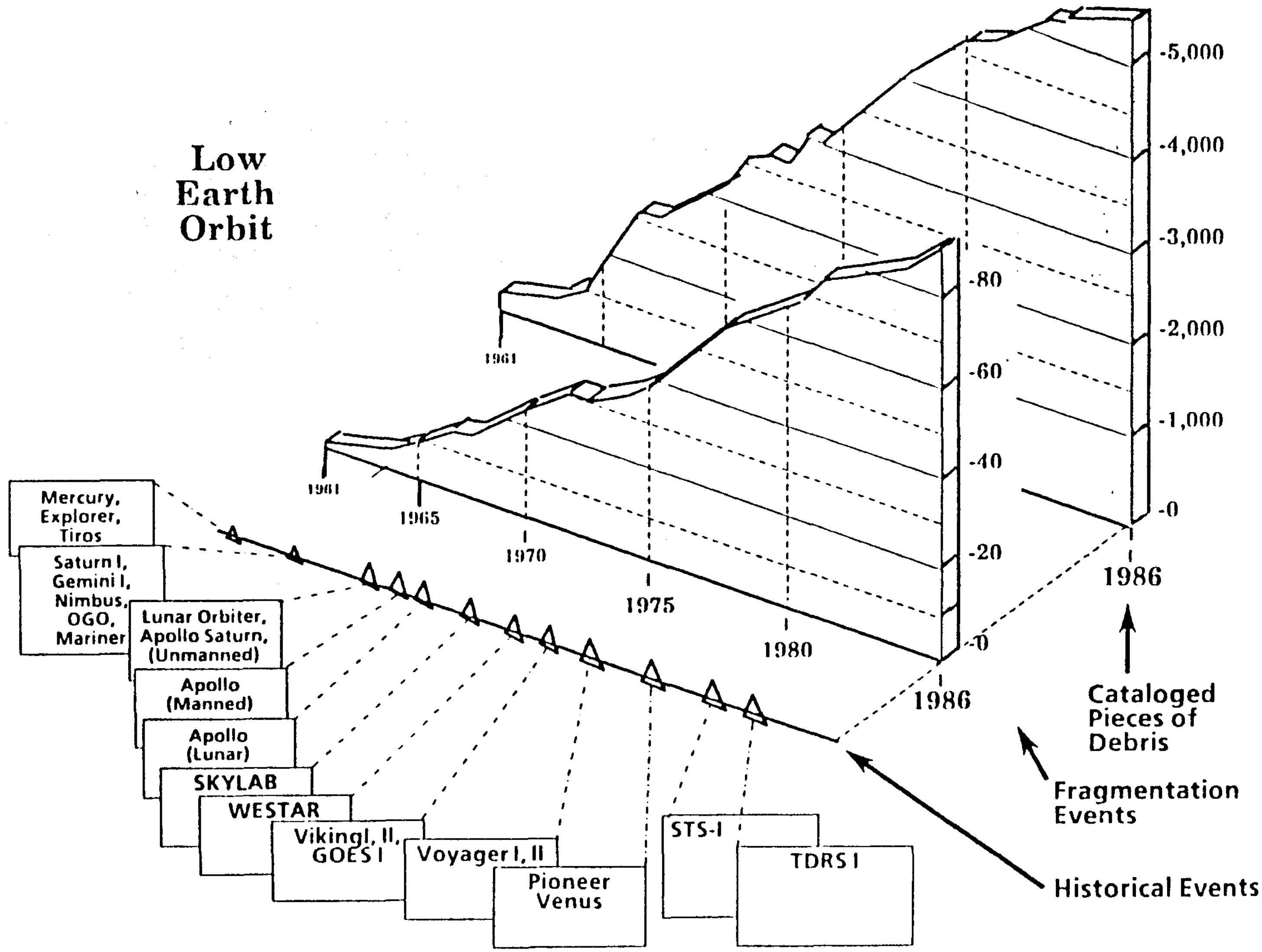


FIGURE 6-6. HISTORY OF ON-ORBIT FRAGMENTATIONS (AS OF 1 JANUARY 1986)

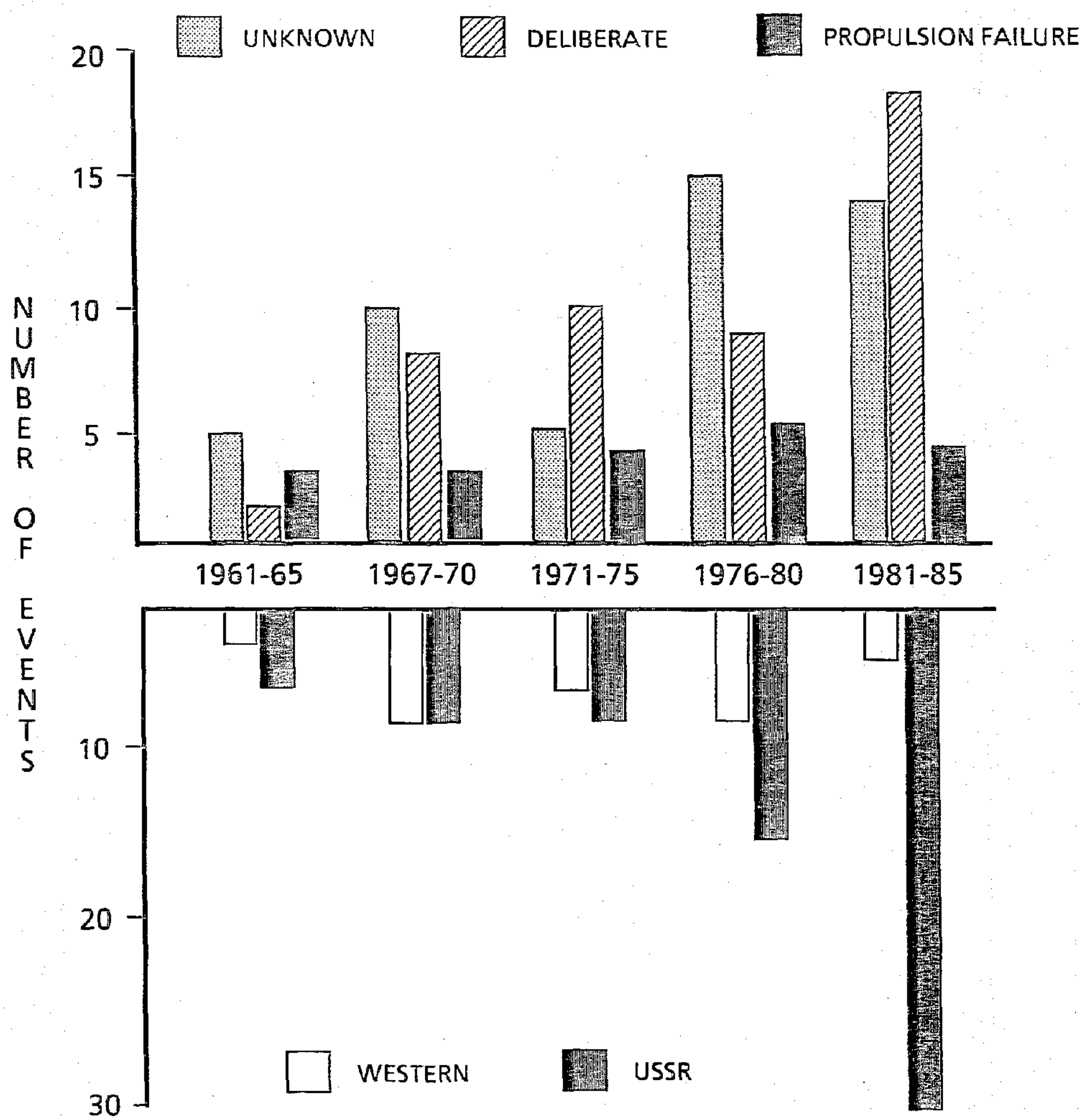


FIGURE 6-7. HISTORY OF SATELLITE BREAK UP EVENTS (REF. 2)

due to unknown causes. Explosions, both inadvertent and intentional, represent the largest single source of space debris and account for approximately 60 percent of the tracked space objects. These are almost equally divided among non-operational payloads and remaining mission related expendable objects, such as rocket stages, shrouds, etc. Debris originating in one collision or explosion event will cluster in orbital parameters (inclination, eccentricity) so that locally, the probability of impacting an orbiter is much higher (Fig. 6-8).

As of July 1982, 49 percent of the cataloged population had originated from a total of 44 break ups. In November 1986, an Ariane 3rd stage, launched nine months earlier, exploded and created a cloud

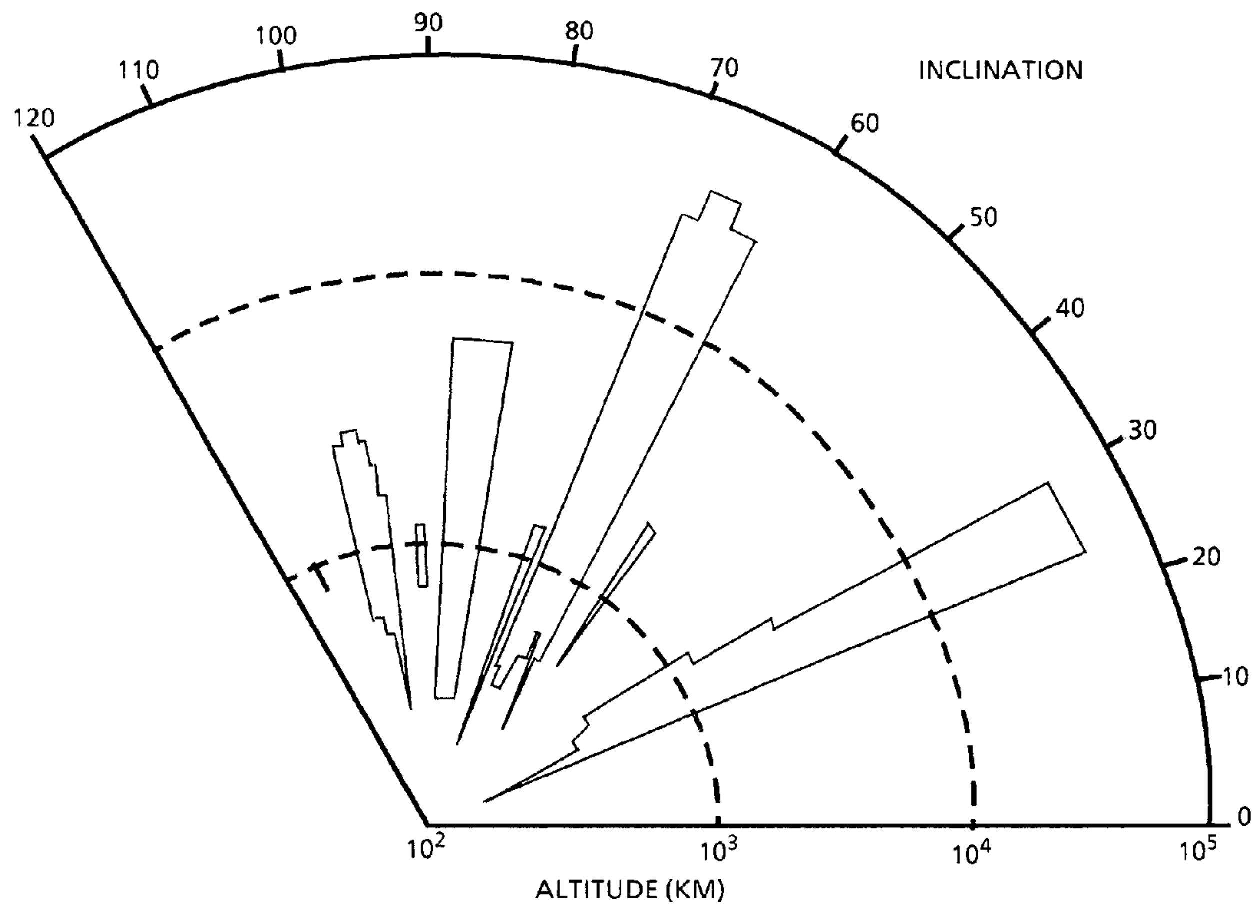


FIGURE 6-8. SATELLITE BREAK UP DEBRIS REGIONS

of debris in polar orbit, centered at 490 mi. altitude, but spread as low as 270 mi. and as high as 840 mi. Ariane 3rd stages are known to have exploded on orbit at least 3 times before this, as indicated by SPACECOM tracking data. On orbit explosions also have been associated with second and upper stages along with casings from Proton, Ariane, Delta, Titan, Atlas and Atlas/Centaur spent stages. There have been ten Delta 2nd stage explosions in orbit prior to 1981, but none since 1982 (see below).

The increase in LEO hazard level caused by the explosions of several US ELV second stages in the early 80's (see Sec. 6-2) is less pronounced at elevations of 600 to 1200 km than in the 300 km range because the relative debris level is lower at these altitudes. It is estimated that for an explosion which produces 500 fragments, the time between collisions involving one of these fragments would be about 50,000 years.

Since 1986 steps have been taken to stop such explosions by venting all residual cryogenic fuel in jettisoned 2nd and 3rd stages (i.e., fuel depletion burn). This residual fuel tended to explode upon thermal cycling and overpressurization due to solar heating, especially for sun-synchronous orbits. A recent change in operating procedures requires residual liquid fuel of spent second stages (and upper stages, if liquid fueled) to be vented to prevent and control on-orbit explosion generated debris. However, Ariane upper and transfer stages have exploded on-orbit as recently as 1986 and 1987 since ESA has yet to adopt a venting policy.

Ground simulated Atlas explosions, used as calibrations tests for fragmentation, produced about 1300 fragments. On September 20, 1987, the Soviet satellite Cosmos 1769 (suspected to be nuclear powered) was intentionally destroyed on-orbit producing a cloud of debris at about 210 mi. altitude and 65° orbital inclination. Reference 25 lists past satellite breakups and the number of cataloged objects generated by the breakups. Extrapolating the number of on-orbit explosions and break ups, the SPACECOM catalog could expand by up to a factor of 10 in the next 20 years.

6.3.3 Orbiting Nuclear Payloads

Special on-orbit hazards are posed by the increasing number of nuclear power sources, both active reactors and passive fuel cells.[13,24] Therefore, approval of nuclear missions is subject to more rigorous risk assessments, planning and review by an Interagency Nuclear Safety Review Panel (INSRP). There are about 50 potentially hazardous satellites in orbit today, carrying over 1.3 tons of nuclear fuel, much in the form of long life toxic isotopes. These pose both on-orbit collision and re-entry hazards (see Ch. 7). The 48 radio-thermal generators (RTG) and fuel cores orbiting today are in the most crowded LEO region at about 1000 km altitude. Both US and Soviet satellites have exploded or spawned debris in this belt. However, since 90% of the Soviet nuclear material in RORSAT satellites has been intentionally ejected into higher orbits at 900-1000 km at $\sim 65^{\circ}$ inclination, the hazards to population due to re-entry or possible ground impact have been removed. This procedure is intended to increase the orbital lifetime to more than 1,000 years to allow sufficient time for the radioactivity to decay. The eventual retrieval and elimination of these materials is possible by sending them, for example, into escape orbits or into the Sun. Hypervelocity collisions with nuclear satellites and their fragments could endanger, contaminate and disable both manned and unmanned spacecraft with perigees well below 1000 km.

6.4 ASSESSMENT OF COLLISION HAZARDS IN ORBIT

6.4.1 Collision Hazard in LEO

Low Earth Orbits generally include the altitude range of 200 km to 4000 km. This region has the largest spatial density (Number/km^3 - see Fig.6-1) of space objects, with a maximum of $\sim 1.7 \times 10^{-8}$ objects/km^3 between 800 and 850 km and $\sim 2.5 \times 10^{-8}$ objects/km^3 between 950-1000 km altitude. This corresponds to a mean time between collisions of 1/1800 years for a satellite with a cross section of 100 m^2, the size of the Soviet Mir Space Station (Fig. 6-3). Figure 6-9 shows the observed population of satellites, as modified by the debris density. This density exhibits two maxima, one near 800 km (480 miles) altitude and the other near 1400 km (840 miles). The actual debris population is likely to be considerably larger than that shown in Figures 6-6, 6-8 and 6-9. Decay of space objects, i.e., re-entry to Earth, occurs primarily from low altitude orbits and results from atmospheric drag which increases with the level of solar activity. A typical orbital lifetime at 300 km is less than one month; below 200 km, it is just a few days. These de-orbiting spacecraft will re-enter Earth's atmosphere and contribute to re-entry hazards (see Ch. 7).

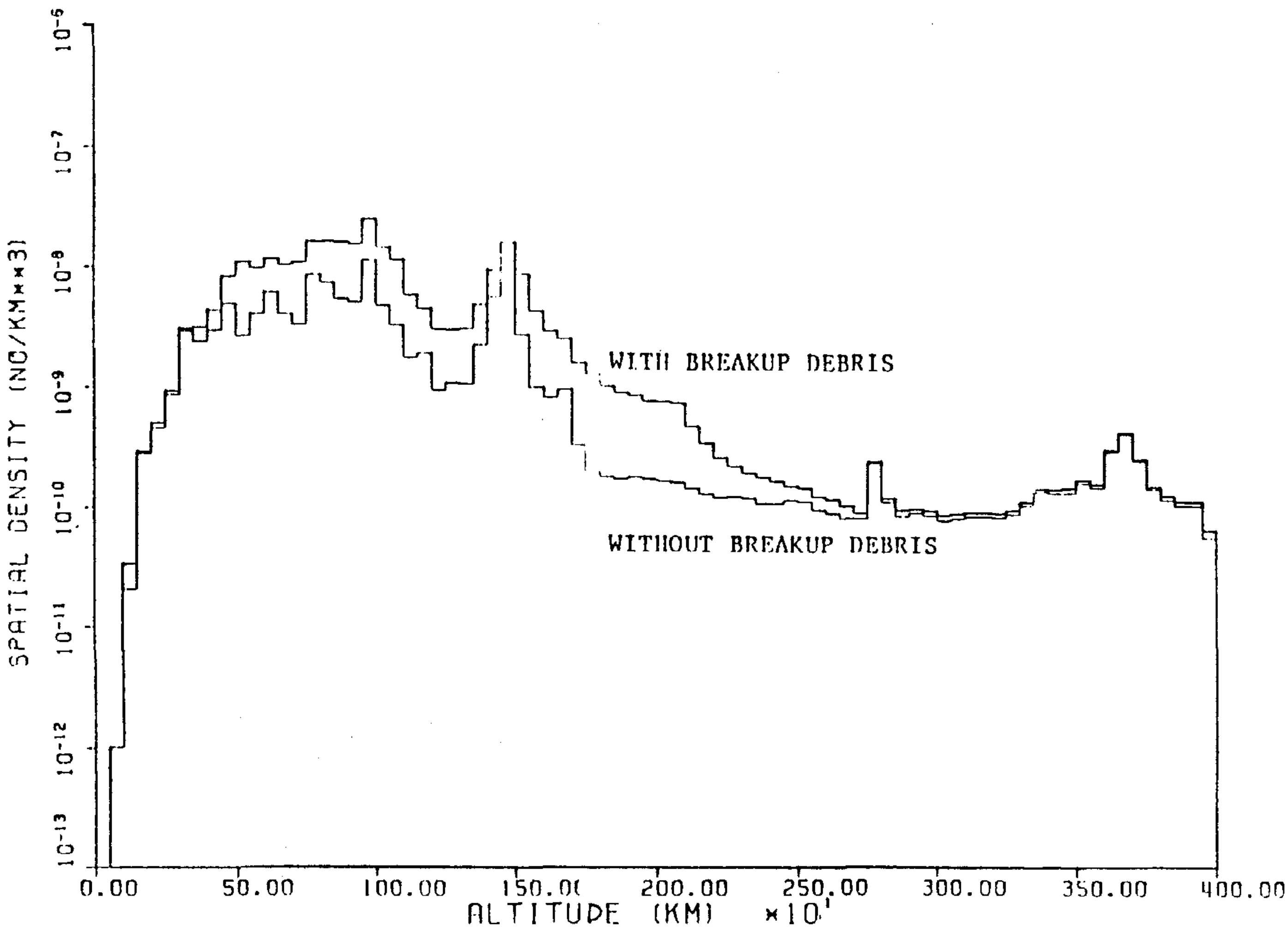


FIGURE 6-9. CONTRIBUTIONS OF BREAK UP DEBRIS TO LEO SPATIAL DENSITY

If the worldwide satellite population continues to increase at ~150-180 /year (as was the case for the past 5 years)[5] and all these objects penetrated the maximum density altitude band (950-1000 km), the LEO spatial density would still not be expected to increase by a factor of 10 until between the years 2044 and 2100.

Many Earth satellites (~83%) which reside in LEO decay in orbit within a few days to several years. Solar flare and sunspot activity cycles periodically "purge" these orbits (see Refs. 13,29 and Chs. 4,7).

Inactive satellites, jettisoned rocket motors and launch or break up debris in LEO could undergo hypervelocity impacts (at ~10km/second) with active satellites in circular orbits and with others in elliptical orbits which traverse this altitude range.

Launch activity is an important factor contributing to space hazards through the generation of man-made debris. Table 6-2 shows the number of space launches since 1980 and the projected number of space launches anticipated in the next decade.[5,6,9] The current annual USSR space activity amounts to about 105 launches per year. The Soviet program accounts for roughly 95 % of the total, largely because the useful on-orbit life of Soviet satellites is much shorter than that of equivalent US spacecraft.

TABLE 6-2. YEARLY LEO LAUNCH ACTIVITY

	YEAR															
	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95
US	13	17	17	23	17	3	0	21	24	15	11	12	15	13	14	15
USSR	89	98	101	95	92	108	108	108	111	114	115	114	114	114	114	114
Other	3	8	3	9	3	7	5	5	6	5	5	22	17	21	18	14
Total	105	123	121	127	112	118	123	134	141	134	131	148	146	148	146	143

Figure 6-2 shows the relative flux distribution of meteorites and man-made objects in LEO. The meteorite flux data were based on indirect ground based measurements, including observation of meteors burning up in the atmosphere. The man-made flux data were taken from the 1986 Satellite Catalog of tracked space debris.

6.4.2 Collision Probabilities in LEO

Collision probabilities are useful in assessing space hazards, estimating collision hazards between operational spacecraft and orbiting objects quantitatively and determining the likelihood of satellite debris collisions.

Models developed for deriving probability estimates usually use the following assumptions:

- Objects in orbit are randomly distributed and each object is assigned an effective cross section.
- The collision cross section is usually the geometric cross section of the satellite.
- Orbital planes within the debris population have random distributions in the azimuthal coordinate.

Several models based on kinetic theory and celestial mechanics provide estimates of collision hazards to operational spacecraft in LEO.[11,16,20] The impact probability, per orbit or per crossing a certain orbital torus, must be multiplied by the on-orbit satellite lifetime (or the mission duration) and the cross section of the object to estimate its overall collision risk.

Probability derivations are simplified if the object density is assumed to have only an altitude dependence and all other dependencies are replaced by averages. While the latter removes the possibility of including angular orbital dependencies in the solution, it nevertheless provides a reasonably accurate estimate of the collision hazard.

One procedure used to determine the altitude dependent object distribution is to define an Earth centered spherical grid, consisting of surfaces of constant radius spaced every 50 km from 150 to 4000 km in altitude, and surfaces of constant polar angle (latitude) spaced every 5 degrees.[8] The object density within the above defined space cells is computed based on the percentage of time an object spends in the 'spherical cell.' Figure 6-2 is typical of the type of density distribution which results from this model. The mean rate of collision probability, P, is defined as,

$$P = \int_{o}^{to} C(r,t)\, dt$$

where C(r,t) is the collision frequency equal to,

$$C(r,t) = \sigma_{eff} \bullet \rho(r,t) \bullet v(r,t)$$

Where: ρ = object density

σ_{eff} = effective cross section

v = mean speed of object relative to debris

r = object distance from Earth's center

t_o = the elapsed time.

Applying this to the example of the Shuttle Orbiter at 300 km altitude, with a debris distribution similar to that shown in Figure 6-2, gives a predicted time between collisions approximately equal to 25,000 years[8]. These models estimate the collision probability for a Shuttle Orbiter at 150-300 km altitude to be roughly 1 in 25,000 years. The chance of an orbiter colliding with debris in LEO, over its lifetime, is about 10^{-3} at present and may exceed 10^{-2} by the year 2000. The larger collision risk for spacecraft which operate in the 600 to 1200 km range of maximum debris population, is offset by the smaller cross sections of operational spacecraft at these altitudes. This result assumes a typical Shuttle cross sectional area of 250 m^2 and a relative impact velocity of 7 km/s. Man-made debris of size 4 cm and smaller do not present a significant hazard to LEO spacecraft with dimensions comparable to that of the Shuttle. A future Space Station 100 m across in LEO at a 500-550 km altitude, would have a mean life to collision of ~170 years without debris, but of only ~41 years given the present debris strewn near-Earth environment.

Inclusion of the latitude dependence in the probability estimate yields similar results. Table 6-3 gives the predicted time between collision as a function of orbital inclination with the same LEO debris population used previously (see also Fig. 6-8).

TABLE 6-3. COLLISION TIMES FOR A SHUTTLE ORBITER WITH LEO DEBRIS[8]

Shuttle Orbit Inclination Angle (deg)	Time between Collisions (years)
28.5	2.7×10^4
56	2.0×10^4
82	1.6×10^4
90	1.5×10^4
98	1.4×10^4

Greater debris hazards are anticipated for spacecraft operating at higher altitudes, particularly in the range from 600 to 1200 km where debris density is greatest (Fig.6-2). Table 6-4 gives the estimated time between collisions for a small spacecraft, of 5 m^2 collision cross section, with man-made debris assuming a relative speed of 7 km/s. There is evidence that some spacecraft in LEO have already collided with either natural or artificial orbiting debris.

TABLE 6-4. TIME BETWEEN ON-ORBIT COLLISIONS VERSUS LEO ALTITUDE[8]

Orbit Altitude (km)	Collision Time (years)
648	1.8×10^5
741	5.3×10^5
833	4.8×10^5
926	6.1×10^5
1019	7.5×10^5
1111	1.5×10^5
1204	3.5×10^5

6.4.3 Collision Hazard in Geosynchronous Orbit (GEO)

Conceptually, the geosynchronous orbits can be visualized as a spherical shell several kilometers thick located at an altitude approximately 36,000 km above the Earth. Spacecraft in geosynchronous orbit move with the rotating Earth at arbitrary angles of inclination with respect to the equator. The geostationary orbit represents a particular subclass of the geosynchronous orbits in which objects move synchronously with the rotating Earth, but with positions fixed relative to its rotating coordinate system. The geostationary ring denotes a particular region in geosynchronous space, of approximately several hundred kilometers in width, encompassing these orbits.

The main characteristics of geosynchronous orbits are:

- Orbital period is equal to one sidereal day (1436.2 minutes or 24 hours).
- An infinite variety of orbits exist each with the same average altitude as a geostationary orbit.
- Objects in orbit cross the equator twice each day with average velocity of 3075 m/s.
- The equatorial crossing point of the object drifts cyclically along the equator due to unbalanced Earth gravity.
- Objects remain permanently in orbit (as in the geostationary ring).

The main characteristics of geostationary orbits are:

- Altitude above Earth is 35,787 km (19323 nautical miles) ± 50 km.
- Orbit is exactly circular over the Earth's equator (± 1^0 latitude).
- Orbital period is 1436.2 minutes or roughly 24 hours.
- Objects in orbit have an orbital velocity of 3075 m/s.
- Objects remain permanently in orbit, i.e., the decay rate is very slow and secular, about 1 kilometer per thousand years.
- Objects in orbit are subject to weak luni-solar and Earth gravitational perturbations which result in slow drift in east-west and north-south directions about the two geo-stable points at 75.3°E and 104.7°W longitude. This results in eventual clustering of inactive satellites in these regions.

Semi-geosynchronous orbits (i.e., at half the GEO altitude with 12 hour periods) are also used for communication satellites. Such highly elliptical "molnyia" (lightning) orbits are favored by the Soviets because the satellite spends most of its time above the Soviet Union moving slowly near apogee, but crosses rapidly over antipodal regions near perigee. Such orbits degrade more rapidly due to atmospheric friction near perigee.

The largest concentration of operational spacecraft lies in the geostationary belt and currently numbers over a hundred spacecraft. Extinct satellites also continue to orbit in the crowded GEO orbits, presenting a mounting collision damage hazard to new communication satellites (Fig. 6-3). Some nations and organizations have begun to move inactive satellites out of GEO to prevent cluttering of the GEO ring. However, according to Ref. 3 (Ch. 4), the removal of inactive satellites from GEO stations at the end of their useful life is not yet a general practice. The policy of using disposal orbits for defunct satellites has recognized shortcomings which may introduce new hazards to active payloads (e.g., the potential for misfire or explosion, eventual migration of "removed" payloads to GEO due to luni-solar perturbations and solar wind pressure, added cost for stationkeeping and orbital maneuvering propellants and decreasing reliability with life on-orbit.)

The peak spatial density (number per km^3) of satellites at GEO altitudes (35,750 to 35,800 km) is due to about 543 satellites, of which only about 150 are geostationary. The others are in either geosynchronous, or semi-geosynchronous highly elliptical "molnyia" orbits. The corresponding spatial density value is $\sim 7.55 \times 10^{-10}$ objects, still 2-3 orders of magnitude below that in LEO.

The current geosynchronous population, as tracked by USSPACECOM, consists of about 116 active communication satellites plus at least as many uncontrolled objects drifting through the geosynchronous corridor. The latter includes inactive satellites and debris which drift around the Earth or oscillate about the two geo-potential stable points. USSPACECOM can track an object of the size of a soccer ball in GEO and of about ≥ 10 cm. in LEO (Figs. 6-1, 6-4). Figure 6-10 shows the relative positions of the commercial communication satellites in GEO. The number of active GEO satellites over the past few years and the estimated number of GEO launches in the coming decade is shown in Fig. 6-10 and Table 6-5.[5,6]

TABLE 6-5. GEO LAUNCH ACTIVITY

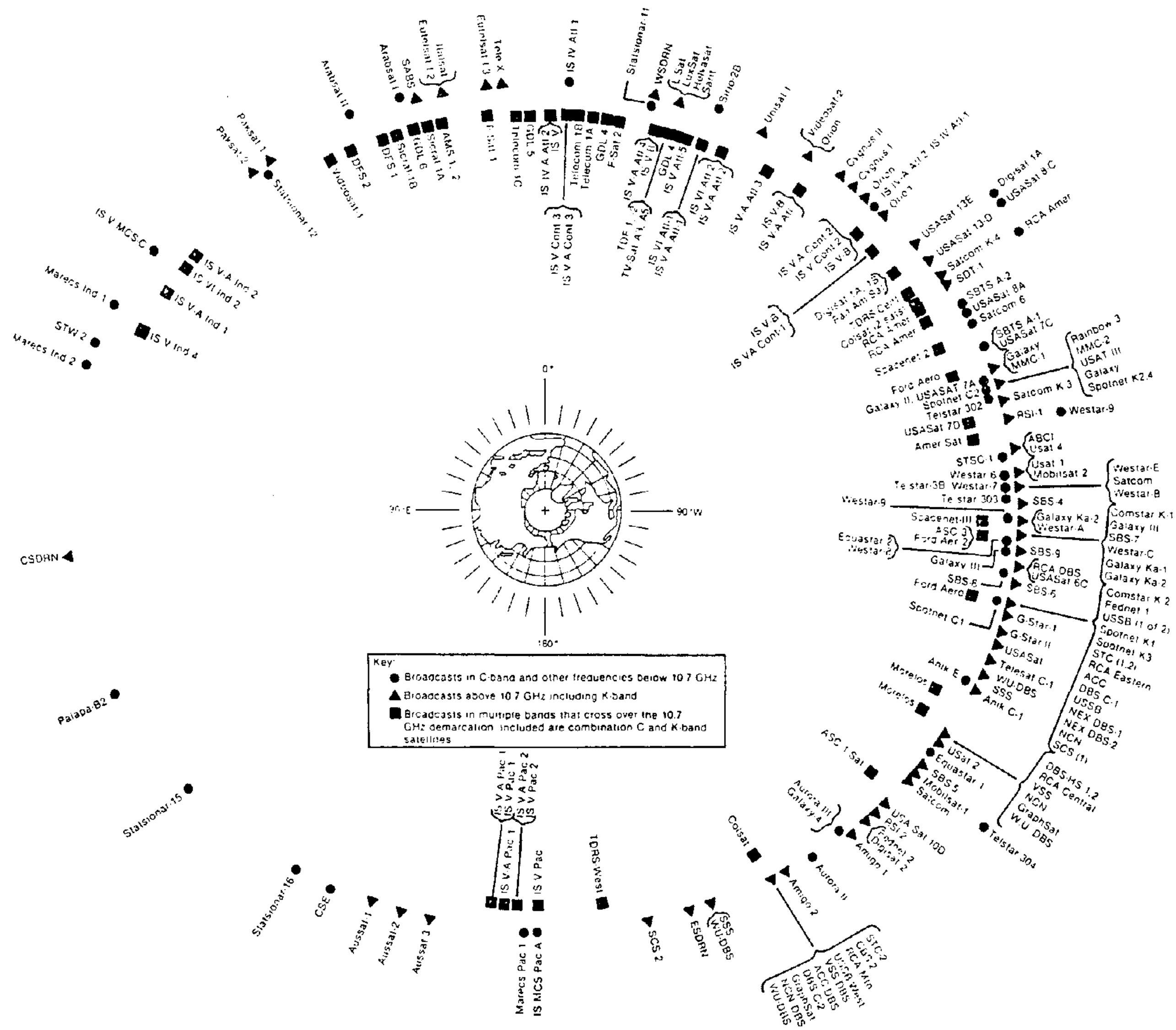
	Year										
	85	86	87	88	89	90	91	92	93	94	95
US	16	14	13	15	16	14	17	14	19	20	27
USSR	5	5	5	5	5	7	7	7	7	7	7
Other	6	9	7	4	3	8	3	1	6	12	9
Total	27	28	25	24	24	29	27	22	32	39	43

Thus, collisions in GEO are restricted to object encounters at a fixed altitude of approximately 36,000 km, actually an equatorial torus of 1^0 in latitude and 35, 785 ± 50 km altitude above the Earth's equator. Such collisions can involve both man-made objects and natural objects (meteoroids).

Estimated collision probabilities with debris in GEO are of the order of 10^{-5} at present, but could reach 2×10^{-3} over the life of the satellite, (i.e. 1 in 500) by the year 2000. Therefore, at current GEO population levels, collision hazards do not appear to be a major problem.[1-4,9,17] The collision hazards in GEO tend to be lower than in LEO for the following reasons:

(1) the lower spatial density of GEO satellites, although new communication satellites are increasingly crowding GEO orbits(Fig. 6-2);

(2) the relative velocity difference between objects orbiting in GEO is less than for LEO;



NOTE:
1. IS refers to INTELSAT satellites.
2. The following are USSR satellite series: CSDRN, Ekran, EŞDRN, Gorizont, Loutch, Potok, Prognoz, Raduga, Statsionar, Volna, WSDRN.
3. Multiple orbits are for legibility only. All geosynchronous satellites orbit at approximately 22,300 miles above the earth.

FIGURE 6-10. PLANNED LOCATIONS OF COMMERCIAL COMMUNICATIONS SATELLITES IN GEO AS OF 1984. (REF. 6)

(3) most active spacecraft in GEO require accurate position control and station-keeping above their Earth subpoint, thereby reducing the likelihood of mutual collisions.

These considerations, however, are offset by the limited orbital slots available in GEO and the steady increase in the number of GEO satellites launched each year (Fig.6-10). Also, meteoroids cross the GEO belt with high relative velocities, so their background collision hazard remains at a level comparable with that of LEO. An unknown factor is the amount of unmonitored debris in GEO, because objects at such high altitude are more difficult to detect and monitor with radar or optical telescopes.

A number of articles discuss the collision probabilities of satellites in GEO.[10-20] In general, the collision probability is a complicated function of orbital parameters, relative position, velocity, projected areas of the spacecraft and time. The collision probability, P, of satellite collisions assuming a uniform distribution of space objects is,

$$P = A \bullet \rho \bullet v \bullet t$$

where: ρ = object density

A = projected area of the satellite

v = relative velocity of the target satellite

t = time interval associated with the (periodic) satellite motion

Takahashi[15] and Chobotov[11,16] have developed models for estimating collision probabilities for GEO satellites. Both models use the above relation as the basis for derivation of collision probabilities. Takahashi assumes the target satellite stays within fixed longitude/latitude bounds by appropriate station keeping. The satellite motion includes a small diurnal oscillations superimposed on a steady longitudinal drift. Maneuver corrections are applied every 15 days to maintain the satellite within the fixed longitudinal bound.

The right hand side of Figure 6-5 illustrates the diurnal oscillation/drift motions assumed by Takahashi. The satellite orbital bounds were assumed to be 0.01°, 0.05°, and 2 km for the longitude, latitude and altitude respectively.

If the orbital bounds for the diurnal motion are expressed in terms of increments in longitude Δ LON, latitude Δ LAT and altitude Δ ALT, the collision probability in three dimensions per orbit takes the form:

$$P = N \bullet (2\Pi R) \bullet L^2 \bullet (\Delta \text{ LON} \bullet \Delta \text{ LAT} \bullet \Delta \text{ ALT}) \bullet (\Delta \text{ LON} + (2/\Pi) \bullet \Delta \text{ LAT} + \Delta \text{ ALT}/R)$$

where L is the satellite diameter. The incremental bounds Δ LON, Δ LAT and Δ ALT are set by the magnitude of the diurnal motions along the longitude, latitude, and radial coordinates which are assumed to be equal to 0.01°, 0.05° and 2000 meters respectively. If an additional factor of 1/10 is introduced to account for the fact that collisions are only possible one out of every ten diurnal periods due to the longitudinal drift, then with these substitutions the above equation takes the form:

$$P = 9.51 \times 10^{-9} \times L^2 \text{ per half day}$$

This yields a satellite collision probability of $7 \times 10^{-6} \times L^2$ per year. For satellites having dimensions typical for those used in space communications, i.e., L = 2 meters, the probability of collisions in the geostationary orbit is extremely small. This changes when large space structures are considered, such as proposed satellite "farms," solar power satellites or orbiting space platforms. For an orbiting satellite of dimensions approaching 125 meters, the annual likelihood of a collision is about one in ten. For a hypothetical satellite "farm" of dimensions of 1000 meters, the expected frequency of collision increases to approximately once every 52 days.

The Chobotov approach considers the collision probability between geostationary satellites in circular orbits (in the equatorial plane) and geosynchronous satellites moving in an orbital plane with small inclination angle i and orbit eccentricity e. The satellite density, ρ, is proportional to the relative dwell time the satellite spends within a spatial volume defined by the following "bounds":

Longitude bound = $2 \Pi R$, Latitude bound = $2 R \sin i$, Altitude bound = $2 R e$,

where: R is the distance of satellite from Earth's center.

For a geostationary satellite of radius R_s, the probability of collision, P, with another satellite in one revolution or a 24 hour period is on the order of $P = 2.83 \times 10^{-13} R_s^2$ per day.

For a population of over 200 satellites, assuming one satellite every 2^0 longitude, each with radius of 50 meters, the probability is 2.2×10^{-9} per day. Hence, the probability of a collision between a satellite in a circular geostationary orbit with other satellites in low inclination orbits is extremely small.

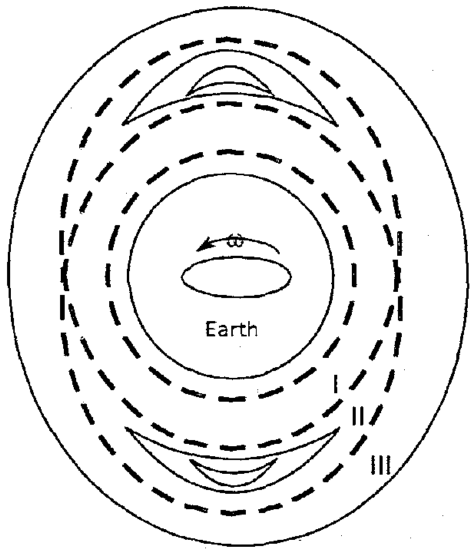
This probability of a collision between a spacecraft and spent GEO transfer stages is approximately two orders of magnitude less than that between two active GEO spacecraft, because of to the relatively small percent of the time (approximately 3%) that an object in an elliptical GEO transfer orbit spends at geosynchronous altitudes. The semi-geosynchronous ("molnyia") orbits favored for Soviet communication satellites are highly elliptical with low perigees and high relative near-Earth velocities.

To summarize, the low typical spatial densities in GEO of 2.5 - 7.5×10^{-10} objects/km^3, due to the roughly 550 objects which orbit in the 35, 750 + 50 km bin, combined with lower relative velocities in GEO and with typical station keeping capabilities, the probability of on-orbit collision is negligible at present(24).

6.4.4 Gravitational Drift Forces in GEO

Secular gravitational forces play an important role in altering the orbital characteristics of geosynchronous satellites. Depending on the point of origin of these forces, their effect on the orbit can be markedly different. These forces include the gravitational forces associated with the Earth's oblateness and the gravitational attraction of the Moon and Sun.(26)

The oblateness of the Earth (bulge in its in the equatorial plane) produces longitudinal drift forces in the east-west direction associated with the two geo-stable points located near 104.7°W and 75.3°E longitude. Without station-keeping capability, these forces cause GEO satellites to move in elliptic orbits in the longitudinal (and radial) direction with an oscillation period of about 820 days. Figure 6-11 shows a pictorial view of these drift oscillations.[27] The amplitude of excursion about these



Region I: Orbital periods shorter than 24 hours.
Region II: 24 hour orbits.
Region III: Orbital periods longer than 24 hours.

FIGURE 6-11. SATELLITE ORBITS RELATIVE TO EARTH

geo-stable points depends on the initial orbital departure from the geo-stable points, with the amplitude being zero for orbital paths that happen to cross the equator at the geo-stable points.

A second type of gravitational force is associated with the gravitational attraction of the Moon and Sun, which generate 'drift' forces along the north-south direction. The latter forces act to alter the inclination of the geosynchronous orbit causing an initial change in orbital inclination of about 0.86° per year. A maximum inclination of 15° is achieved in about 27 years at which point the inclination proceeds to decrease to zero in another 27 years. Superimposed on the above cyclical motions are small amplitude oscillations in the longitudinal and radial directions. These diurnal oscillations are characterized by a cyclic period of one (sidereal) day and have vastly smaller amplitudes (a factor of 10^6 and 10^3, respectively) compared to the longitudinal and radial motions described previously.

6.4.5 Collision Encounters in Geosynchronous Orbits

While slot allocation of GEO satellites generally attempts to maintain a minimum separation of two degrees longitude, in practice several satellites may share a common longitudinal location. This has led to procedures developed by the United States Air Force Satellite Control Facility (USAFSCF), recently designated the Consolidated Space Test Center (CSTC), to monitor all close approaches between primary communication satellites and other trackable objects coming within 300 km of these satellites. Predictions are made for all close approaches every seven days and appropriate user agencies are notified when the separation distance approaches 50 km. Collision avoidance maneuvers are considered at 5-8 km separation and are implemented if near simultaneous tracking of both space objects one to two days before encounter (closest approach) verifies the predicted positions of the satellites as accurate.

Typical data on geosynchronous orbit encounters over a 6 month period show that for 21 satellites examined there were 120 predicted encounters within the 50 km minimum miss distance.[15-17] Of these, several were in the 1-5 km range and required collision avoidance actions. The mean distance of closest approach was 21 km with a standard deviation of 13 km. Collision probabilities for these satellites were found to be up to two orders of magnitude greater than would be expected based on average density of objects in the geosynchronous corridor.

A total of six fragmentation incidents have occurred in the geosynchronous corridor, which have been suggested by some to be the possible result of actual collisions. In at least one of these, the satellite broke up into smaller debris components.

The question arises as to the potential liability of satellite owners and users for collision damage resulting when their spacecraft becomes inactive, remains in GEO, and collides with an active satellite. The accumulation of significant numbers of inactive satellites in GEO poses increasing collision hazards for active satellites. Takahashi estimated this collision probability using the same method previously applied (see Sec. 6.4.3) in the case of collisions between active satellites. Inactive satellites are assumed to have motion perturbations dictated by the Earth and by luni-solar gravitational/drift forces. Diurnal oscillations caused by the Earth's gravitational perturbations are superimposed on long-term (2-3 years) orbit evolution about one of two geo-stable points located at 75°E and 105°W longitude. Figure 6-12 shows a sketch of the long-term orbital evolution relative to Earth fixed coordinates. An additional secular motion excursion occurs in the north-south direction, causing a latitude variation of $\pm 14.7^{\circ}$ in a 54-year period.

The collision probability is estimated by determining the likelihood of collision in one sidereal day of a satellite confined within geosynchronous bounds of 0.1° longitude, 7.35° latitude and a 30 km

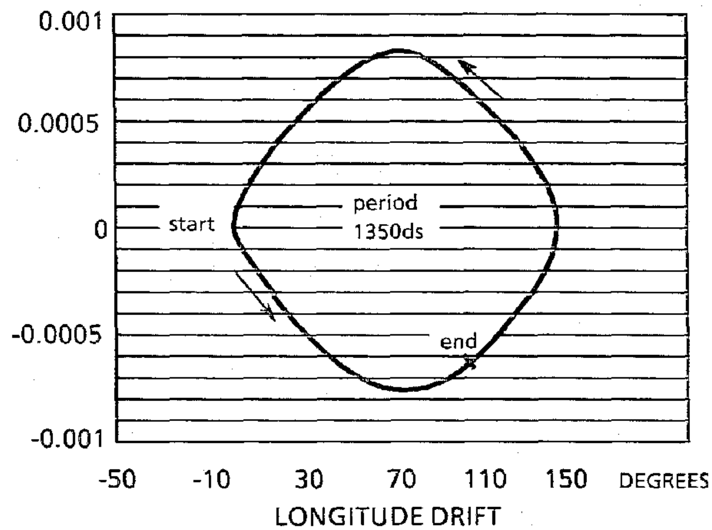


FIGURE 6-12. INACTIVE SATELLITE TRAJECTORY

altitude range. The effect of the secular orbital oscillations is to reduce the collision probability by a factor of 1/900. The estimated collision probability between an active and 'N' abandoned satellites of dimension 'L' then becomes:

$$P = 5.185 \times 10^{-13} \times N \times L^2 \text{ per half day.}$$

This gives a probability of 6.0×10^{-6} per year for a collision between an active satellite and an assumed total of 1000 abandoned satellites, each 4 meters diameter.

If the active satellite is assumed to be a large space platform of 125 meters across, the probability of collision with an estimated 1000 inactive satellites in one year increases to:

$$P = 730 \times 5.185 \times 10^{-13} \times 125^2 \times 1000 = 0.00591 \text{ per year}$$

Similarly, if a large solar power satellite with hypothetical dimensions of 1000 meters will be stationed in GEO, the collision probability in 1 year will become a sizeable 0.38 per year.

Hence, large GEO satellite clusters or platforms will have a high probability for collisions, if the number of abandoned communication satellites is allowed to approach 1000.

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7. RE-ENTRY HAZARDS

7.1 DEFINITION AND NATURE OF RE-ENTRY

Re-entry occurs when an orbiting spacecraft comes back into the Earth's atmosphere.[1] Any object placed in Earth orbit will eventually de-orbit and re-enter the atmosphere; this includes launch and breakup debris of satellites and spent rocket stages. Above 200 miles altitude, space is considered a perfect vacuum.[2] In reality, space is never a perfect vacuum and regardless of the orbital altitude of an object, it creates drag which eventually degrades the satellite's orbit. The solar wind and solar flares impinge on orbiting spacecraft and gravitational perturbations (both terrestrial and luni-solar) modify the spacecraft orbit and shorten its lifetime in space. The result is that spacecraft tend to spiral slowly towards the Earth's surface. When objects re-enter the atmosphere, their orbits decay rapidly and many of them burn up prior to impacting the Earth's surface.

There are two different sets of conditions associated with either controlled or uncontrolled de-orbit to consider when evaluating risk from re-entering satellites and other space debris.[15,16] Controlled de-orbit usually applies to manned and reusable spacecraft which are designed to survive re-entry and be recovered. In this situation, retrorockets are fired at a scheduled time in order to place the vehicle into a transfer orbit which intersects the surface of the Earth. If the Earth had no atmosphere, the intercept point would be the intended impact point. With the atmosphere, however, the vehicle decelerates further and falls short of the predicted vacuum impact point. The impact point still can be predicted reasonably accurately under these conditions. Thus, the controlled de-orbit can be planned so the spacecraft will impact near a predetermined recovery point, minimizing the risk of inadvertent impacts on ships or ground and sea structures.

There are three major sources of uncertainty associated with predicting uncontrolled re-entry characteristics, namely: the atmospheric conditions at the time an object begins to re-enter, the time of actual impact with the Earth's surface and the area in which the re-entering object will impact. These uncertainties associated with uncontrolled re-entry increase proportionately with the object's orbital altitude and on orbit lifetime.

When an object has been orbiting for a period of time, a number of changes could have taken place over its lifetime. If the spacecraft failed in some way before it reached final orbit, its orbital parameters (inclination and eccentricity) could have changed. It may have strayed from its planned orbital path, failed to achieve final orbit or broken up in an explosion causing pieces to disperse in different directions. All of these failure modes have a direct impact on the variables (surface area, mass, shape of fragments and orbital characteristics) used in the prediction of re-entry hazards.

Small changes in orbital characteristics can drastically affect the manner of an object's passage through the atmosphere. The frictional heating and drag (deceleration) experienced in the atmosphere have large effects on the object. Small deviations from the predicted conditions of re-entry may result in large differences in re-entry hazards and the associated casualty expectation (see Section 7.6). These differences could be due to further break up caused by the shock of entering the atmosphere at high velocity, the burning and ablation (vaporization) experienced during re-entry or changes in direction or velocity due to the weather and wind conditions that slow re-entering fragments differentially at lower altitudes.

7.2 ORBITAL DECAY

The basic concepts of energy and angular momentum (see Ch. 4) can be used to answer most questions dealing with orbital and re-entry trajectories. They are used to predict the initial re-entry point and probable ground impact points. Orbiting satellites control their positions in space by using small rocket thrusters, thereby changing their velocity and direction. This process is called "station-keeping" and requires rocket fuel and special on board communications and control equipment. Therefore, it is possible, to some extent, to choose the initial atmospheric re-entry point when dealing with controlled re-entry.[3] However, few satellites have the ability, capacity or life expectancy to provide the station-keeping capability towards the end of their life.

All space objects that orbit the Earth do so because of the various forces acting on them. These forces change the position and velocity of the object relative to Earth in such a way that their orbital characteristics become very predictable. The Satellite Surveillance Center (SSC),US Space Command (USSPACECOM), within the Cheyenne Mountain Complex in Colorado, monitors each satellite's past and present positions and predicts its future using these various orbital characteristics and dynamic processes. To determine a satellite's position at any given time, the computer uses an algorithm based on the laws of Space Mechanics.[2,3,12] The computer can predict the orbital path of the object with the object's historical position and velocity information. The Space Surveillance Center (SSC) of the US Space Command processes tracking and monitoring data obtained by the Space Surveillance Network (SSN) to predict re-entries. Space debris of the more than 90 satellite collisions or spontaneous break ups and 20 payload explosions in space have been documented to date (see also Chapter 6).[4,5,8]

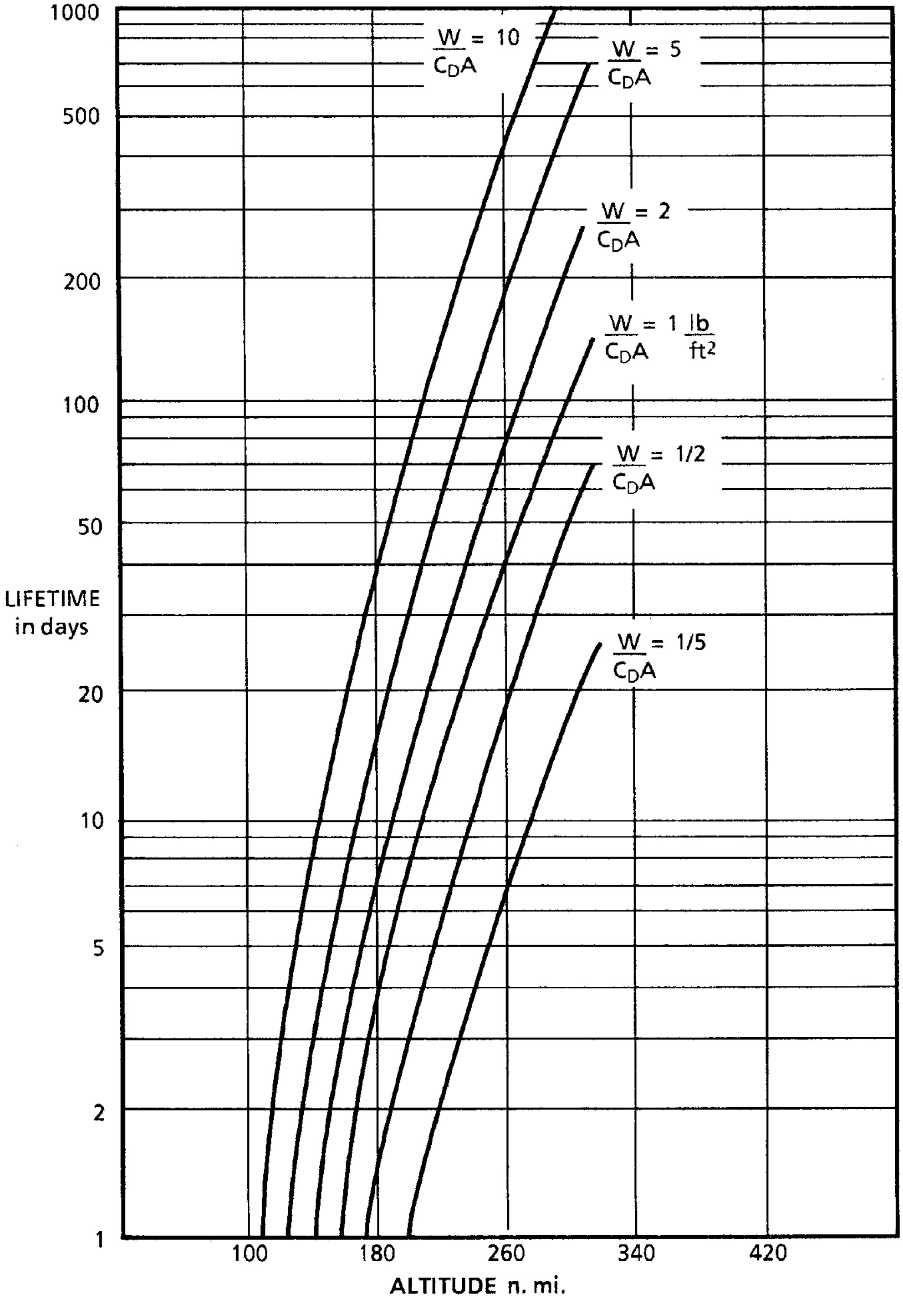
External perturbations due to the Earth's oblateness, the gravitational tugs of the Sun and Moon, the solar plasma storms and atmospheric friction cause long-term changes in the orbital parameters of satellites. These forces also affect the on orbit lifetime and re-entry. Theoretically, all forces acting on near-Earth satellites can affect a satellite's on orbit lifetime. The effects of solar storms on the atmosphere and the oblateness of the Earth have a much more significant effect than the gravitational attractions of the Sun, Moon and the other planets. NASA/Marshall scientists have taken these factors into account in designing an orbital lifetime prediction program. This program, called LIFTIM, uses a direct numerical integration of the time rates of change due to atmospheric drag using a Gauss-Legendre procedure in conjunction with the Jacchia atmosphere model.[6]

An orbiting object loses energy through friction with space plasmas above the atmosphere so that it falls into a slightly lower orbit and eventually spirals towards the Earth's surface. As the object's potential energy, represented by its altitude, is converted to kinetic energy, its orbital velocity increases. As an object's orbital trajectory is brought closer to Earth, it speeds up and outpaces others in higher orbits. Thus, a satellite's orbital altitude decreases gradually while its orbital speed increases. Once it enters the upper reaches of the atmosphere, atmospheric drag will slow it down more rapidly and eventually cause it to fall to Earth.[4]

Atmospheric drag, particularly near perigee, leads to the gradual de-orbit and re-entry of satellites. Satellites in LEO with less than 90 minute periods, corresponding to orbital altitudes of 100-200 nmi (or 185-370 km), re-enter within a couple of months. Above about 245 nmi (455 km) orbital altitudes, orbital lifetimes exceed several years. Above about 500 nmi (900 km) altitudes orbital lifetimes can be as long as 500 years.[5] Figure 7-1(a & b) illustrate Earth orbit lifetimes of satellites as a function of drag and ballistic coefficients (see Section 7-3) for circular ($e = 0$) and elliptical orbits with a range of altitudes. For elliptical orbits, the lower the perigee altitude, the higher is the apogee decay rate (P) and the shorter the on orbit lifetime.

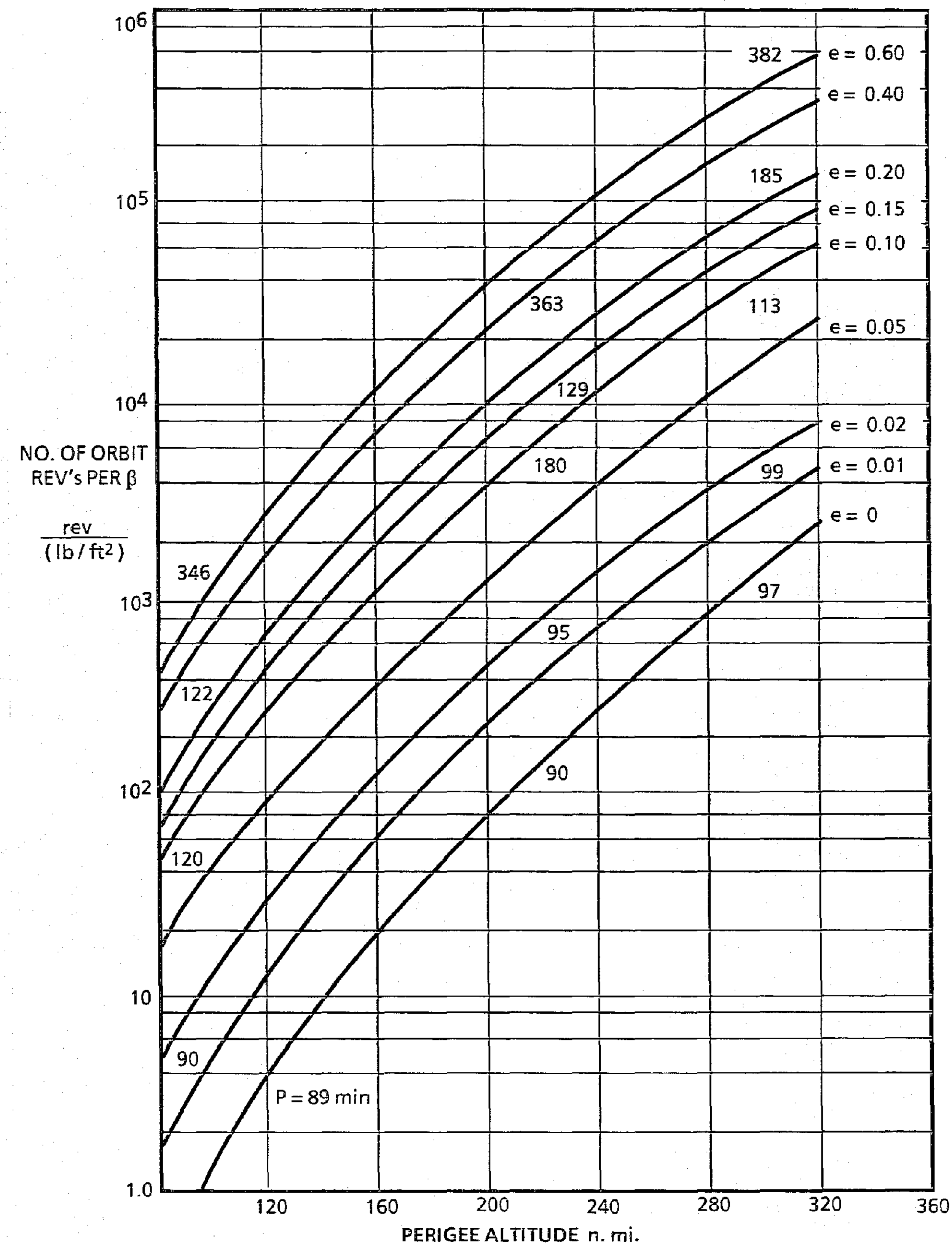
The ballistic coefficient β is equal to W/C_DA, where W is the spacecraft weight, C_D is the drag coefficient (which varies with shape) and A is the projected frontal area of the re-entering object. The more mass per unit area of the object, the greater the ballistic coefficient and the less the object will be consumed during its atmospheric crossing. The ballistic coefficient of a piece of debris is an important variable in the decay process as illustrated in Figure 7-1(a & b). A fragment with a large area and low mass (e.g., aluminum foil) has a low β and will decay much faster than a fragment with a small area and a high mass (e.g., a ball bearing) and will have a shorter orbital life. The combination of a variable atmosphere and unknown ballistic coefficients of spacecraft and launch and orbital debris make decay and re-entry prediction an inexact science at best.[7]

An examination of 104 successful space launches of 1985 revealed that the payloads from no less than 47 had re-entered within a year of launch. As a rule of thumb, it is suggested that about 70 percent of the annual mass put into orbit re-enters the atmosphere within 1 year of launch. Another 5 percent of the original annual mass may be expected to re-enter within 5 years from launch.[8] For example, from July 1 to October 1, 1987, of the 121 objects which de-orbited, 53 were payloads launched in that period.[17]



NOTE: 1 n. mi. = 1.852 km and 1 day = 24 hrs = 1440 min

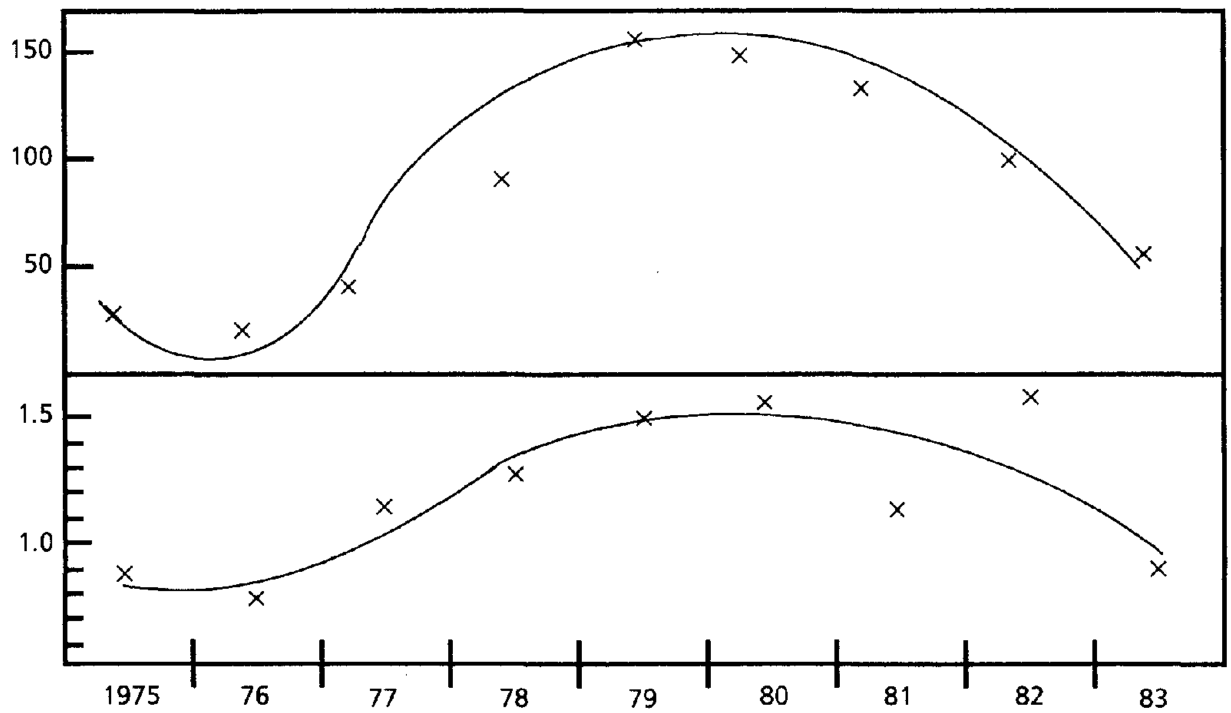
FIGURE 7-1a. EARTH ORBIT LIFETIMES - CIRCULAR ORBITS (Ref. 1b)



NOTE: e is the eccentricity of the orbit, the ballistic coefficient, β, is assumed to be 10 lb/ft². if 1/2 β is used the decay rates are double those in the figure.

FIGURE 7-1b. EARTH ORBIT LIFETIMES - ELLIPTICAL (REF. 1b)

USSPACECOM's SSC currently tracks about 7000 cataloged objects and may issue Tracking and Impact Prediction (TIP) messages which predict re-entry times and points of impact for about 500 re-entries each year. For example, in 1979-1980, 900 new objects were cataloged, but the total tracked population decreased by 300. The satellites were "purged" during the solar sunspot maximum which effectively increased the atmospheric density in LEO, thus, increasing orbital decay rates. Atmospheric drag is directly related to solar activity: High solar activity heats the upper atmosphere, increasing the atmospheric density by more than 10 times the average density at most satellite altitudes. This exerts a greater braking force on satellites and causes an above average number of objects to re-enter the atmosphere.[9] Thus, satellites decay in much greater numbers near Sunspot maximum than at a time of low solar activity (Figure 7-2).[10] Hence, the 11 year sunspot



UPPER PART: RELATIVE SUNSPOT NUMBERS LOWER PART: RATIO OF DECAYED vs. NEWLY APPEARED OBJECTS.

FIGURE 7-2. EFFECT OF SOLAR ACTIVITY ON SATELLITE DECAY

cycle is a periodic natural "sink", removing orbiting satellites from the near-Earth environment and thereby increasing re-entry hazards.

During the past 5 years there have been an annual average of 548 decays from lower altitude orbits (i.e., about three satellites re-entering every 2 days). Almost 83 percent of Earth satellites reside in LEO orbits (see Chapter 6) with periods of less than 225 min (about 4 hrs) and are near term re-entry candidates (see Figs.4-3 and 7-1). The total number of satellite decays per year is shown in Figure 7-3.[11]

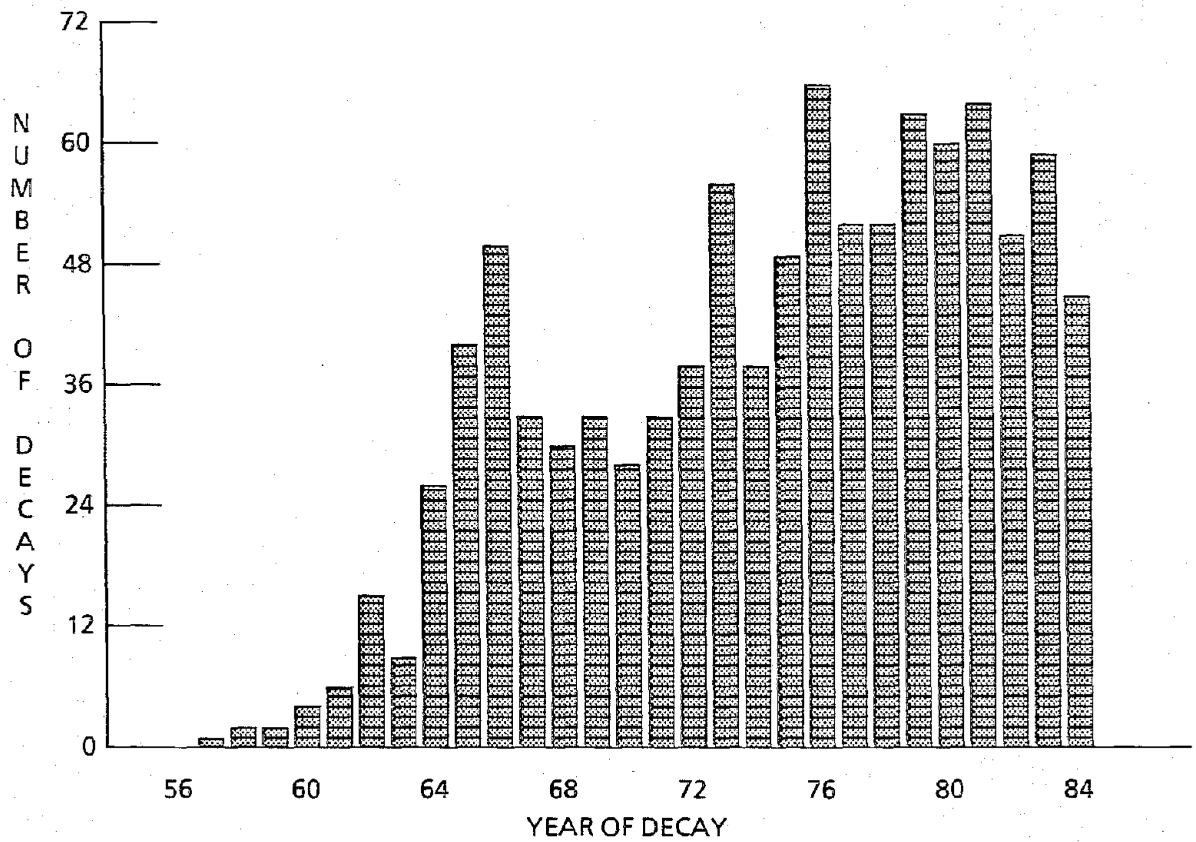


FIGURE 7-3 CATALOGED SATELLITE DECAYS

7.3 RE-ENTRY SURVIVABILITY

The information mentioned above would suffice to predict re-entry and ground impact points for spacecraft only if no other variables affected the re-entry process. In reality, the Earth's atmosphere, which is very sparse at high altitudes, interacts with the spacecraft. A vehicle approaching the Earth's atmosphere from space possesses a large amount of kinetic energy, due to its high relative velocity, and potential energy due to its orbital altitude above the Earth. When it encounters the atmosphere, a shock wave forms ahead of the vehicle, heating the atmosphere in this region to very high temperatures. The high temperatures due to friction with atmosphere reduce the vehicle's velocity and convert the vehicle's potential energy into heat absorbed by the object and its wake. If the vehicle slows down quickly, the total amount of heat to be absorbed by the vehicle is reduced. This explains the blunt (high drag) shape of re-entering spacecraft in the pre-shuttle manned space program. However, the total heat generated in the shock wave is still too great to be absorbed by metals which heat up and melt. Therefore, since it takes significantly more heat to vaporize material than to heat or melt it, materials used in heat shields were designed to ablate (vaporize) in the presence of the extreme temperatures. The net effect is that ablative protection allows objects to survive re-entry.

If the total energy of the spacecraft were converted to heat, it would vaporize the vehicle. The survival of meteorites to ground impact is proof that not all of the energy is converted into heat, but enough is converted to cause surface ablation. Actually, a large portion of the total energy is diverted away from the vehicle. If the object conducted the heat away from the forward surface and the total body could absorb the heat of re-entry without breaking up, then the object would re-enter the Earth's atmosphere and descend to Earth in a predictable way.(12) Heat shields and special shaping of forward surfaces are used to minimize frictional heating effects on the rockets and payloads during space launches, to protect them from heat and control ablation.

Surface heating effects depend on the vehicle's shape, composition, altitude and velocity. For re-entry at small angles of inclination when the vehicle deceleration rate is small, the surface heating rate is correspondingly small. For re-entry at large angles of inclination where the vehicle decelerates rapidly in the atmosphere, the surface heating rate will be greater but the time spent in the atmosphere will be shorter.(3)

Spacecraft which are not designed to survive re-entry generally do not have ablative surfaces nor are they very stable aerodynamically. The usual sequence of events in the re-entry process is as follows:

1. As the vehicle starts to re-enter, heat is generated by the shock wave and a portion is absorbed by the surface of the structure. As the structure heats up thermal energy is radiated out at a significantly lower rate than it is being absorbed.
2. The heated structure weakens and when the aerodynamic forces exceed its structural strength, it starts to come apart.
3. The heating process continues on the remaining parts of the structure, repeatedly breaking it up into still smaller pieces.
4. These structural pieces continue to heat up and eventually melt and vaporize if there is sufficient temperature and time exposure. Some structural elements can survive if they are massive or were shielded from the heat by other parts of the structure.

After the atmospheric re-entry point has been predicted, various other conditions must be taken into account to predict a ground impact point. Some of these conditions are orbital corrections due to frictional heating, break up due to atmospheric shock, drag and prevailing meteorological conditions. All of these factors are important when assessing the hazards from re-entering objects to people and property.(12)

7.4 RE-ENTRY IMPACT PREDICTION

The ground trace of an orbit is the path over which the satellite orbits the Earth (see Figure 7-4). If

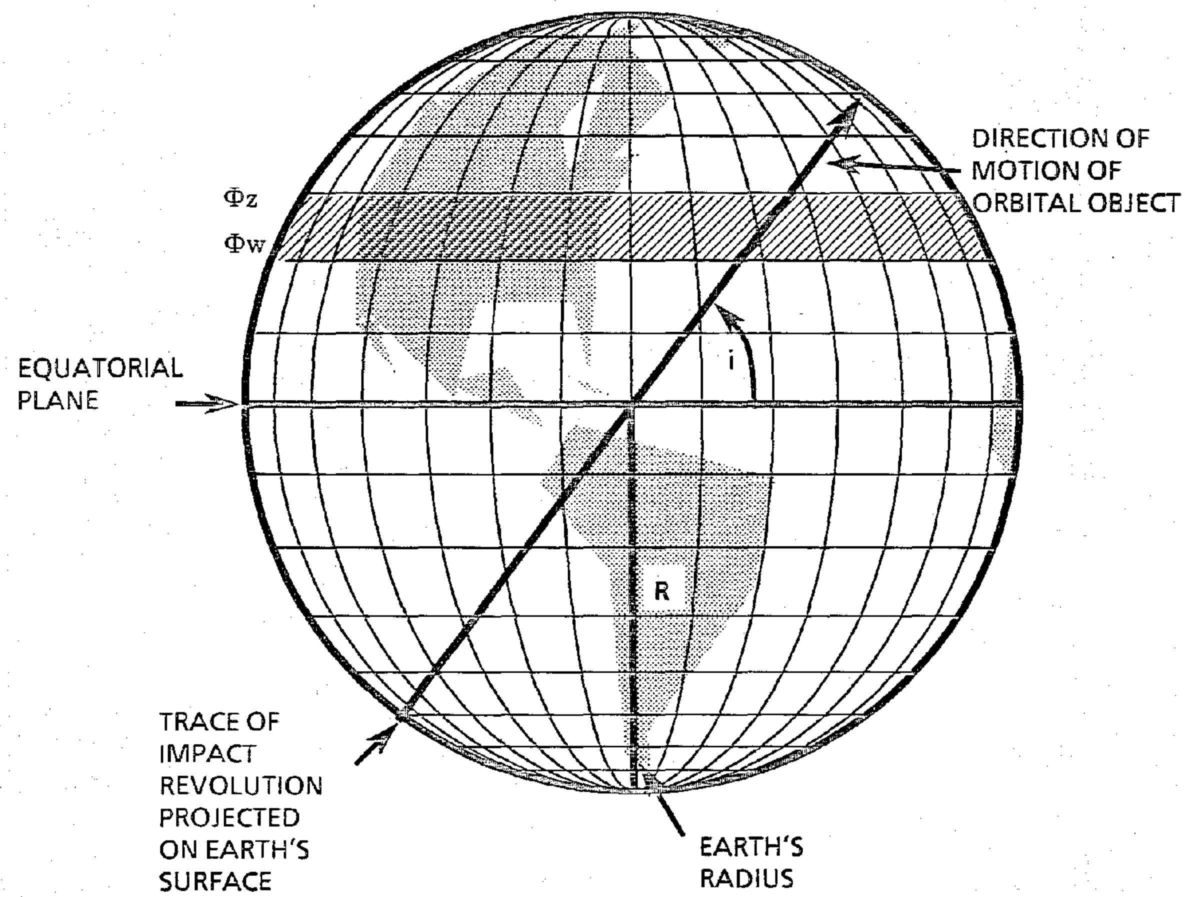


FIGURE 7-4 PLAN VIEW OF THE EARTH AND GROUND TRACK

there were a string between the center of the Earth and a satellite, the course marked by the intersection of the string with the surface of the Earth would be the trace of the orbit. Depending on the orbit, this ground trace could cover a large portion of the surface of the Earth (see Figure 7-5). If a satellite is tracked on a regular basis, it is possible to anticipate its approximate re-entry time and make an approximate prediction of the impact point. However, this does not give control over the position of the impact point and impact prediction uncertainties are usually rather large (on the order of 10's to 100's of miles).

One of the most critical factors in the re-entry process is the ballistic coefficient of the object, as discussed above. The ballistic coefficient is the ratio of gross weight to the drag coefficient multiplied by the reference area (W/C_DA). The relationship between the ballistic coefficient and the

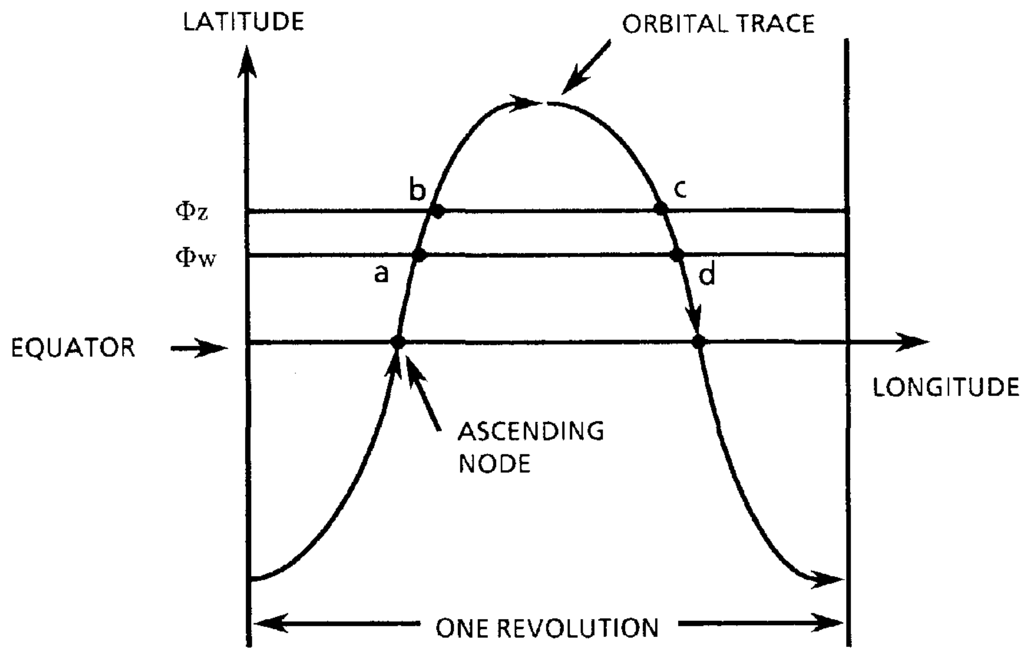


FIGURE 7-5 ORBITAL TRACE GEOMETRY

orbital lifetime is also linear, as illustrated in Figure 7-1(a & b). Small particles tend to have shorter lifetimes at a given orbital altitude than larger ones. This has been observed in the case of solid rocket motor debris where measurements made shortly after motor firings have shown a rapid increase in debris levels, but relatively rapid decay of small debris. A second indirect confirming observation is the shape of the debris flux curve as a function of debris size.[13](See Chapter 6).

As a satellite re-enters the atmosphere it decelerates. As discussed above, the deceleration rate is a function of many variables: entry angle, lift to drag ratio (L/D), the ballistic coefficient, the orbital parameters, the Earth's rotation and oblateness, atmospheric density aberrations and winds. The entry angle and ballistic coefficient affect the chance that a satellite or debris object will survive re-entry and landing. The satellite may skip due to the lift caused by the object's angle of attack upon entering the atmosphere, each skip associated with a change in velocity, speed and entry angle. As discussed in Chapter 4, every orbit has an angle of inclination, which along with the apogee and perigee, defines the trace of an orbit.

During re-entry the original orbital inclination of the satellite remains relatively constant. This holds for the inclination angle of pieces of the satellite that return separately as well as pieces of a satellite which break up during re-entry. This near consistency holds because the magnitude of the orbital velocity in the inclination plane is very large. A vertical (radial) change in velocity does not change the orbital angle of inclination, but it changes the atmospheric entry angle (called radiant). A change in the velocity component perpendicular to the plane of the orbit may affect the angle of inclination, but the magnitude of this change is minor compared to the magnitude of the velocity in the orbital plane.

7.5 IMPACT DISPERSIONS

Most satellites to date have been inserted into orbit with little or no consideration given to their eventual re-entry. The primary reason for this is that re-entering satellites are not likely to result in hazardous impacts given that 2/3 of the Earth's surface area is covered by oceans. Most of the objects which re-enter are likely to fragment and burn up in the upper atmosphere and make only negligible changes in its chemical composition. Even if an object does survive, only one third of the Earth is land area and only a small portion of this land area is densely populated, so the chance of hitting a populated land area upon re-entry is relatively small.

There is no standard way of computing impact dispersions currently. The calculations are two-fold. Estimates must be made for the number of pieces which will survive re-entry and the area over which each piece could cause damage, the "casualty area". For each piece of debris that will survive re-entry, a man-border area is added to the representative area of each incoming piece (see Volume 3, Chapter 10). The representative area is the maximum cross section area of the re-entering piece of debris. The man-border allowance is usually a ten inch addition in the radius to allow for the center of a person standing outside the actual impact radius but close enough to be hurt.[16] The splatter and rebound of fragments from hard ground impact must also be considered in these calculations.

7.6 RE-ENTRY HAZARD ANALYSIS

Most re-entering satellites and space debris are not controlled and the uncertainties of orbital decay are such that impact areas cannot be determined. Re-entry risk estimation generally assumes that the satellite can impact anywhere on Earth between the maximum northern and southern latitudes associated with the inclination of the orbit (see Figure 7-4).[16] Uncontrolled re-entry may be due to launch failures when the spacecraft fails to achieve final orbit, when the perigee/apogee kick motors malfunction and retain the satellite in a degradable transfer orbit or from second and upper stages jettisoned in orbit after burn out.

The probability of a re-entering spacecraft and/or its fragments landing within a particular latitude band depends on both the orbital inclination and the latitude spread of the ground track. Satellites in orbit spend disproportionately more time within the 1° wide band near the maximum latitudes. This is due to the change in direction of the satellite in this area, illustrated in the orbital ground trace of Figure 7-5, and is clearly visible in the probability distributions shown in Figure 7-6. In this figure the sharp peaks for each angle of inclination occur in a very small range around the latitude

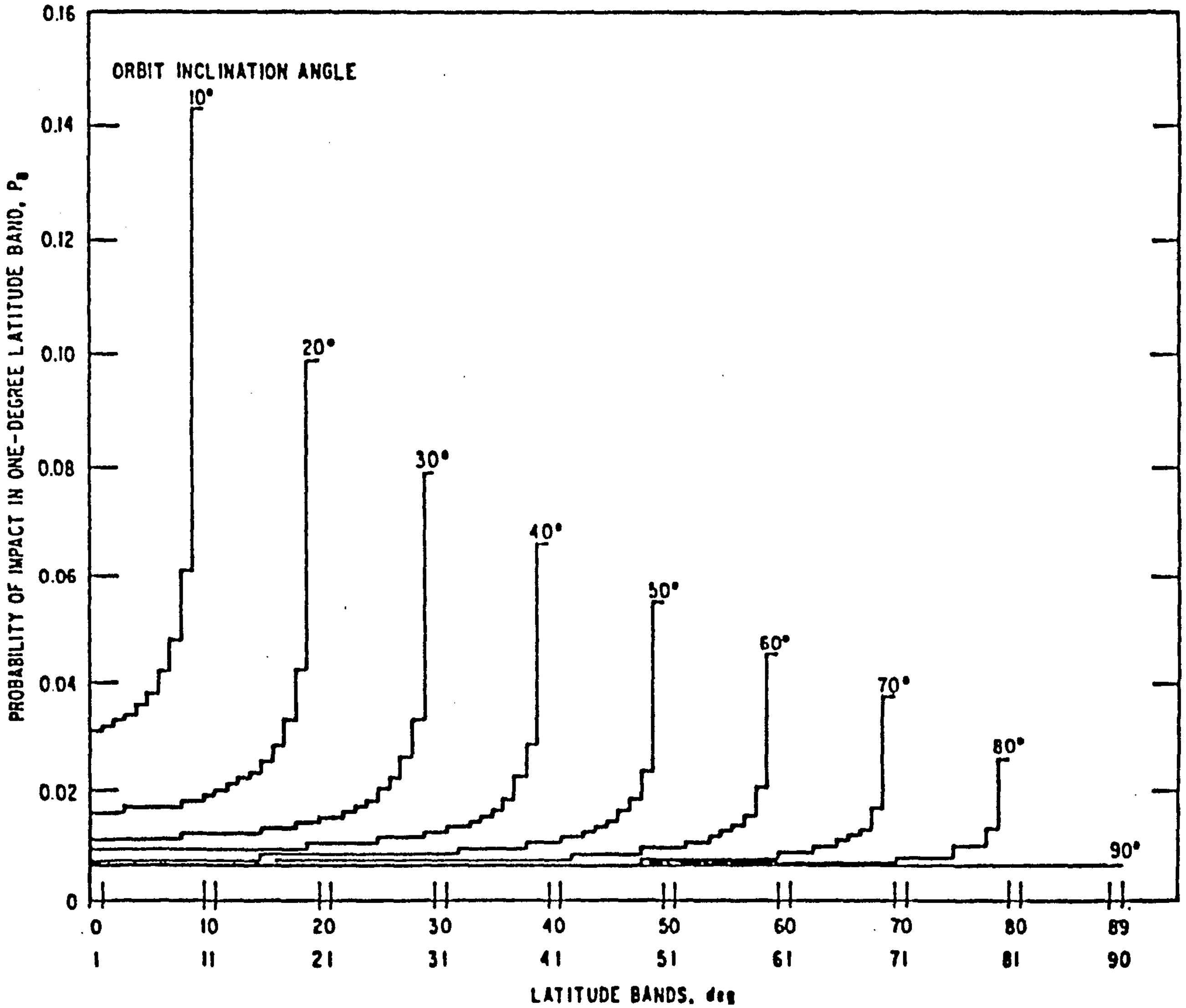


FIGURE 7-6 PROBABILITY OF IMPACT IN ONE-DEGREE LATITUDE BANDS (REF. 15)

extremes. The probability of impacting within a specified longitude range is assumed to be uniform (equi-probability over 360° of longitude). A corresponding bivariate probability density can be constructed for the location of such random debris impact. This assumes that the satellite or debris from the satellite survive the aerodynamic heating of re-entry. Once the probability density for ground impact has been established, the distribution of population within the probable impact area must be considered, as shown in Figure 7-7.[15] In this figure the population distribution is combined for the northern and southern hemispheres as a matter of convenience. Although the population number and distribution has changed in the interim, the approach used in Fig. 7-7 is still valid.[15] An orbiting object will spend an equal amount of time, within a certain band width, on both the north and south sides of the equator.

The casualty expectation is usually computed using the formula:

$$E_c = P_i \times (\text{Population Density}) \times A_c$$

Where P_i is the impact probability, the population density is the number of inhabitants per unit area, and A_c is the casualty area of the debris that survive to impact. Figure 7-8 presents an updated world-wide (average) casualty expectation, as a function of orbital inclination angle and debris impact casualty area.[19] In the example shown, a satellite in an orbit inclined at 26°, with debris having a casualty area of 100 sq. ft., will produce "on the average" 1.2×10^{-4} casualties upon re-entry.[15,19] This translates to one chance in 8333 of a casualty resulting from re-entry of this satellite. This is due to the unpredictability of the impact area during uncontrolled re-entry as opposed to the localized casualty area during launch. With no control over the time and location of re-entry, impact could occur in any country between the latitudes of $\pm 26°$.[16,18] Up to now, there have been no reported land impacts, damage and/or casualties by re-entry debris.[20] Roughly 100 of the approximately 3,100 objects resulting from 44 launches between 1956-1972 have survived re-entry and were recovered.[20] Identified re-entry debris include such diverse items as: tank pieces, nozzle pieces, small spherical gas tanks, plastic shrouds and other fragments.[20]

Particular re-entry hazards to the public are posed by orbiting nuclear payloads. Since 1961, both the US and the Soviet Union have launched nuclear power cells into space (See Table 7-1). While there have been no commercial payloads with nuclear materials, it is important to discuss generic re-entry hazards of this type. To date , such missions have required detailed risk analysis and interagency review. However, the US has launched passive, naturally decaying nuclear fuel cells, while the USSR has orbited RORSAT satellites with active nuclear reactors at relatively low altitudes in orbits which decay in a matter of days to weeks. Twenty eight such Soviet nuclear satellites were launched between 1967 and 1985, each carrying roughly 50 kg of U^{235}. Of these, 26 have been transferred successfully into higher altitude parking orbits (over 900 km) at their end of duty to permit decay of radionuclides before re-entry. However, at least six have failed and undergone uncontrolled re-

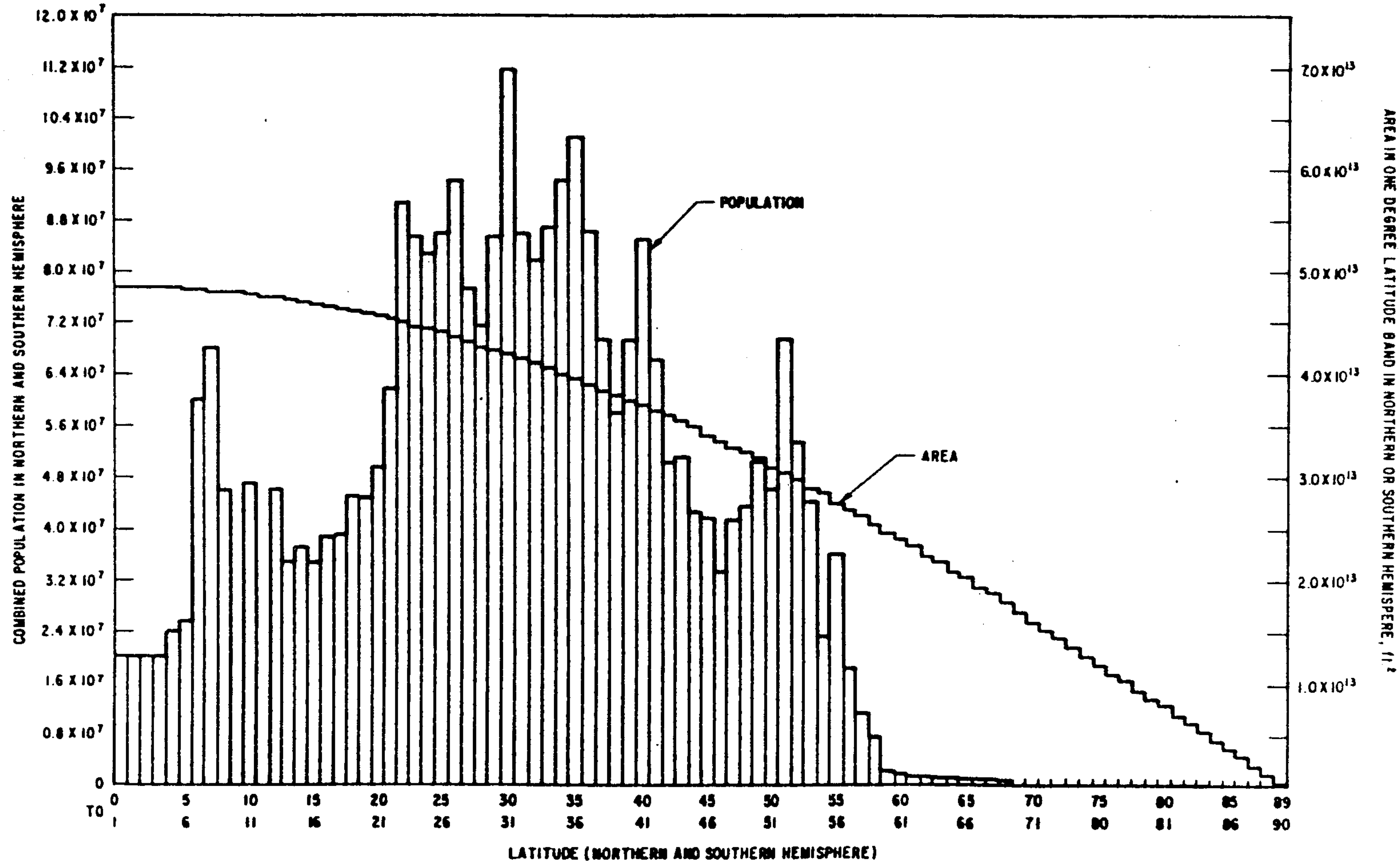


FIGURE 7-7 POPULATION AND AREA DISTRIBUTION OF THE WORLD BY ONE-DEGREE LATITUDE BANDS (POPULATION STATISTICS FOR 1965)

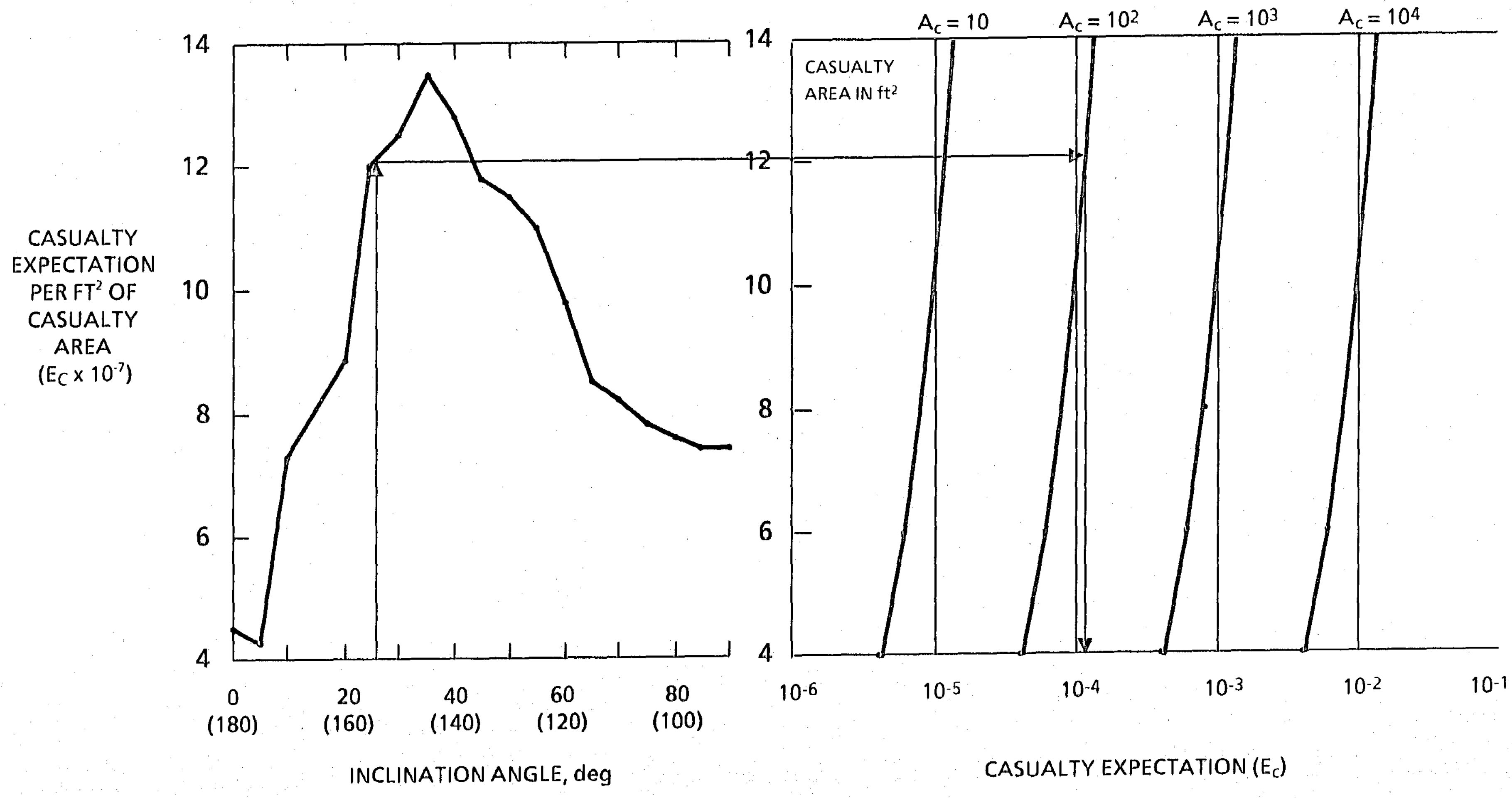


FIGURE 7.8 WORLD-WIDE CASUALTY EXPECTATION vs. ORBIT INCLINATION ANGLE (POPULATION ESTIMATED FOR 1990) (REFS. 15, 19)

TABLE 7-1 RE-ENTRIES OF SPACE NUCLEAR POWER SUPPLIES (REF. 15)

	NAME	LAUNCH DATE	RE-ENTRY	TYPE OF POWER SUPPLY	COMMENTS
USA	Transit 5 BN3	21 April 1964	21 April 1964	Radioisotope	Launch failure. SNAP 9A destroyed over Indian Ocean
	Nimbus B	18 May 1968	19 May 1968	Radioisotope	Launch failure. SNAP 19 recovered off California coast.
	Apollo 13	11 April 1970	17 April 1970	Radioisotope	SNAP 27, designed for deposit on lunar surface, re-entered over Pacific Ocean during emergency return of Apollo 13 astronauts.
USSR	-	25 January 1969	25 January 1969	Reactor	Possible launch failure of ocean surveillance satellite.
	Kosmos 300	23 September 1969	27 September 1969	Radioisotope	One or both of these payloads may have been a Lunikhod, designed for remote exploration of the Moon carrying a Po^{210} heat source. Upper stage malfunction prevented payloads from leaving Earth orbit.
	Kosmos 305	22 October 1969	24 October 1969		
	-	25 April 1973	25 April 1973	Reactor	Possible launch failure of ocean surveillance satellite.
	Kosmos 954	18 September 1977	24 January 1978	Reactor	Payload malfunction caused re-entry near Great Slave Lake in Canada. Local contamination detected.
	Kosmos 1402	30 August 1982	23 January 1983	Reactor	Payload Failed to boost to storage orbit on 28 December 1982.
			7 February 1983	Fuel core	Reactor re-entered at 25° S, 84° E. Fuel core re-entered at 19° S, 22° W.

entry and atmospheric break up, one showering debris over N. Canada in 1978 and two others over the Indian Ocean in 1983 and 1987. In contrast, the US nuclear fuel cells are designed to survive atmospheric re-entry and impacts. Three radio-isotope thermal generator (RTG) power supplies accidentally re-entered as a result of launch and/or orbital insertion failures (in 1964,1968 and 1970); no undue public exposure to radioactivity resulted from any of these.[14]

Although the possibility of a satellite landing in a populated area is small, the hazards are real and in certain instances, potentially very serious. Cosmos 954, the Soviet nuclear satellite that scattered nuclear debris over Canada upon re-entry and caused over $12 million in damages and cleanup costs is one example of a potentially serious re-entry hazard.[21] Fortunately, several other failed or deactivated Soviet RORSAT and US nuclear satellites have returned over oceans (Table 7-1).

Issues related to re-entry hazards are currently under active re-examination and are undergoing research.

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Volume III: Risk Analysis

8. RISK ANALYSIS METHODOLOGY

8.1 WHAT IS RISK ANALYSIS?

Risk Analysis is the technical process and procedures for identifying, characterizing, quantifying and evaluating hazards. It is widely used in industry and by federal agencies to support regulatory and resource allocation decisions. The analysis of risk, also called Risk Assessment (see definitions of terms in Ch.1 and in the Glossary, App. A), consists of two distinct phases: a qualitative step of hazard identification, characterization and ranking; and a quantitative risk evaluation entailing estimation of the occurrence probabilities and the consequences of hazardous events, including catastrophic ones. Following the quantification of risk, appropriate Risk Management options can be devised and considered, risk/benefit or cost analysis may be undertaken and Risk Management policies may be formulated and implemented. The main goals of Risk Management are to prevent the occurrence of accidents by reducing the probability of their occurrence (e.g., practice risk avoidance), to reduce the impacts of uncontrollable accidents (e.g., prepare and adopt emergency responses) and to transfer risk (e.g., via insurance coverage). Most personnel safety and operational/handling precautions and requirements at hazardous facilities (and hardware design reviews and approval for plants and critical equipment) are intended to prevent, reduce the frequency or probability of occurrence of hazardous events and to minimize their potential impacts.

Both normal operations and unforeseen conditions can lead to accidents which cannot be prevented or controlled. In such cases, the residual risk must be accepted and managed by preparing emergency response procedures (e.g., evacuation and medical response plans) to lessen the consequences of such accidents. Deterministic and worst case scenario analyses are often used to assess the scope and exposure impacts of improbable hazardous events with high consequences.

Several recent reports have discussed the role of technical risk assessment inputs to regulatory analysis and policy decision making.[1-3] Since Risk Assessment is a field where safety and loss prevention are the chief concerns, conservatism at various steps in the analysis has often been adopted as a prudent approach. Thus, conservative assumptions have been compounded sometimes in setting unnecessarily stringent regulatory standards and requirements. In practice, excessive conservatism and use of "worst case" analysis has served as a basis for over-design of critical facilities, and over-regulation of industry by setting unnecessarily strict license and permit requirements.[4,5] Several mission Agencies (such as DOD, NASA, DOE, EPA, USBM, OSHA, NIH, NRC) have developed their own risk analysis tools to carry out studies either in support of regulatory standards, criteria and policies or to enable safe operations. For the past few years, an Interagency Task Force for Risk Assessment, led by the NSF, has been working on uniform standards, to the extent possible and practical, for risk analysis methods and their use by federal Agencies charged with protecting the safety and health of the workers and the public. Some of these tools and approaches, whether developed specifically for space applications (Ch.9) or for licensing decisions (e.g., NRC regulations and studies),[8,15,16] are transferable to DOT/OCST for regulation and oversight of commercial launch activities.

Risk Assessment provides the information necessary for Risk Management decisions. Risk Management, in a regulatory context, requires the evaluation of the impact and effectiveness of safety standards and regulations to impose additional controls or relax existing ones.

8.2 RISK PERCEPTION AND RISK ACCEPTABILITY

Subjective judgment and documented societal bias against low probability/high consequence events may influence the outcome of a risk analysis. Perceptions of risk often differ from objective measures and may distort or politicize Risk Management decisions and their implementation. Public polls indicate that societal perception of risk for certain unfamiliar or incorrectly publicized activities is far out of proportion to the actual damage or risk measure (by factors of 10-100 greater than reality for motor, rail and aviation accidents, but by factors of >10, 000 for nuclear power and food coloring).[14] Risk conversion and compensating factors must often be applied to determine risk tolerance thresholds accurately to account for public bias against unfamiliar (x 10), catastrophic (x 30), involuntary (x 100), immediate vs. delayed consequence (x 30) and the uncontrollable (x 5-10) risk exposure.[17]

Different risk standards often apply in the workplace, in view of voluntary risk exposure and indemnification for risk to exposed workers; as opposed to public risk exposure where stricter standards apply to involuntary exposure. The general guide to work place risk standards is that occupational risk should be small compared to natural sources of risk. Some industrial and voluntary risks may be further decreased by strict enforcement or adequate implementation of known risk management and risk avoidance measures (e.g., wear seat belts, stop drinking alcohol or smoking). Therefore, some of these risks are controllable by the individual (e.g., do not fly, take the car to work or smoke), while others are not (e.g., severe floods, earthquakes and tsunamis).

Relative Risk Assessment is a common method of ranking risk exposure levels which enables decision makers to define acceptable risk thresholds and the range for unacceptably high exposure that would require Risk Management resources for reduction and prevention. As Table 8-1 and Figure 8-1 illustrate, there are de facto levels of socially tolerated (acceptable) levels of risk for either voluntary or involuntary exposure to a variety of hazardous factors and activities. Although regulators often strive to assess absolute levels of risk, the relative ranking of risks is an appropriate

Risk Management strategy for resource allocation towards regulatory controls. Cost benefit analysis is often required to bring the burdens of risk control strategies to socially acceptable levels.

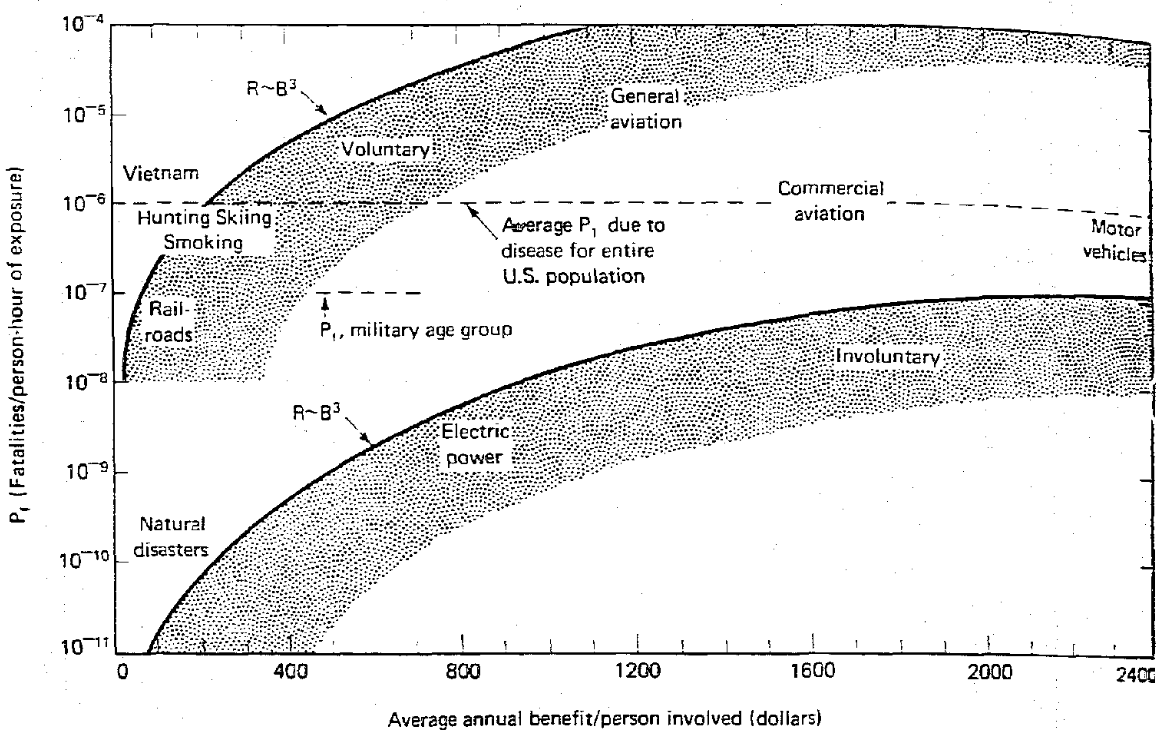


FIGURE 8-1. RISK VS BENEFITS (REF. 9)

Figure 8-1 and Tables 8-1 and 8-2 show estimated risk levels associated with natural and other (occupational, transportation, etc.) hazards that may lead to undesirable health effects and casualties. They show that risk levels vary greatly by causes of harm (chemical, mechanical, natural or man made), probability, degree of control, duration of exposure to the consequence (immediate, delayed, short or long-term), distribution (geographical, localized) in time and space, benefit to society vs. costs of risk reduction and consequence mitigation.

Table 8-1 shows the relative risk exposure to individuals as a casualty probability from various natural and regulated causes.(19) This table and its precursors in the literature(6,17) illustrate that the public voluntarily assumes risk levels which are 100 to 1,000 times larger than involuntary exposures to natural hazards and normal activities. These levels may be used as indicators of socially acceptable risk thresholds to compare when new regulatory standards are set. Note that risk exposure is normalized both to the population exposed and to the duration of the exposure. To compare the risk associated with each cause, consistent units must be used, such as fatalities or dollar loss per year, per 100,000 population, per event, per man year of exposure, etc.

Issues related to acceptable risk thresholds for regulatory purposes and for the public at large are often complex and controversial.(1-5,17,19) The typical approach to establish risk acceptance criteria

TABLE 8-1. INDIVIDUAL RISK OF ACUTE FATALITY BY VARIOUS CAUSES. (REF. 19)

ACTIVITY OR CAUSE	ANNUAL FATALITY RISK FOR EVERY 1 MILLION EXPOSED INDIVIDUALS
1. Smoking (all causes)	3,000
2. Motor vehicle accidents	243
3. Work (all industries)	113
4. Alcohol	50
5. Using unvented space heater	27
6. Working with ethylene oxide	26
7. Swimming	22
8. Servicing single piece wheel rims	14
9. Aflatoxin (corn)	9
10. Football	6
11. Saccharin	5
12. Fuel system in automobiles	5
13. Lightning	0.5
14. DES in cattle feed	0.3
15. Uranium mill tailings (active sites)	0.02
From all causes in U.S.	8,695
From cancer in U.S.	1,833

* Indicates that the risk was regulated by the Federal government in the last 10 years. For these activities or causes, the risks in the table are estimates of risk prior to Federal regulation.

TABLE 8-2. ANNUAL RISK OF DEATH FROM SELECTED COMMON ACTIVITIES

	Number of deaths in representative year	Individual risk/year
Coal mining:		
Accident	180	1.3×10^{-3} or 1/770
Black lung disease	1,135	8×10^{-3} or 1/125
Fire fighting		8×10^{-4} or 1/1,250
Motor vehicle	46,000	2.2×10^{-4} or 1/4,500
Truck driving	400	10^{-4} or 1/10,000
Falls	16,339	7.7×10^{-5} or 1/13,000
Football (averaged over participants)		4×10^{-5} or 1/25,000
Home accidents	25,000	1.2×10^{-5} or 1/83,000
Bicycling (assuming one person per bicycle)	1,000	10^{-5} or 1/100,000
Air travel: one trans-continental trip/year		2×10^{-6} or 1/500,000

Source: Hutt, 1978, Food, Drug, Cosmetic Law Journal 33, 558-589.

for involuntary risks to the public has been that fatality rates from the activity of interest should never exceed average death rates from natural causes (about 0.07 per 100,000 population, from all natural causes) and should be further lessened by risk control measures to the extent feasible and practical.(13)

The societal benefit and the cost trade-offs for risk reduction are widely used guides to set and justify risk acceptability limits. By comparing the risks and benefits associated with certain regulated activities, fair, balanced and consistent limits for risk acceptability may be set and institutional controls on risk may be established. Figure 8-1 is based on Ref.9: Starr's 1969 risk benefit analysis, which, although later challenged in the literature, illustrates several general trends derived from an analysis of fatalities per person hour of exposure to natural hazards and to hazardous human activities, in terms of dollar-equivalent benefit to society. It appears that voluntarily assumed risk levels are a factor of about 1,000 higher than involuntary risk exposure levels over the entire range of benefits. Also, the acceptable risk curve appears to vary as the cube power of the benefit, on this log- normal scale.

A typical regulatory risk threshold used to institute controls is the one-in-a-million casualty probability.[17] Situations at this threshold include: traveling 60 miles by car or 400 miles by air, two weeks of skiing, 1.5 weeks of factory work, 3 hours of work in a coal mine, smoking one cigarette, 1.5 minutes of rock climbing and 20 minutes of being a man aged 60.

By analogy with other industries, inthe case of space operations, Range personnel and commercial launch service firms may be considered voluntary risk takers, while the public at large is involuntarily exposed to launch and overflight risks. While Range Safety and on-site Range personnel are highly trained in risk avoidance and management, the public must be exposed to onlyminimal risk from commercial launch activities.

There are clear but indirect public, economic and other societal benefits derived from commercial space operations, including efficient telephone and video communications, weather forecasting, remote environmental sensing and crop data, better drugs, advanced material fabrication, superior navigation capability and other technology spin-offs. Based on the risk comparability approach illustrated in Ch. 5 (Vol. 2) and the Range Safety controls and practices (Chs. 2, Vol. 1 and 9, 10), commercial launch activities appear to be well within the socially acceptable risk limits at this time.

8.3 EXPECTED RISK VALUES AND RISK PROFILES

There are two fundamental components of Risk Analysis:

- Determination of the probability, P_i (or frequency of occurrence, f_i), of an undesirable event, E_i. The probability of an event is its likelihood of occurrence or recurrence. Sometimes the probability estimates are generated from a detailed analysis of past experience and historical data available; sometimes they are judgmental estimates on the basis of an expert's view of the situation or simply a best guess. This quantification of event probabilities can be useful, but the confidence in such estimates depends on the quality of the data base on actual failures and the methods used to determine event probabilities. Probabilities have long been used in the analysis of system reliability for complex equipment and facilities and to anticipate and control various failure scenarios.

- Evaluation of the consequence, C_i, of this hazardous event: The choice of the type of consequence of interest may affect the acceptability threshold and the tolerance level for risk.

The analytical phase of a Risk Analysis generally consists of three steps:(10) The triad: event (scenario), probability and consequence is sometimes called the "Risk Triplet."

1. The qualitative step involves the selection of specific hazardous reference events E_i (hazard identification) or scenarios (chains of events) for quantitative analysis.
2. The quantitative analysis requires the estimation of the probability of these events, P_i.
3. The next quantification step is to estimate of the consequences of these events, C_i.

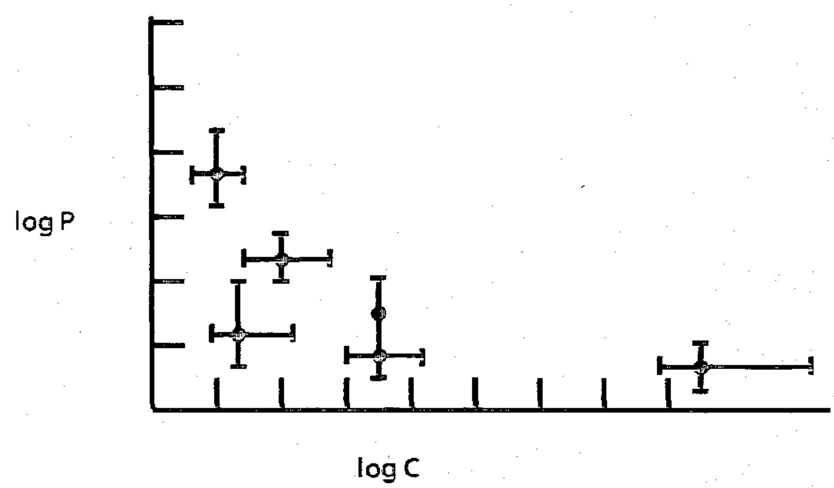
The results of the analytical phase are used in the interpretive phase in which the various contributors to risk are compared, ranked and placed in perspective. This interpretive phase consists of:

4. The calculation and graphic display of a Risk Profile based on individual failure event risks. The process is presented in Figure 8-2.
5. The calculation of a total expected risk value (R) by summing individual event contributions to risk (R_i).

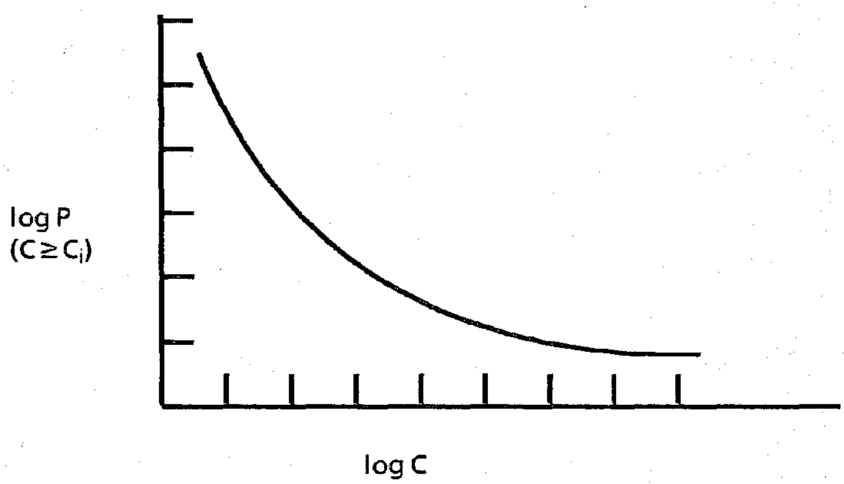
Naturally, all the calculations undertaken involve some uncertainties, approximations and assumptions. Therefore, uncertainties must be considered explicitly. Using expected losses and the risk profile to evaluate the amount of investment that is reasonable to control risks, alternative Risk Management decisions involving avoidance (i.e. probability decrease) or consequence mitigation can be evaluated in terms that are useful to the decision maker. Therefore, a sixth planning step usually included in Risk Analysis is:

6. The identification of cost effective Risk Management options, to be followed by:
7. Adoption of a Risk Management policy and implementation.

The analytical phase yields results in the general form suggested in Table 8-3. There are two useful ways to then interpret such results: expected risk values, R_i, and risk profiles. Both methods are employed for quantitative risk analysis.



(a) Plotting of points corresponding to individual failure events. Logarithmic scales usually used because of wide range in values. The error brackets denote uncertainties in probability estimates (vertical) and in anticipated consequences (horizontal) for each failure mode/event.



(b)Construction of the cumulative probability risk profile curve (as described in text)

FIGURE 8-2. CONSTRUCTION OF A RISK PROFILE

TABLE 8-3. GENERAL FORM OF OUTPUT FROM THE ANALYTIC PHASE OF RISK ANALYSIS

UNDESIRABLE EVENT	PROBABILITY* +	CONSEQUENCES** +	RISK LEVEL
E_1	P_1	C_1	$R_1 = P_1C_1$
E_2	P_2	C_2	$R_2 = P_2C_2$
E_3	P_3	C_3	$R_3 = P_3C_3$
.	.	.	.
.	.	.	.
E_N	P_N	C_N	$R_N = P_NC_N$

*Probability of an event is expressed as a fraction, or in percent (dimensionless). Alternatively, a frequency per year, or per event (in units of 1/time) may be used.

**Consequence, in the case of an accident is a measure of the accident impacts of interest to the analysis (e.g. mission loss, payload damage, damage to property, number of injuries, dollar loss, etc.)

+ Usually point values estimates for P_i and C_i are bracketed by best case - worst case estimates, to indicate the residual uncertainty in point estimates. Orders of magnitude in the range bracketing consequence and probability estimates are not uncommon, as the brackets in Fig.8-2 show.

Expected values are most useful when the consequences C_i are measured in financial terms or other directly measurable units. The expected risk value R_i (or expected loss) associated with event E_i is the product of its probability P_i and consequence values:

$$R_i = P_i \times C_i$$

Thus, if the event occurs with probability 0.01 in a given year, and if the associated loss is one million dollars, then the expected loss is:

$$R_i = 0.01 \times \$1{,}000{,}000 = \$10{,}000$$

Since this is the expected annual loss, the total expected loss over 20 years (assuming constant $) would be roughly $200,000. This assumes that the parameters do not vary significantly with time and ignores the low probability of multiple losses over the period. To obtain the total expected loss per year for a whole set of possible events, simply sum the individual expected losses:

$$\text{Total Risk, } R_T = P_iC_i + P_2C_2 + \ . \ . \ + P_NC_N = $$

$$= \sum_{i=1}^{N} P_iC_i \quad = \sum_{i=1}^{N} R_i$$

This expected risk value assumes that all events (E_i) contributing to risk exposure have equal weight. Occasionally, for risk decisions, value factors (weighting factors) are assigned to each event contributing to risk. The relative values of the terms associated with the different hazardous events give a useful measure of their relative importance and the total risk value can be interpreted as the average or "expected" level of loss to be experienced over a period of time. One particular way in which it is used is to compare it to the cost of eliminating or reducing risk (i.e., as part of the Risk Management strategy) in the context of cost/benefit analysis. Expected values of risk (R) are of prime importance in both business and in regulatory decision making under complex and uncertain situations.

Based on the definition of expected values, if event E_2 has ten times the consequences of event E_1 but only one tenth the likelihood, then the products $R_1 = P_1C_1$ and $R_2 = P_2C_2$ are equal. That is, the events have the same expected level of risk. Thus, expected risk levels provide a balance of probabilities and consequences. In mathematical terms, the expected values may be similar, but the low probability, high consequence event may be of greater concern.[11,12] For example, a company may be prepared to sustain a steady level of relatively small losses or accidents, but is concerned with guarding against truly catastrophic events. This is the motivation behind Risk Management, although, in all cases a range of consequences may be of interest. Determining the point estimates for best and worst case R_i will produce limiting values for the risk estimates and yield a band of uncertainty in risk level.

A common way to interpret the values of probabilities and consequences of different hazardous events is by means of a Risk Profile. This displays the probability distribution for accidents and the range of their severity as a function of likelihood. If sufficient accident data exist, the cumulative probability distribution function is used as a Risk Profile to show the probability of damages at a given level or greater. Figure 8-3 shows an example of a hypothetical Risk Profile for commercial launch operations. A point (P_i, C_i) on the curve can be interpreted to mean there is a probability, P_i, of an accident with a consequence at least as large as C_i. Given a set of ordered pairs (P_i, C_i) obtained during the analytic phase of a risk study, the actual Risk Profile is computed using the laws of probabilities and combinatorial analysis. For actual cases, the risk profile is usually constructed by drawing the lowest decreasing curve so that all the points with $C \leq C_i$ are on or below it. The separate hazardous events with consequences $C_i \leq C$ are combined into a single event with a probability equal to the sum of their individual probability values (i.e., their cumulative probability). Then, the ordinate value P_i in Figures 8-2 and 8-3 indicates the probability of an event, E_i, with a consequence as large as or exceeding C_i $(C \geq C_i)$. The acceptability ranges for risk must be determined and regulatory risk targets must be set consistent with these acceptable risk thresholds. These goals are often set according to ALAP (as low as practical), BAT (best available technology), BPT (best practical technology) or the cost of risk reduction.[17] The relative risk reduction achieved by various controls is also displayed on the Risk Profile to indicate the merit and effectiveness of potential regulatory risk reduction measures.

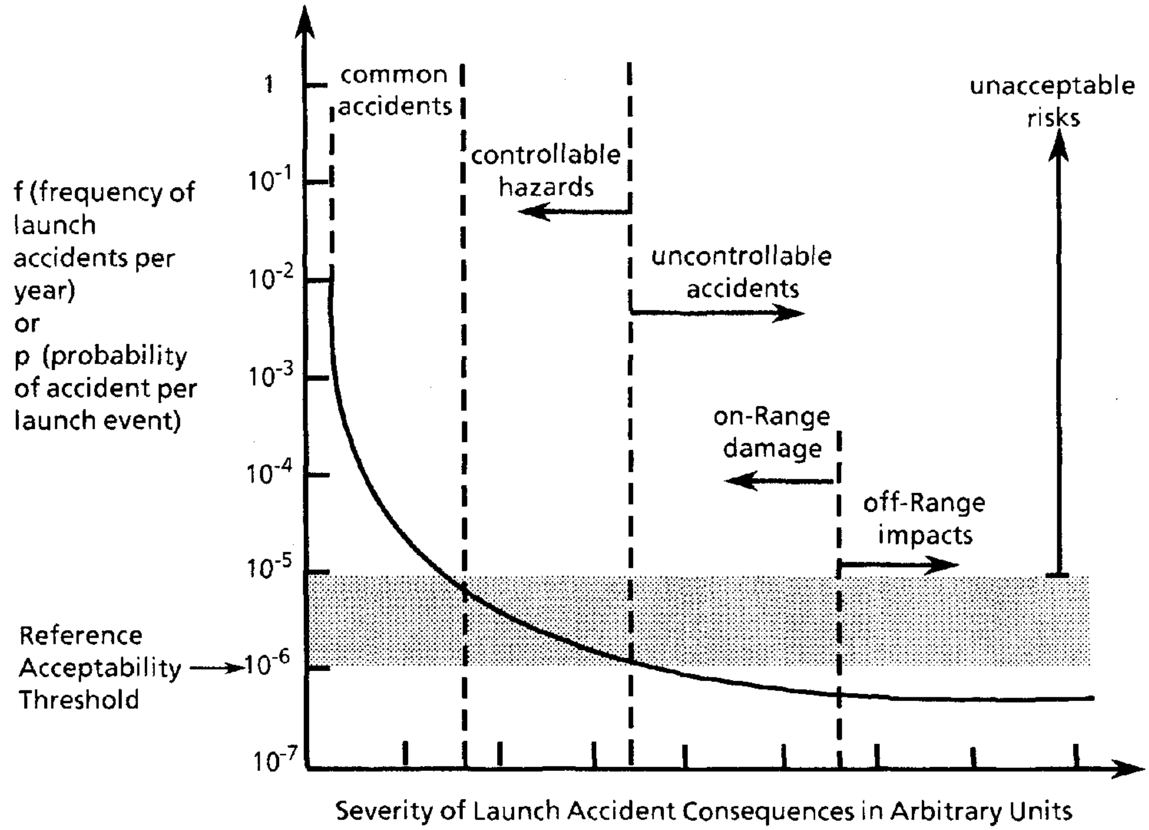


FIGURE 8-3. A SCHEMATIC RISK PROFILE FOR COMMERCIAL ELV OPERATIONS

The frequency or the probability of the undesirable event (launch accidents) is plotted against the consequence magnitude of interest (potential public safety impacts such as dollars loss for property damage, casualty, insurance claims). The shape of the curve could be convex, rather than concave, or even discontinuous, depending on the scale and the data points available. Shaded area denotes de-facto acceptable risk levels or design/operation safety goals based on established ELV launch practices at Government Ranges.

8.4 IDENTIFICATION OF HAZARDS, PROBABILITY ESTIMATION AND CONSEQUENCE MODELING

Fault tree (or event tree) analysis has been successfully applied in many technical fields to identify and logically order scenarios leading to equipment breakdown, financial loss or other system failures to be controlled (see section 8.7). Fault trees have been applied occasionally to problems associated with space launches, mission planning and approval (Chapters 9 and 10). This results in an extensive set of analyses of the potential launch failure events and consequences.

Consequences of observed or anticipated accidents are often modeled by extrapolation from small scale tests, limited observations, simulations and scoping calculations. The goal of quantitative risk assessment is not only to identify and rank hazards, but to analyze the low probability events of high consequence. This can focus corrective action, improve management of risk factors and optimize resource allocation. These extreme events are feared most by both public and regulators. They are often used as "worst case scenarios" or extreme "catastrophic" failures that serve as the basis for conservative design and regulatory requirements.[11,12]

However, catastrophic failures are seldom observed. Therefore, their probability of occurrence and consequences are uncertain and difficult to quantify. The Three Mile Island nuclear reactor accident was this type of rare event. It occurred after 500 reactor years without a significant accident, yet was qualitatively anticipated and approximated. A severe earthquake along the San Andreas Fault, with catastrophic impacts on the San Francisco Bay area, is another example of an anticipated hazard of low probability and high consequence that is difficult to predict and control. Future levels of risk are usually predicted by statistical analyses of relevant experience, although, a complete time series and representative sampling of hazardous events seldom exists. Predictions are often based on inference, event reconstruction, interpretation and extrapolation, rather than on observed events.[11] Because industry and regulators learn to improve safety and reduce risk based on prior experience, Bayesian statistics are sometimes used to reflect the decrease in the probability/frequency of hazardous events when "learning" improves the odds.[11,13] Alternative computational methods to infer a risk profile envelope have been developed (e.g., trend analysis) that include low probability, high consequence events, when the high consequence results from a number of intermediate events and the structure of such a composite event can be analyzed and quantified.[12] However, such predicted or composite risk profiles are often controversial, as is discussed in Chapter 9, which reviews the application of Risk Assessment methods to space launch and orbiter systems and missions.

8.5 UNCERTAINTIES AND RELIABILITY

Risk Analysis is not an exact science. Despite this, it is widely used to support regulatory and industrial decision making and to allot resources. Risk analyses performed by different analysts on the same issue may lead to different results. The reason is that there are substantial uncertainties intrinsic to risk assessments deriving from incomplete knowledge and identification of potential failures, from incorrect modeling assumptions used in the quantification of hazardous events or, more likely, from the variability in the possible type, time, place and circumstances of an accident. Different (and possibly incomplete) data bases and assumed failure rates of components may be used and thus lead to discrepancies in results. Different statistical analyses of the same data base may be justified by stated assumptions and lead to further discrepancies in results. Furthermore, the choice of a certain risk analysis methodology may influence, and even determine, the conclusions of the analyses. Judgments by experts evaluating and ranking the hazards, i.e. the Delphi approach, are often subjective. Hence, the risk analysis process has inherent limitations and uncertainties which must be taken into consideration in decision making.

Tests to establish reliability of complex components or systems are usually expensive, making a minimum of tests desirable. On the other hand, true probabilities are based ideally on results from very large samples. When only a few items are tested, the results may not be truly representative. Tossing a normal coin two or three times may result in heads each time. This may lead to the erroneous assumption that the result will always be heads. The next three tosses may all be heads again, all tails or combinations of heads or tails. With more and more tests the average probability of a head (or tail) will be found to approach 0.5. The problem then arises as to how much confidence can be placed on past results to predict future performance. The term confidence level is used for this purpose. Tables have been prepared to indicate the relationships between test results, reliability and confidence. One such table is shown below in abbreviated form (Table 8-4):

TABLE 8-4. NUMBER OF TESTS THAT MUST BE PERFORMED WITHOUT A FAILURE TO PROVIDE A SPECIFIC MINIMUM RELIABILITY AT ANY CONFIDENCE LEVEL. (Ref. 6)

MINIMUM	CONFIDENCE LEVEL				
RELIABILITY (%)	90%	95%	97 1/2%	99%	99 1/2%
75	8	11	13	16	19
80	11	14	17	21	24
85	15	19	23	29	33
90	22	29	35	44	51
95	45	59	72	90	103
96	57	74	91	113	130
97	76	99	122	152	174
98	115	149	184	229	263
99	230	299	370	460	530

Since there are residual uncertainties associated with the quantification of risk, confidence limits must be placed both on failure probabilities (usually 60%-90% brackets) to reflect this uncertainty. A 60 percent confidence interval means that there is a 60 percent chance that the actual failure rate falls within the range of given estimates. A 90 percent confidence limit means that there is a 90 percent chance that real events will fall within an estimated range. Confidence limits are based on observations: if no failures occurred in 1,000 trials, there are still three failures possible in the next 1,000. If 10,000 tests were successfully completed, that would statistically correspond to a probability of three failures in 10,000 events with 95 percent confidence (i.e., a reliability of .9997). In addition, there may be large uncertainties in the consequence estimate, so that for any "best guess" point estimate, "worst case" and "best case" limits are needed.

Most assemblies and systems actually do not have constant failure rates, especially when the system does not have many components that are similar or have similar characteristics, such as large mechanical units. Instead of being exponential, the distribution of failures may be Gaussian, Weibull, gamma or log normal. The chief difference is in establishment of failure rates. Means of improving reliability as indicated above remain the same. Table 8-4 is based on the simplest assumption of a binomial distribution, where the outcome of any trial can be either failure (F) or success (S), randomly occurring with probability .5 (like tossing coins for Head/Tail outcomes).

8.6 RELIABILITY VERSUS SAFETY

Reliability Analysis often provides useful inputs to quantitative safety analysis since failure rates (observed or design goals) for safety critical components and subsystems permit the evaluation and control of adverse safety impacts. Often, to ensure safe operation, safeguards are incorporated into system engineering design, such as: redundant features; manual overrides for automatic components (valves, switches) which are safety critical and special quality assurance, acceptability and maintainability specifications. Space launch vehicles and payloads have been traditionally provided with redundancy in the in-flight destruct or other termination system and the flight control and communications subsystems (see Chapter 2, Vol.1). This ensures that a guidance failure or a failure in boost, sustainer or upper rocket stages will not lead to undesirable off-range risk exposure and that risk to the public will be avoided and controlled by the Range Safety Officer's ability to safely destroy the spacecraft on command.

Reliability data on components and subsystems are essential to predicting performance. Table 8-5 shows as an example theestimated probability that a certain number of failures will occur in the next 20 tries for a hypothetical launch vehicle, based on assumed operational performance reliability figures in the range of historical values and on a skewed binomial distribution. (See also Ch.3, Vol. 1 for published reliability figures on commercial space vehicles.)

However, it must be noted that although reliability figures feed safety analyses directly, a highly reliable system is not necessarily safer. A key issue is the trade off between reliability and safety: adding sensors and control systems to detect malfunctions in a critical subsystem may enhance safety, but decrease the overall reliability. A stick of dynamite is an example of a highly reliable, but clearly unsafe object: when triggered intentionally or unintentionally, it will explode reliably. It is unsafe because of its high energy content, its explosive potential and its low trigger threshold. Safeguards may enhance handling safety, but decrease functional reliability. In favor of the reliability of simplicity, some engineers would trade the sophisticated injection pumps in modern rockets for simple gravity fed ("big dumb") rockets.

TABLE 8-5. RELIABILITY USED AS PERFORMANCE PREDICTOR FOR A HYPOTHETICAL LAUNCH VEHICLE

	VEHICLE RELIABILITY (R)				
	0.98	0.975	0.97	0.96	0.95
EXPECTED "EVENT" (NUMBER OF LAUNCH FAILURES IN 20 TRIES)	PROBABILITY OF EVENT (P PERCENT)*				
0	67	60	54	44	36
1	27	29	34	37	38
2	5	6	10	15	19
3	<1	1	2	4	6
4	0	0	0	0	>1

An illustration of the use of binomial distribution skewed to higher probability of the "event," defined as "x failures in the next 20 consecutive launches." Note that the higher the assumed reliability, the higher the probability of "sucess" (i.e., fewer failures in 20 launch attempts).

Both human error and infrequent operational or accidental failures, can lead to catastrophic accidents with a low probability of occurrence and potentially high risk exposure. Indeed, in the case of space launch systems and operations, it is the low probability and high consequence event that would dominate the public risk exposure. The likelihood of occurrence and the public safety impacts of any accidental failure in such highly reliable subsystems and systems must be quantitatively assessed in order to appropriately define acceptable and expected levels of risk, and to regulate commercial space activities via the licensing process (see Chapters 9 and 10).

Table 8-6 shows the kind of basic component failure rates which are used in probabilistic system failure computations. These apply to all mechanical and electrical systems across industries. Similarly, human error must often be factored into estimating probabilities of systems breakdown, since operator error or judgment errors in responding to minor failures can have major consequences. Table 8-7 shows that high stress work conditions lead to more frequent human error than routine functions and operations. Human failure rates are typically higher than equipment failure rates and may compound them because of improper or incomplete operator training in recognizing critical situations or because of panic/stress response to an accident. Considerable attention has been paid to human/ machine interfaces and to crisis training of personnel. The same considerations should apply in analyzing a launch "go/no go" decision, or a command destruct decision for a space system, as for a reactor operator or a flight controller in a busy airport tower.

8.7 RISK ASSESSMENT METHODS

The adoption of an appropriate analytical technique is important to any meaningful qualitative or quantitative failure and/or risk analysis. Each risk quantification method discussed and illustrated below has its own special merits, strengths, weaknesses and an optimal domain of application (see Table 8-8). Only if sufficient empirical and statistical data are available is the probabilistic modeling

TABLE 8-6. COMPONENT FAILURE RATES

Automatic Shutdown	10^{-2}/demand
Emergency Shutdown System	10^{-3}/demand
Defective Materials (Seals)	10^{-4}/demand
Defective Pumps	10^{-3}/year
Faulty Gasket	10^{-5}/year
Brittle Fracture (pipes)	10^{-5}/year
Pipe Failure - 3' rupture	8×10^{-5}/section year
Spontaneous Failures (tanks, etc)	10^{-6}/year

TABLE 8-7. HUMAN FAILURE RATES (Ref. 18)

Task	Probability of Error/Task
Critical Routine	10^{-3}
Non-critical Routine: errors of omission and commission	$10^{-2} - 10^{-3}$
High Stress Operations.	$10^{-2} - 10^{-1}$
Responses after major accident during:	
- 1st minute	1
- to + 5 minutes	9×10^{-1}
- to + 30 minutes	10^{-1}
- to + several hours	10^{-2}

of hazardous events justified. For the very infrequent catastrophic event, a deterministic analysis of consequences (i.e., scoping calculations to estimate the type and magnitude of impacts assuming that the accident has occurred) may be sufficient in order to consider possible risk management (prevention and emergency response) and to estimate the associated sensitivity to assumptions. Deterministic consequence modeling of an unlikely catastrophic event is acceptable and even necessary whenever accident statistical and heuristic data available do not suffice to justify quantitative estimates for its occurrence and observation based scoping estimates for the magnitude of its consequences.

There are several inductive methods of risk analysis which assume a particular failure mode or failure initiating event. The effects on the system performance are then analyzed in order to infer the propagation of failures (failure chains) and to assess the sensitivity of the system operation to the

TABLE 8-8. STRENGTHS AND WEAKNESSES OF SELECTED RISK QUANTIFICATION TECHNIQUES (REF. 10)

STRENGTHS	WEAKNESSES
Preliminary Hazard Analysis (PHA)	
• May be applied during very early stages of project development. • Very straightforward to carry out. • Provides documentation of results.	• Difficult to show effects of mitigation or to prioritize the causes of one undesired outcome as does not show multiple causes of undesired event in same place. • Not particularly useful at later stages of development or for reanalysis.
Fault Tree Analysis (FTA)	
• Logical presentation of event sequences of concern. • Shows relative significance of events and causes. • Readily demonstrates effectiveness of mitigation or redesign. • Can be used in sensitivity analyses. • Can cover human errors as well as equipment failures.	• Time consuming and requires careful identification of both top events and causes. • Requires skill to handle common mode failures, dependent events, and time dependencies. • May be difficult to justify/obtain probabilities needed for quantification.
Event Tree Analysis (ETA)	
• May be used to develop critical events and consequences by starting with a single failure, or may start with critical event and develop consequences. • Orders events in time sequence in which they occur. • Displays logical relationships.	• May need FTA or some other method to develop probabilities. • Very time consuming if starting with individual failures. • May be incomplete if all events not identified. • Difficult to handle partial failures or time delays.

postulated initial failures (bottom to top). The methods listed below focus primarily on on hazard identification and on the probabilities of occurrence of hazardous events:

Inductive risk analyses methods used in industry to determine what failed states are possible include:

- The Preliminary Hazards Analysis (PHA) - This is the most general and qualitative identification and listing of potentially hazardous conditions, which is used to guide design, or the definition of procedural safeguards for controlling these. Often, PHA suffices to identify causal failure chains, possible safeguards and risk prevention options.

  The list of hazardous events to be prevented or controlled can be developed into subevents. PHA is usually carried out at an early stage of design and operations planning in order to allow both design and operational controls to be implemented in a cost-effective manner. Table 8-9 is an example of a Preliminary Hazard Analysis list of failures/malfunctions, used to identify safety critical failures and hazardous conditions and consequences, used to suggest risk control (prevention, reduction and avoidance) strategies. The PHA technique has been used primarily in the chemical and petroleum industries and in the design of critical facilities.

The PHA, although chiefly an inductive method, can also be used in deductive analysis since it is primarily a systematic and hierarchical listing of failures, accidental events and circumstances leading to potentially catastrophic or major undesirable consequences. Such listing of failure events and their enabling conditions simulates closely and is complementary to a FTA (see below) since it permits the definition of hazardous chains of events and affords insight in the initiating (i.e., causal)

TABLE 8-9. MALFUNCTIONS AND FAILURES (REF. 6)

POSSIBLE EFFECTS	POSSIBLE CAUSES
Mechanical malfunctions Equipment will not operate Vibration and noise Bearing problems	Broken part Separation of couplings Separation of fasteners Failure to release holding device or interlock Binding due to heavy corrosion or contamination Misalignment of parts Misaligned, loose, or broken rotating or reciprocating equipment or parts Broken or worn out vibration isolators or shock absorbers Bearings worn due to overloading Bearings too tight or too loose Lack of lubrication
Power source failure Complete inactivation of power dependent systems Lack of propulsion during a critical period Guidance failure of a moving vehicle Failure during flight airborne systems Inability to activate other systems Failure of life support systems Failure of safety monitoring and warning systems Failure of emergency or rescue systems	Prime mover failure Internal combustion unit – Fuel exhaustion or lack – Oxygen exhaustion or lack – Lack or failure of ignition source for chemical reaction – Interference with reaction – Mechanical malfunction – Failure of the cooling system – Failure of the lubricating system Blockage of steam, gas or water used to drive turbines Excessive wear of power equipment Mechanical damage to power equipment Poor adjustment of critical device Failure of connection to electric generator Excessive speed due to lack of control Loss of electrolyte for battery or fuel cell
Electrical system Failure Entire system inoperative Specific equipment will not operate Interruption of communications Detection and warning devices inactivated Failure of lighting systems Release of holding devices	Faulty connector or connection Failure to make connection Conductor cut Fuses, circuits breakers, or cutouts open Conductor burned our Switch or other device open or broken Short circuit Overloading

factors enabling failure. The unlikely adverse end event can also be analyzed in terms of more probable subevents, down to the common minor failures in the domain of daily occurrences.

- The Failure Mode and Effect Analysis (FMEA) - This is a more detailed analytical procedure, which is used to identify critical and non-critical failure modes. Single point (component) failures which can lead to system break down are thus identified and fixes, such as redundancies or operational bypass, are designed into the systems to prevent them. FMEA can be quantified if failure probabilities for components can be used to derive the percentage of failures by mode. Critical and non-critical effects are used for managing risk and preparing emergency response plans.

- Failure Mode Effect and Criticality Analysis (FMECA) - This type of analysis is a more detailed variant of FMEA. It is used for system safety analysis, to enable detailed assessment and ranking of critical malfunctions and equipment failures and to devise assurances and controls to limit the impacts of such failures (i.e. risk management strategies). FMECA is usually a tabular listing of: identified faults, their potential effects, existing or required compensation and control procedures, and a summary of findings.

- Fault Hazard Analysis (FHA) - This method is particularly useful for inter-organizational projects that require integration, tracking and accountability. It is typically used for space systems when numerous contractors design, test and certify various subsystems which must be integrated into a payload or a final launch system. FHA forms display in column format: the component identification by subsystem; a failure probability; all possible failure modes; the percent failures by mode; the effect of failures, up to subsystem interfaces; the identification of upstream components that initiate, command or control the failure and any secondary failure factors or environmental conditions to which the component is sensitive.

- Event Tree Analysis (ETA) - This approach is equivalent to the qualitative part of Fault Tree Analysis (FTA, see below) and is used to display the likely propagation of failures in a system. Figure 8-4 is an example of an Event Tree which is used to isolate a failure propagation sequence and identify enabling conditions which can be controlled. Event trees are used in FMEA, FMECA and FTA and require identification of all failure initiating events. Figure 8-5 is an example of an event tree for commercial space operational failures.

- Double Failure Matrix (DFM) - This method is used to list single vs. double subsystem failures, only after failure categorization by effects on the system have been completed. Namely, Fault Categories I-IV correspond to the severity of impacts on the system: I. negligible, II. marginal, III. critical and IV. catastrophic. Then, for each subsystem the component failures and the corresponding fault categories are listed in matrix form to determine how many ways a certain hazard category can occur (single and multiple failure modes).

- Hazard and Operability Analysis (Haz-Op), or Operability Hazard Analysis (OHA) - This is another method of safety analysis widely applied in designing complex chemical facilities.[10] This procedure involves the examination of design, piping and instrument diagrams (P&ID) and operation flow charts in order to ask a "what if" question at each node. What would happen if a deviation from normal operations and design conditions occurs at this point (Figure 8-6)? This method is equivalent to the FMEA analysis in the sense that it permits

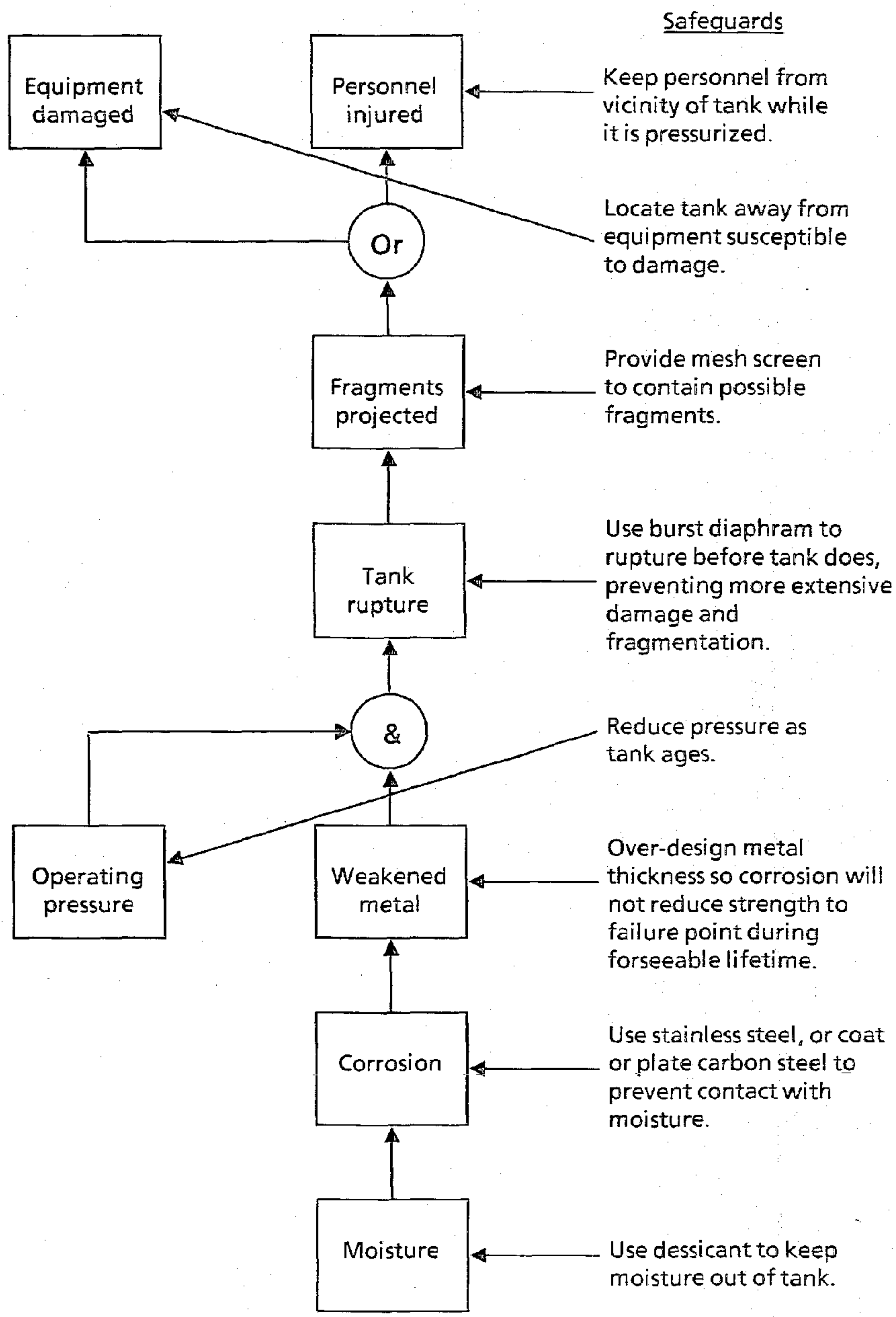


FIGURE 8-4. SEQUENCE OF EVENTS LEADING TO RUPTURE OF A PRESSURIZED TANK (Ref. 6)

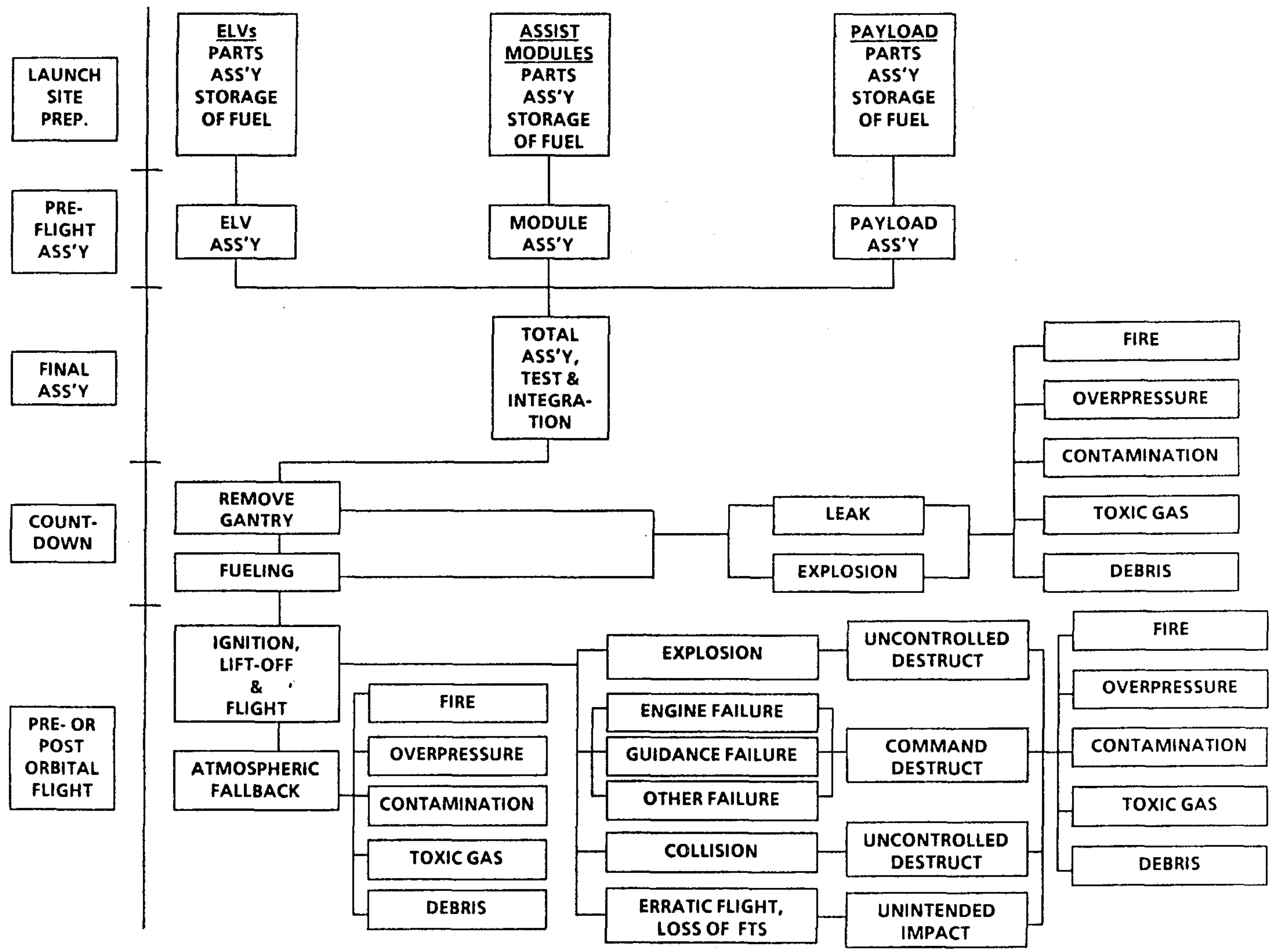


FIGURE 8-5. EVENT TREE FOR COMMERCIAL SPACE LAUNCH OPERATIONS

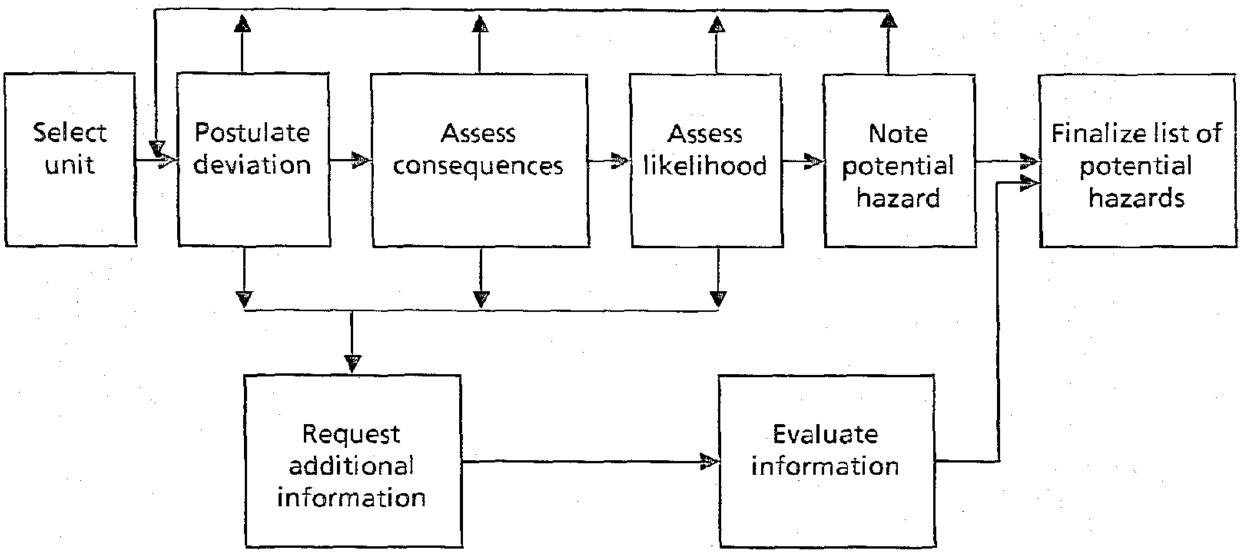


FIGURE 8-6. HAZ-OP METHODOLOGY (REF. 10)

identification of critical failure initiators, single point failures, malfunction chains and their effects on other parts of the system propagation of failure. Design flaws that require safeguards to insure operability (double valves, bypass redundance logic, manual overrides, etc.) can thus be uncovered. This method is used both in pre-design and post-design analyses to achieve design verification, set acceptance criteria, meet objectives in system operation, provide procedural modifications to ensure safe operation and emplace monitoring of safety critical items. A Haz-Op variant is LAD (Loss Analysis Diagram), used to compare design options and determine the risk acceptability levels or safety margins in design. Similarly, contingency analysis is used as a complementary risk analysis method to Haz-Op, in order to manage risks, when loss of control or a critical accident occurs.

In contrast to the above approaches, deductive risk analysis methods require reasoning from the general to the specific: A system failure is postulated and the subsystem failure modes leading to it are analyzed and broken down to the terminal or initiating failure event level ("top to bottom" or "top down" approach). Most accident investigations are of this type and are used to determine how a system failure can occur.[8] This includes:

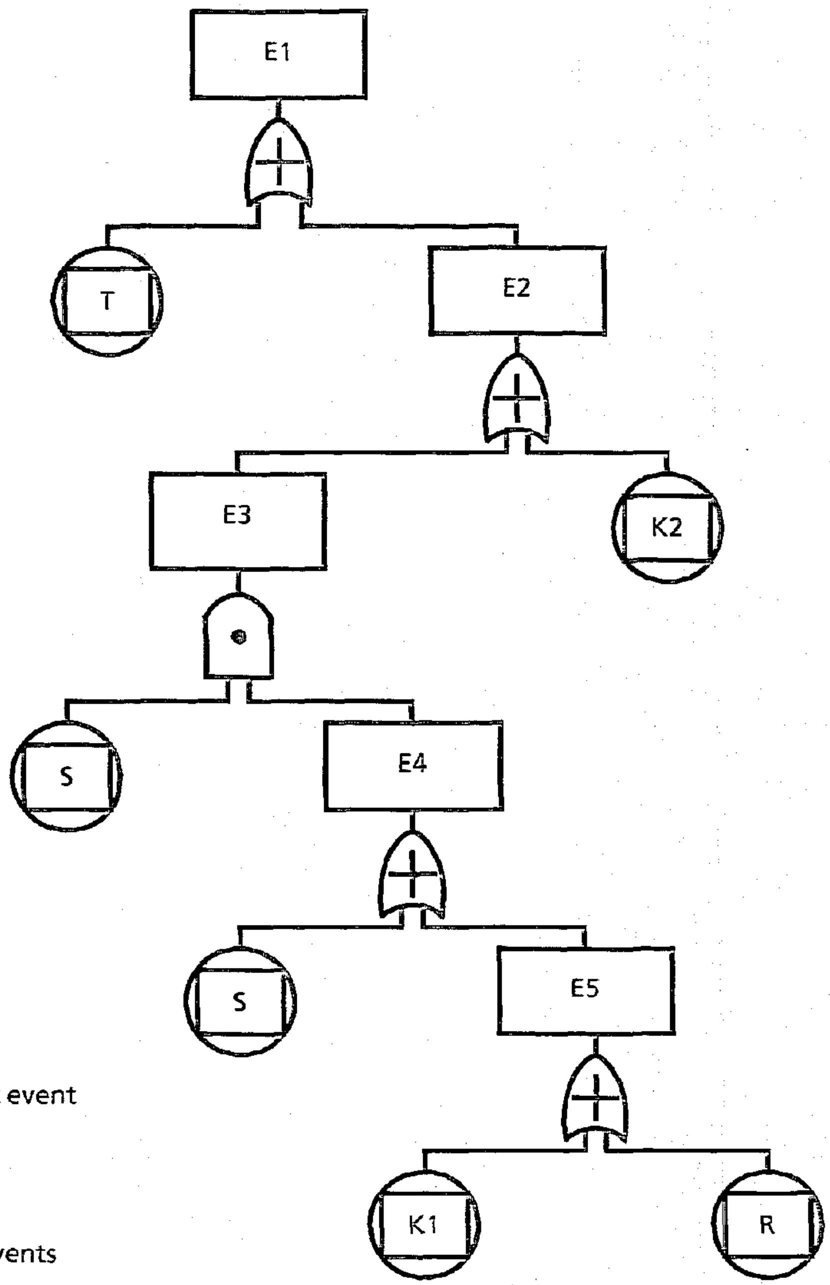
- The Fault Tree Analysis (FTA) Methodology for Hazard Assessment - The FTA technique is a logical method for display and analysis of the hierarchical linkage and propagation of failure events leading to the adverse end result, placed at the top of the "tree". Branches in this logic tree represent alternative failure paths leading to the stipulated end event and display interdependencies of failures. A staged fault tree (Figure 8-7) allows the definition of intermediate levels of the events and conditions that are necessary or enable failures to propagate to the top of the tree. The intermediate failure events may, in turn, result from the aggregation of lower level failures from system- level down to subsystems and component failures. The bottom levels display the failure initiating or tree terminal events. Critical factors and interrupt modes for failure chains can be identified and quantitatively examined. The nodes of the fault tree represent logic AND or OR gates.

The AND gate represents the simultaneous occurrence of conditions or events necessary to result in failure propagation up the tree. An OR gate indicates that each individual failure event entering is capable of leading to higher level failures. Careful consideration must be given to the independence or mutual interdependence of events entering a particular logic gate to insure the correct use of joint and conditional probability concepts. In the case of ELV launch or orbital failures, a fault tree may be used to highlight single point (critical) failures and "common cause" (not independent) failures which must be "designed out" by redundancy or greater safety margins. Clever analysts use "exclusive OR gates," by defining mutually exclusive sets of failure events or aggregating lower level failure events into complementary groups to facilitate estimation of probability at each node of this event fault tree. FTA can be used both for qualitative, and for quantitative analysis of hazards. However, qualitative results must be combined with accurate failure rate data in order to achieve meaningful quantitative results.

Assuming independence of failures, there are five "minimal cut sets", i.e., intersection of events, whose probabilities are added at OR gates (provided that individual failure probabilities are very small so that probability products are negligible compared to their sum), and multiplied at the AND gate.

$$
\begin{aligned}
E_1 &= T + E_2 = \\
&= T + (K_2 + E_3) = \\
&= T + K_2 + (S \cdot E_4) \\
&= T + K_2 + S \cdot (S_1 + E_5) = \\
&= T + K_2 + (S \cdot S_1) + S \cdot (K_1 + R) = \\
&= T + K_2 + (S \cdot S_1) + S \cdot K_1 + S \cdot R
\end{aligned}
$$

The minimal cut sets are T, K_2, $S \cdot S_1$, $S \cdot K_1$ and $S \cdot R$ (two singles and three doubles). The largest contribution to the probabilities will come from the single point failures T and K_2 (critical failures), since the small probabilities of occurrence for the individual failure events, S, S_1, K_1 and R, the product of their probabilities will make a very small, and possibly negligible, contribution to the final event probability. Probabilities of simultaneous failures at AND gates necessary for a higher level failure to occur, may be multiplied in some approximations only if



Legend: Faults
E_1 - Top event
E_2 ...E_5 intermediate fault event
R = primary failure
S, S_1 = primary failure
K_1,K_2 = primary failure
T = primary failure
Circles denote terminal events

Rectangles denote fault events requiring further branching development and analysis into sub events.

- OR gate with two independent input events, either of which can lead to failure, but which cannot occur simultaneously (plus denotes additive probabilities).

- AND gate specifies a causal failure where input faults jointly cause the output fault and the dot denotes a product of probabilities (both must occur for failure).

FIGURE 8-7. BASIC FAULT TREE SCHEMATIC (REF. 8)

conditional probabilities for interdependent failures are subtracted and the correct dimensionality is preserved. Usually, probabilities of independent events at OR gates are added, if $P < 1$. Correct dimensionality must be observed for all types of logic gates.[8]

Each branch of a failure event tree must be quantified in a consistent manner using either frequency units (1/time dimensions, rate per year, per hour or per event) or normalized dimensionless probabilities. By using observed or projected/expected values for the frequency or probability of various failure modes and by analyzing how they occur, the likelihood of each hazardous event can be quantified. Risk is the product of this probability (or frequency) by the consequence magnitude of the undesirable event. The correct probabilistic dependencies (conditional, joint, mutually exclusive) for the occurrence of failure events of the lower branches permit their quantitative aggregation at gates and up the tree. References 1, 3, 7, 8 and 10 discuss and illustrate the application, use and practice of FTA and other Probabilistic Risk Analysis (PRA) methods, such as FMEA, in industry and Government.

The NRC and DOE have made extensive use of PRA in analyzing, licensing and regulating the operation of nuclear power plants; in prioritizing generic nuclear industry, transportation and waste disposal safety issues and in performing environmental impact analyses.[14-16] DOD has also used PRA to develop and test nuclear weapon systems. PRA is a comprehensive and integrated analysis of failures capable of revealing their interrelationship and their likelihood. Thus, in spite of its uncertainties, high cost, effort and limitations, PRA has proven useful to regulators of technological risk both to highlight gaps in knowledge and areas of research need and in directing the industry and regulatory efforts towards redress of high leverage safety problems. PRA's have aided in formulating safety goals, criteria and defining risk acceptability levels and numerical compliance targets for industry.

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9. APPLICATIONS OF RISK ANALYSIS TO SPACE LAUNCH OPERATIONS

9.1 LAUNCH RISK ANALYSIS OBJECTIVES

Risk Analysis is not an end in itself, but rather a means to accomplish other goals: the identification of hazards and the assessment and quantification of risk provide insight to the overall acceptability of a program, such a commercial space launch campaign, from operational, regulatory or societal viewpoints. If the associated risk level appears unacceptably high to the public agency sponsoring or regulating the activity, the analysis can provide information needed to control and reduce the risk. The whole Range Safety Control process (see Ch.2, Vol.1) is predicated on risk avoidance, minimization of accident impacts and the protection of population centers (see also Ch. 10). Risk values related to space-launch activities may be generally categorized in two ways: (1) the probability of vehicle failure, including all possible failure modes, that could lead to debris impact events and their probabilities; and (2) consequence estimation, i.e., expected casualties or damage. The probability of debris impacts generally means that at least one object impacts in a specific area. The casualty estimation generally used is one of two types: (1) the probability of casualty, defined as the probability of one or more persons sustaining an injury; or (2) the expected number of casualties, defined as the number of persons expected to sustain an injury as a result of at least one object impact in a specific area. These concepts have also been discussed and illustrated in the context of Range Safety destruct actions (Ch. 2, Vol.1 and Ch. 10) and re-entry hazards (Ch. 7, Vol.2).

The following is a list of general uses and applications of Risk Analysis in the context of space mission planning, approval and implementation:

- A risk study can serve as a tool in the total decision making process for the Range or the sponsoring organization.
- Excessive risk may reveal the need for a Flight Termination System (FTS) or other program restrictions (e.g., restrict land overflight or launch azimuths).[29,32]
- Results are a tool to help underwriters price commercial space insurance.
- Results may indicate the requirement that an existing or pre-designed FTS or other critical ELV system be redesigned, if such a redesign can significantly reduce these risk levels via greater safety margins or introducing redundancies.[33]
- Results may indicate the need for evacuation of Range personnel, enforcement of roadblocks, restricted sea lanes or airspace, movement of critical equipment, call-up/purchase of additional real estate or justification for currently controlled land.[2 b]

- Results might show the necessity to modify the support plans for other Range support elements permitted within the evacuated area, i.e., manned optical tracking sites.
- Results can be used in the development of ELV flight safety operational support plans to include procedures, destruct criteria and whole vehicle versus destruct case (many fragments) impact decisions.[10,11]
- Results can be used to alert the Range or Sponsor management to excessive on-site or public risk exposure levels for given launches or total programs. It is then the decision of management on which course to proceed.[17]
- Results might identify launch scenarios and patterns that require mission operational procedure changes or hardware redesign/modification to allow the selection of less hazardous options, based on cost/benefit or operational constraints and priorities.[18]
- Results may indicate the need to construct new facilities in cases where it is not acceptable to use existing facilities.[20]
- Results might reveal the need and advantage of providing positive protection for nonevacuated personnel (shelters, barricades, bunkers, blockhouses, etc.) and critical equipment required in the evacuated area.[20]
- Results can be used to establish and define limiting criteria which may be used both quantitatively and qualitatively. Impacts of single launches or cumulative impacts of space launch programs can be compared in this manner.[19,32]
- Risk studies can provide documented evidence that specific hazards were considered in an objective and rational manner in developing operation plans.[8-13]
- "Risks to launch" results identify the reliability of the Range support equipment and personnel and can be used for the following purposes:[19, 32]
    a. Identify high risk from inadequate Range support elements and, therefore, assist in increasing total reliability and reducing hazards involved in launching.
    b. Increase Range operational safety and supportability.
    c. Increase Range capability and attractiveness to potential users.

A general method that satisfies all possible analytical problems related to space operations does not exist, as discussed in Ch. 8. Historically, the National Ranges have developed their own computer programs for risk studies and analyses, as appropriate to specific tests, launch vehicle systems or Range operation problems. Although no standardization exists at present between the Ranges regarding methodology, computer programs and analytical tools (mainly because of different siting and demographics, but also because of specialized uses of each Range), the major types and elements of space risk analysis do recur. Moreover, there are technology transfer and standardization efforts in progress at ESMC and WSMC. A typical Risk Analysis requires five basic categories of data:

1. Systems failure modes and their probabilities.
2. Impact probabilities and distributions resulting from failures or normal launches.
3. A measure of lethality of impacting debris.
4. Location and nature of population and structures placed at risk by the mission.
5. Launch plans, subject to Ground Safety and Range Safety constraints.

Various elements of these categories may be considered in development of a Risk Analysis for a space launch vehicle, mission and/or operation.

The end result of a Risk Analysis for a specific launch and orbital mission is valid only to the degree of reliability and completeness of the inputs and their applicability to a given launch vehicle or site. A result valid for one Range may be meaningless for another, because flight corridors, destruct criteria and impact limit lines are designed to be site-specific and are tied to the launch azimuth. Risk Analysis results may have orders of magnitude uncertainties, since they generally reflect compounded uncertainties in both initial and boundary conditions, i.e., in assumptions, modeling simplifications, approximations and possible errors of omission in the anticipated failure modes and times. Risk studies, as applied to date to space operations, have been used as aids in the decision-making process in conjunction with other factors (proven Range capability, experience, precedent, national interests and priorities, etc.). Therefore, there are no general, uniform and firmly established acceptable risk levels for space operations,[1] although policy decisions and risk acceptability guidelines have often been based on matrix-type risk assessments (Ch. 10).[3-6]

Several mission agencies have developed such matrix-type risk classification, ranking and evaluation procedures, which facilitate the objective definition of acceptable and unacceptable ranges of risk. The formal DOD risk matrix for space launches is illustrated in Ch.10.[5] The DOD qualitative hazard probability classification ranges from Level A (frequent), B (probable), C (occasional), D (remote), to E (improbable). Similarly, the consequence severity categories, which account for damage, injuries or both are: I, catastrophic; II, critical; III, marginal; and IV, negligible. Hazard analyses attempt to rank failures and accidents in a two-dimensional probability/consequence matrix and assign a hazard index to each accident accordingly (e.g. 1A, 2E, 4D). Then these can be judged acceptable, undesirable or unacceptable according to suggested criteria.[3] The logic flow of a general risk assessment procedure, as it typically applies to DOD space operations, is shown in Figure 9-1.[16]

NASA has, however, established explicit launch safety criteria and numerical risk acceptability goals, as detailed in Sec. 9.2.[7] NASA uses a mishap (or accident) severity classification consisting of three

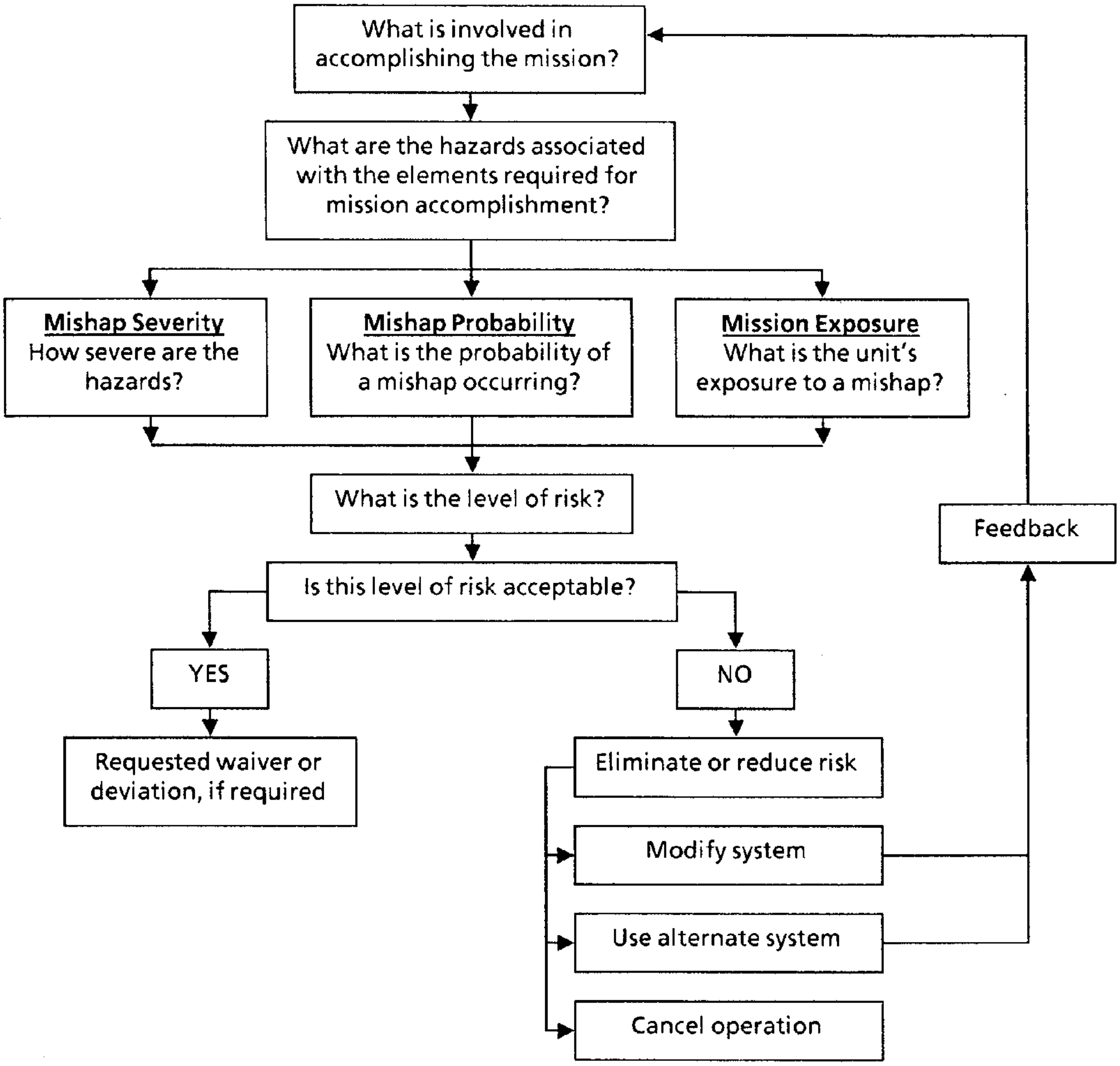

FIGURE 9-1. GENERALIZED RISK ASSESSMENT PROCEDURE (Ref. 16)

hybrid categories: A - causes death, damage exceeding $500,000 or destruction of space hardware and/or spacecraft; B - causes permanent disability to one or more people, damage valued at $250,000-500,000; C - causes only occupational injuries and/or < $250,000 damage.[7a-c] NASA has traditionally required Safety Assessment Reports (SAR) for all missions that may deviate from proven safety procedures and set safety criteria and standards.

DOE has also developed and used extensively risk ranking matrix methodologies, that combine and trade off the frequency and the severity of an event. However, the severity of consequence classes, A, B and C from worst to least, differ by loss type (fatalities, property loss, or environmental pollution effects). The accident frequency scale ranges from probable (1-100 years return period), to reasonably probable (100-10,000 years), remote (10 thousands to ten million year) and to extremely remote beyond this return period for the accident or event. Note that the probability of an event corresponding to a 100 years return period is 10^{-2} per year. The matrix risk ranking scheme permits first order (probable and severe) risks to be defined, down to fourth order (remote - C, or extremely remote -B events).[37]

9.1.1 System Failure Modes and Probabilities

Launch Vehicle physical data used may include:

- Propellants
- Explosive/fuel chemical properties
- Fragmentation characteristics
- Mass
- Shape
- Ballistic coefficients
- Flight dynamics
- Flight Termination System (FTS)
- Guidance and control
- Stage burn times and separation characteristics
- Lethality of debris, as represented by the Lethal Area

The failure modes and associated probability of failure are required if other than a normal launch is addressed.[9,10] Estimates for failure mode probabilities are typically based upon knowledge of the vehicle's critical systems and expert assessment of their reliability combined with historical data, when available.[8-11,17,18] The single point (critical) failure systems, such as the FTS, are designed, tested and certified to very high reliability standards: at WSMR the FTS reliability quoted for a non-redundant FTS required for a typical sub-orbital research or sounding rocket system is .997 at a 95% confidence level. However, higher reliabilities with failure probabilities of 10^{-6} apply to redundant FTS systems required for large ELV's. Typically, FTS designs are required to be "single fault tolerant" i.e., redundant.[6]

The total probability of an ELV operational failure includes contributions from all foreseeable failure modes which can lead to either thrust termination or malfunction turns. The occurrence of failures during a critical time interval, such as the boost phase or stage separation, permits the estimation of failure rates versus time into flight. Illustrative figures for the two major failure modes for Titan 34D as a function of time into flight are given in Table 9-1. These figures are based on an analysis of past launch performance data for the Titan family of vehicles, corrected for learning, i.e., the improvements in manufacturing, assembly and operational procedures which take place after a failure is diagnosed, analyzed and fixed.[38,39]

TABLE 9-1. TITAN III/34D FAILURE RATES USED AT WSMC*

FLIGHT TIME (sec)	KEY FAILURE RATE ($SEC.^{-1}$) MALFUNCTION TURN	THRUST TERMINATION
0	1.93×10^{-5}	1.93×10^{-5}
60.4	1.93×10^{-5}	1.93×10^{-5}
181.5	1.93×10^{-5}	1.915×10^{-4}
258.5	1.93×10^{-5}	1.915×10^{-4}
259.5	3.14×10^{-5}	9.53×10^{-5}
260.5	6.27×10^{-5}	1.93×10^{-5}
476.0	6.27×10^{-5}	1.93×10^{-5}

* - Based on VAFB/WSMC and historical launch failure data, Reference 39.

9.1.2 Impact Probabilities

The regions or areas exposed to launch operations or accident hazards must be identified (see Ch. 4). These may be subdivided into smaller sections, critical locations of people or buildings that are specified for subsequent risk calculations. All risk analyses require estimates of the probabilities of debris/fragments from failed vehicle impacting within hazardous distances of personnel or structures in the region.[17,23] The probability of an impact, P_i, for a public area requires consideration of all failure chains which could endanger it and always implies an FTS failure whose probability is P_f, given that a critical vehicle failure of probability P_v has occurred.

The design and engineering associated with the development of a system is geared to produce a properly functioning vehicle. As a consequence, there are generally no data defining vehicle performance characteristics after a critical failure has occurred, except environment definition and vehicle response scenarios assumed. These data are required for meaningful risk assessment. To provide such data, several computer models discussed below in Sec. 9.2 have been developed to simulate vehicle responses after a given gross failure mode has occurred.[19] These computer models are used as part of the computational process for generating debris impact probability density functions. These models combine, statistically and dynamically, well defined vehicle data with expert engineering estimates to predict vehicle performance after a failure occurs (e.g., Table 9-1). Sometimes failures that occurred during design verification and system tests can be used to infer in-flight failure behavior. Also, Mishap Reports, which are based on failure diagnostics and accident investigations, help to refine these computer programs or their external data files with field data.[33,34] Failures possible during each launch and flight phase must be considered separately, in order to isolate those with the potential for public safety impacts.

9.1.3 Debris Lethality

An important aspect of the vehicle data problem that must be addressed prior to performing risk calculations is to determine what occurs after vehicle failure and fragmentation (whether on command or spontaneous) leading to ground impact. The number of fragments, their sizes and shapes will ultimately define the hazard and casualty area for a given vehicle or fragment impact (Table 9, Ref. 37b). Debris are characterized by their size, mass, area and ballistic coefficient to determine if they survive re-entry and their terminal velocity at ground impact. The data items which are often developed for this part of the problem include: an impact energy distribution budget, secondary explosive energies available (if any) at impact, secondary fragments which may result from impact (splatter effect) and ricochet probabilities and characteristics.[20,22] Also, the likelihood, severity and extent of toxic vapor clouds, pool fires and blasts are used to calculate hazard areas for the various hazard mechanisms (see also Ch. 5, Vol. 2 and Ch. 10).

9.1.4 The Meaning of Casualty Expectation

The quantity most frequently employed to evaluate the risk associated with the testing and operation of a space launch system is called casualty expectation, E_c. This quantity corresponds to the expected or mean number of casualties or injuries if an ELV is launched according to a specific mission plan. The specific approach to compute casualty expectation is adapted by the National Ranges to fit their specific problems and launch situations.[17-23] In general, E_c is obtained by considering the following quantities:

- The area, A, in which debris impacts can occur, partitioned into A_i subsets of areas.
- The fragment impact probability density (P_i) on A_i produced by a given system failure.
- The hazard area, A_{Hi}, associated with an impact on A_i, is the effective casualty (lethal) area for an impacting piece of debris.
- N_i, the number of people in A_i at risk from debris impacts.
- V, vulnerability, i.e., the likelihood that a structure (hardened or not) within A_{Hi} can be penetrated by debris or that a person can be injured as the result of impact. This is only explicitly factored when estimating risk to off-shore oil platforms and on-site facilities.[17,20]

These quantities are then used in an equation of the form

$$E_c = \sum_i P_i \frac{A_{Hi}}{A_i} N_i$$

The E_c estimate, as a measure of risk for a given test, is often calculated by summing the risk over the hazard area for the test with each element of the sum. These are weighted according to the probability, as a function of time after launch, of the i-th failure mode which may require destruct or lead to vehicle fragmentation (Table 9-2). It must be noted that Ec is not the probability of a casualty, because it can be >1 in special cases. For illustration of the difference, in case of one accident per 1,000 with an average of 5 casualties per accident, Ec is 5/1,000, but the probability of a casualty is 1/1,000.

9.1.5 Population/Structures Data

The major purpose of a launch risk analysis is to determine the magnitude of hazards to personnel and structures posed by a launch and/or total program. Public risk exposure is of concern primarily near the launch site and during the first minute after launch, when, if the vehicle fails, it may veer towards populated areas protected by impact limit lines. The FTS must also fail (a double failure must occur) in order to violate the destruct limits designed to protect the public. The probabilities of such double failures are typically very low, on the order of 10^{-6} to 10^{-8}.[37] Locations of buildings and structures and the distribution of population throughout the area must be known, as well as other facts, including:

- Sheltering capability of occupied structures, i.e., the ability to withstand debris impact and protect against overpressures from explosions or impact kinetic energy conversion;
- Frequently, population distributions may be functions of the time of day or week and may be significant in risk tradeoff studies;
- Risk levels can be directly affected and controlled to some extent by population control, sheltering, Range clearance or by preventing people from entering these areas (e.g., road-blocks).

Based on such an analysis combined with mission profile constraints, the Impact Limit Lines (ILL) beyond which the vehicle and its fragments should not impact are determined for each launch to protect population and structures. Infringement of the ILL warrants a positive destruct action (see Ch.2, Vol.1).

9.1.6 Launch and Mission Planning

The actual implementation of operational plans under launch conditions ultimately determines the actual risk exposure levels on and off-site.[11-13,18] Integral to the analysis are the constraints posed by the following:

- Launch area/Range geometry and siting
- Nominal flight trajectories/profiles
- Launch/release points
- Impact limit lines, whether based on risk to population/facilities or balanced risk criteria.
- FTS and destruct criteria
- Wind/weather restrictions
- Instrumentation for ground tracking and sensing on-board the vehicle
- Essential support personnel requirements.

The Range Safety Group (or its equivalent) typically reviews and approves launch plans, imposes and implements destruct lines and other safeguards, such as NOTAM's (Notice to Airmen), Air Space Danger Area notifications and radio- frequency monitoring (see Ch.2, Vol 1).

The launch (normal and failure) scenarios are modeled and possible system failure modes are superimposed against the proposed nominal flight plan. Hazards and risk resulting from all known or hypothetical failures are summed in the overall E_c for the launch. A range of values (risk envelope) rather than a single probability or casualty expectation value is determined. The hazard to third parties is dependent upon the vehicle configuration, flight path, launch location, weather and many other factors (see Ch.5, Vol.2). It should be possible to tabulate casualty expectations and impact probabilities for a particular range, vehicle and typical flight path, but this information is not easily available in the public domain presently.

9.2 LAUNCH RISK ANALYSIS TOOLS.

9.2.1 Pre-launch Safety Requirements.

Any contractor or launch vehicle manufacturer using a National Range must comply with extensive safety requirements,[4-6] and submit sufficient data regarding the mission trajectory and vehicle performance to support the mission safety evaluation, operational planning and approval.[8-12] A Blast Danger Area around the ELV on the launch pad and a Launch danger Area (a circle centered on the pad with tangents extended along the launch trajectory) are prescribed for each ELV depending on its type, configuration, amount of propellants and their toxicity, TNT equivalents, explosive fragment velocities anticipated in case of an accident, typical weather conditions and plume models of the launch area.

The list of safety documents that a Range User must comply with is a comprehensive set of Ground and Range Safety requirements.[5-7,16] The scope of the effort involved to apply them to mission analysis and approval is well illustrated in a four volume Integrated Accident Risk Assessment Report (IARAR), which includes quality assurance and certification of critical components and subsystems, electro-explosives, hazardous propellants and chemical information, vehicle description and payload/system safety checks.[8] In the case of man-rated space systems, like the Shuttle, the customary safety requirements and the lengthy lead time required for mission planning and approval become even more cumbersome.[29-32] More typical are the mission approval documentation submitted to the Range, such as the Flight Plan Approval and Flight Termination reports illustrated by Refs. 10-13 and 15.

A Flight Safety Plan and supporting data must be supplied by the User to the National Range, prior to mission approval and operational planning.[36] Each launch is evaluated based on:

- Range User data submission requirements from the hazard analysis view point;[18,22]
- launch vehicle analyses to determine all significant failure modes and their corresponding probability of occurrence (FMEA's and Reliability Analyses);[9a,b]
- the vehicle trajectory, under significant failure mode conditions, which is analyzed to derive the impact probability density functions for intact, structurally failed and destructed options;[11-13]
- the vehicle casualty area based upon anticipated (modeled) conditions at the time of impact;[10,13]
- computed casualty expectations given the specific launch and mission profile, population data near the Range and along the ground track.[10,15] Shelters may be provided, or evacuation policies adopted, in addition to restricting the airspace along the launch corridor and notifying the air and shipping communities (NOTAM) to avoid and/or minimize risks;
- An Accident Risk Assessment Report (ARAR) prepared to identify hazards of concern, causes, controls and verification procedures for implementing such controls.[8]

The ESMC and WSMC Range Safety Requirements specify the data submissions expected from Range Users to enable hazard assessments prior to granting launch approval, including:

- determination of significant failure modes and derivation of impact probability density functions (PDF);

- evaluation of casualty area based on vehicle break-up analysis;
- computation of dwell times over land; impact probabilities; casualty expectations based on land area, geography and population densities;
- sample calculations and documentation.

Missions involving nuclear power packs or payloads must qualify based on very stringent safety criteria and are approved only after review by an Interagency Nuclear Safety Review Panel (INSRP). Detailed risk assessments have been performed by NASA, DOE, DOD and their contractors for the INSRP prior being allowed to launch satellites with nuclear power sources such as Radioisotope Thermal Generators (RTG) on- board the STS.[25-28]

9.2.2 Risk Models and Safety Criteria Used at National Ranges.

The Range Safety Group, Range Commanders Council (RSG/RCC) has reviewed a number of the computer models used by five of National Ranges (including the White Sands Missile Range - WSMR, Western Space and Missile Center - (WSMC), the Pacific Missile Test Center - PMTC, US Army Kwajalein Atoll - USAKA, and the Armament Development Test Center - ADTC) to assess launch-related risks to on-Range personnel and the public.[1] Different models and computer codes are used at the Eastern (ESMC) and Western (WSMC) Test Ranges, and at the NASA/GSFC Wallops Island Launch Facility (WFF) because launch vehicles, mission objectives and site specifics vary.[7, 18, 19]

The evaluation of launch associated hazards is based on Range destruct criteria designed to minimize risk exposure to on and off-Range population and facilities. Computer models are used to simulate missions for optimization and approval or run in real time for Range Safety Control Officers to monitor flight performance.

The DOD Ranges do not have published requirements for acceptable levels of public risks, presumably because national security interests can take precedence in testing new launch systems and launching defense payloads and spacecraft. Since launch risk exposure to the public is primarily controlled in real-time by the Range Safety personnel rather than the Range User, the residual and uncontrollable hazards to the public are re-entry hazards due to failures to achieve proper orbit and premature re-entry of the payload.

The NASA/WFF Flight Safety Plan, compares the risks associated with a specific mission to "acceptable risk criteria," such that:

- casualty expectation $\leq 10^{-7}$ for planned or accidental impact and re-entry of any part of the launch vehicle over any land mass, sea or airspace;

- probability of impact with potential damage to private property $\leq 10^{-3}$ (unless an SAR is prepared and approved or a waiver is obtained);
- probability of impact with flight support aircraft (for meteorological monitoring, or tracking support of $\leq 10^{-6}$ (note that other aircraft are excluded by NOTAM and airspace restrictions);
- probability of impact with ships and boats within the impact area (inside a 50 mile radius from the launch points) of $\leq 10^{-5}$. (Some Ranges observe a 20mi. radius;[37b] Wallops Flight Facility surveys out to 100miles.[40])

From 1961 to 1983, Wallops has experienced 14 launch failures out of over 10,000 sub-orbital launches of sounding rockets, resulting in an observed land impact probability of 2.8×10^{-3}. Of these, only three impacted outside the launch site area (i.e., $P = 6 \times 10^{-4}$). Assuming an average population density of 64 per sq mi., the casualty expectation based on this observed vehicle failure rate is 8×10^{-9}. Similarly, for debris dispersal over water, a ship traffic density of 2.6×10^{-5} per sq. nmi per day was used, resulting in an expected 3.7×10^{-7} probability of a sustainer impacting a ship. For comparison, Wallops threshold ship-impact probability criteria are $\leq 10^{-5}$, corresponding to 20x increased allowance for ship impact.

Range Safety Reports, Safety Analysis Reports (SAR's) and other such probabilistic Hazard Analyses must be prepared by Range Users for Mission Approval at most National Ranges whenever a new launch vehicle configuration (e.g., a Titan with an IUS or Centaur upper stage), an unusually hazardous payload (e.g., a nuclear powered spacecraft) or a trajectory with land overflight are involved (i.e., whenever "deviations" from approved safe procedures, vehicles and programs are filed). Similar reports are needed for US-sponsored launches from foreign territories. Either the User submits the data for the Range to carry out its own hazard analyses or the User prepares such a document on request.[6]

Safety Assessment Reports (SAR's) were typically prepared by NASA GSFC/Wallops Flight Facility (WFF) for sub-orbital launches from foreign territory. Two references are representative of the types of launch hazards of concern and the NASA approach to risk assessment: The SAR for Project CONDOR involved launches in 1983 from Punta Lobos, Peru, using Taurus-Orion, Terrier Malemutes, Nike-Orion, Black Brants and similar sub-orbital vehicles to launch retrievable atmospheric sounding research payloads.[7e-g]

Range Safety Guidelines minimize post-launch risks to the public by imposing a number of restrictions: e.g., no land over-flight corridors are selected if it is possible to have launches and flight paths over water. However, for land-locked launch sites such as WSMR, strict overflight criteria restrict both land and airspace corridors to on-Range and Extended Range areas.[2] There are no intentional off-Range land impacts permitted for any normally jettisoned booster and sustainer casings and sufficient safety margins are provided within the destruct corridor to avoid impacts on population centers by accidentally or intentionally generated debris. For WSMR launches, typical observed limits on risk to nearby population centers are land impact probabilities of $< 10^{-5}$ on-range and $< 10^{-7}$ off-range, resulting in casualty expectations of $< 10^{-7}$ to 10^{-9}.

Models, run sequentially or in parallel, are designed to compute risks based on estimating both the probabilities and consequences of launch failures as a function of time into the mission. Inputs and external data bases include data on mission profile, launch vehicle specifics (e.g., solid or liquid rockets, stages, configuration), local weather conditions and the surrounding population distribution. Given a mission profile, orbital insertion parameters and desired final orbit, the risks will vary in time and space (see Ch. 10). Therefore, a launch trajectory optimization is performed by the Range for each proposed launch, subject to risk minimization and mission objective constraints. The debris impact probabilities and lethality are then estimated for each launch considering the geographic setting, normal jettisons, failure debris and demographic data to define destruct lines to confine and/or minimize potential public risk of casualty or property damage.

The National Ranges use either a circular or an elliptical footprint dispersion model to analyze vacuum and wind-modified instantaneous impact points (IIP) from both normal stages jettisoned during launch and launch debris (failure or destruct).[(1)] The debris dispersal estimates generally assume bivariate Gaussian dispersion distributions.[(19,21)] Risk contours are estimated as impact probabilities or casualties expected per unit area centered on the IIP (nominal impact points) or on a specific site (land, community or Range) of interest. All these models are similar in approach, but quite site-specific in the use of databases, which depend on Range location and on the geographic area and associated population distribution at risk. The models may be run either as simulation to assist in analyzing and selecting launch options, or can be run in real- time, to monitor a launch operation.

The information and risk computation logic flow depicted in Figs. 9-2, 9-3 are used in a computer program developed to calculate relative risks to population centers along the flight corridor ground-track, namely the LARA - Launch Risk Analysis program and its later upgrades.[(19,21)] The LARA program is in use at WSMC and PMTC and is being introduced at ESMC. Figure 9-4 shows a sample real-time debris footprint display monitored by Range Safety Officers at WSMC during each launch operation. It is based on computed and wind-corrected trajectory and LARA impact patterns moving with the tracked vehicle and their position relative to the fixed, prescribed destruct and impact limit lines. If the failed vehicle encroaches these lines, a destruct decision must be made or withheld according to clearly formulated destruct criteria.

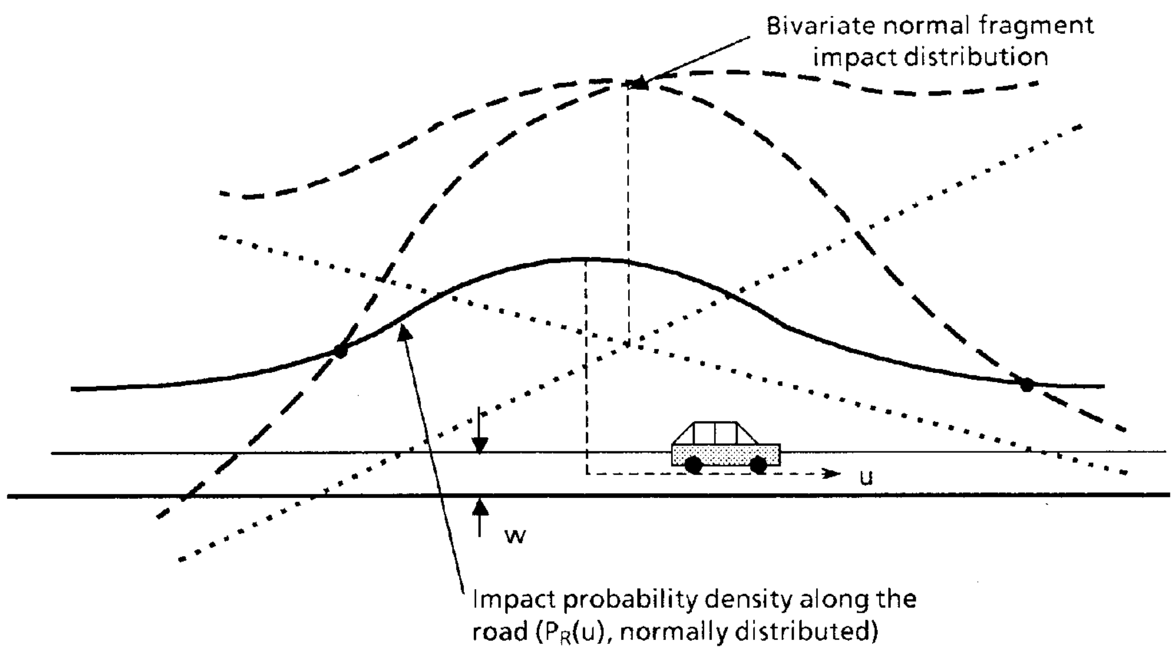

FIGURE 9-2. IMPACT PROBABILITY COMPUTATION FOR ROADS

Since WSMR is a land-based Range, safety considerations are particularly important in authorizing tests that might endanger the public. Computer models in use at the Range support pre-mission simulations of normal and failed flights, as well as real- time tracking and destruct decisions based on vacuum and drag corrected IIP's. The library of risk computation and utility codes used by Range Safety include: SAFETY.SITE (generates scaled maps of the range and tracking installations), SAFETY.DMA (converts maps to desired coordinate scale), SAFETY.GIP (predicts both vacuum and drag corrected impact coordinates) and several other external modules for population data and impact point prediction. The WSMR Hazard Analysis method and its application to launches of sub-orbital vehicles with recoverable payloads was illustrated in a 1986 study.[2b] Other risk analyses have been performed for specific tests and launch vehicles based on tailored models using the vehicle characteristics and launch geometry.

WSMC has an extensive array of software developed to assist in evaluating hazards to facilities and population centers and devising appropriate risk control options.[19-21] These include: LARA, CONDEC (Conditional Casualty Expectation), RBAC (Risk Based Destruct Criteria), ACE (which combines CONDEC and RBAC to compute casualty expectation along arbitrary destruct lines), SLCRSK and LCCRSK (which compute probabilities and expected magnitude of damage to the reinforced launch control center and to other VAFB facilities, such as SLC-6, for certain launch azimuths).[20] Other special purpose models are: BLAST, to assess explosion shock wave far-field impacts; SABER,

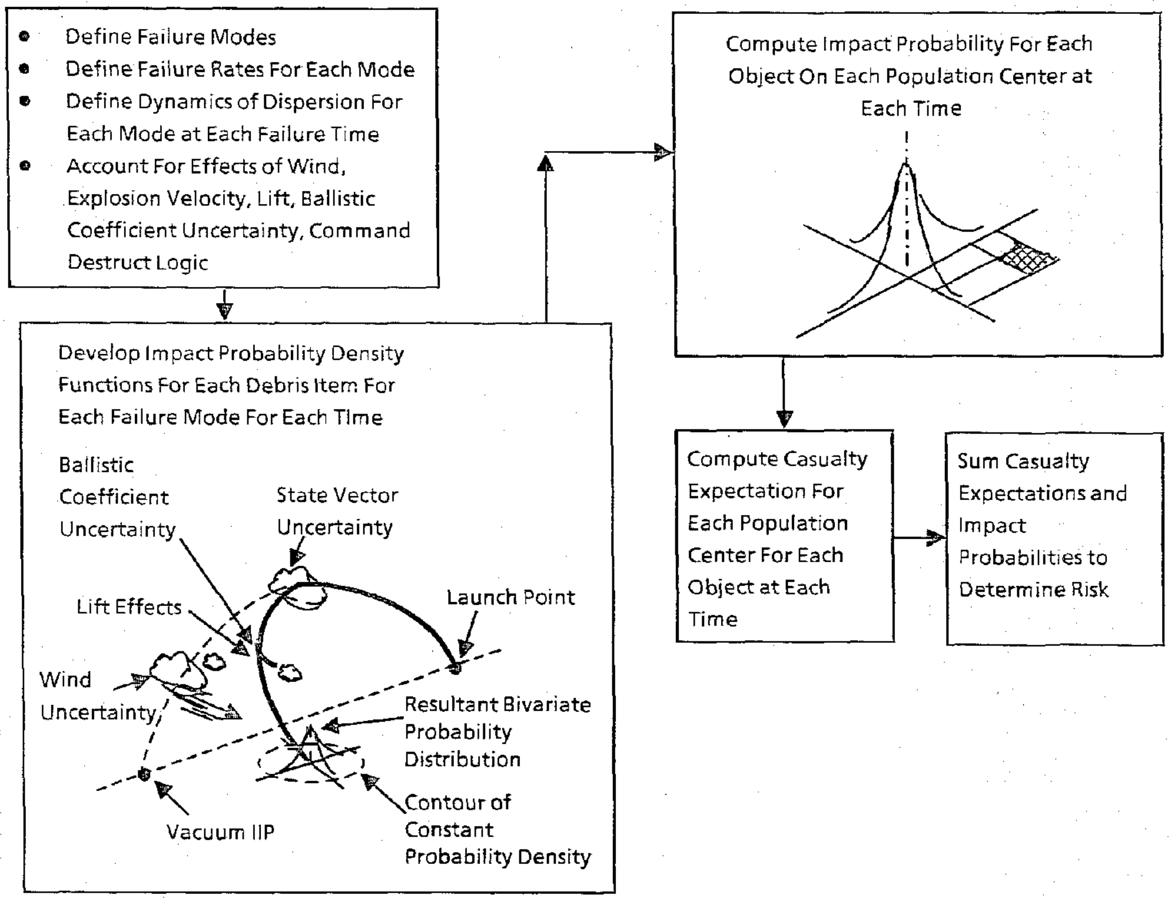

FIGURE 9-3. RISK COMPUTATION

to evaluate supersonic boom effects; REEDM, for hot toxic gas predictions and a series of cold spill toxic prediction algorythms for toxic releases.

ESMC has its own library of codes used to support launches as pre-flight simulations and real-time monitoring and display. These include: BLST, similar to BLAST above; COLA, a collision avoidance program used to ensure that a proposed launch will not jeopardize any satellite in orbit; RAID, the major real-time Range Safety program which displays the ELV position and IIP based on tracking data; RSAC and RSTR, which provide plots in site-centered coordinates; REED, used for launch and post launch environmental analysis of exhaust cloud effects; RIPP, an interactive impact point and destruct line plot and RSIP (Range Safety Impact Predictor), which computes impact position parameters along the trajectory with and without wind data. Other codes are used to assess the fate of an errant ELV, such as RSPFT (Range Safety Powered Flight and Turns) and RSTT (Range Safety Tumble Trajectory), to predict malfunction behavior for each vehicle type and nominal trajectory; and RSMR, which computes the maximum pad-to-impact range for a vehicle and its debris. External

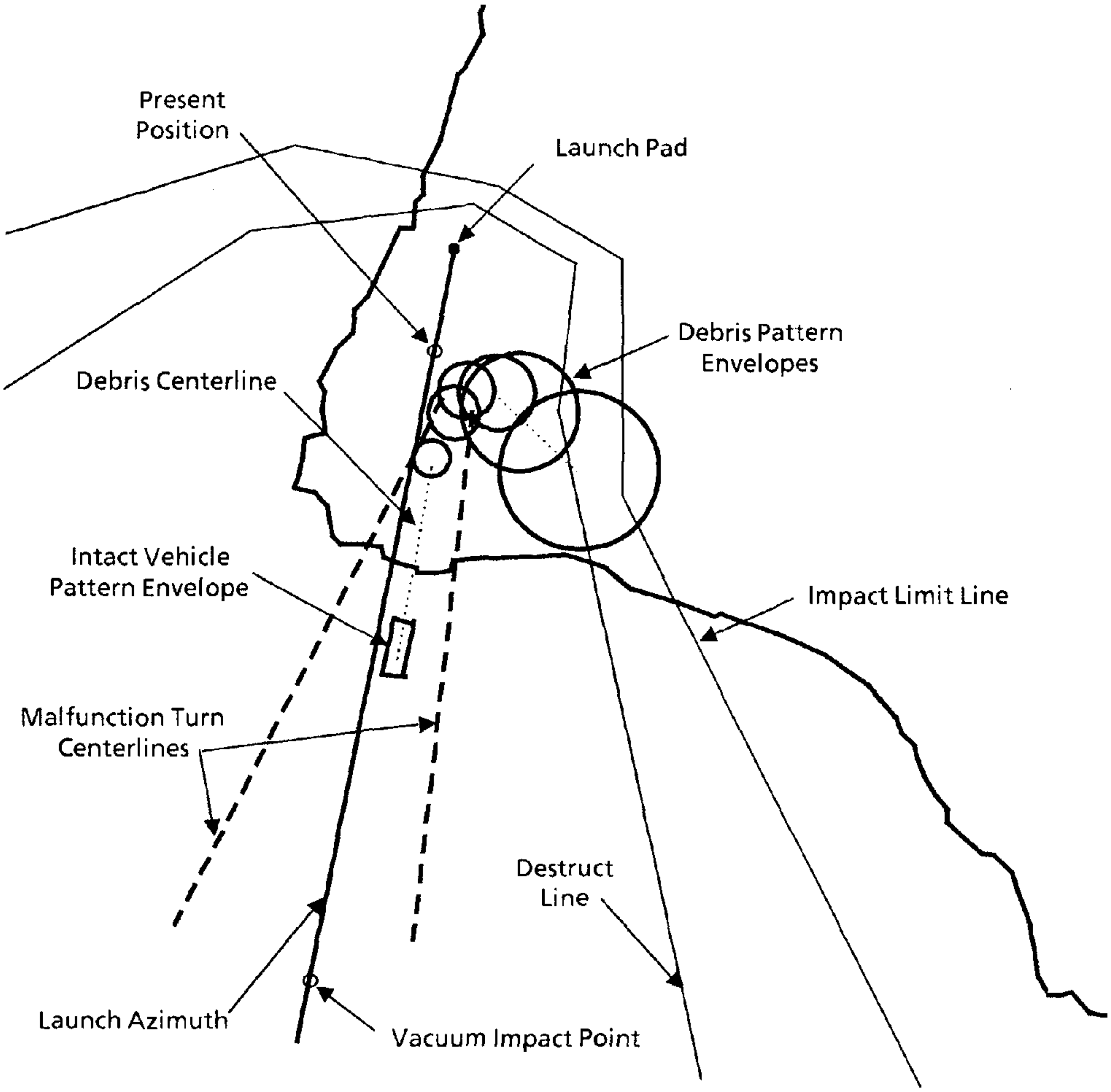

FIGURE 9-4. REAL TIME DEBRIS FOOTPRINT DISPLAY

modules are used to update wind corrections (RSRK, for Range Safety Radiosonde Data) and assess risks to ocean traffic (RSSP or Range Safety Ship Hit Probability).

For any developmental vehicle, safety assessments must precede flight testing and launch approval. For example, the new commercial launch vehicle Conestoga has been flight tested recently; Conestoga failure modes and rates were based on previous experience with the Aries rocket and the Minuteman I second stage motor, which were reconfigured as the Conestoga. Special attention was given to the possibility of impact and damage to off-shore oil platforms in the Gulf Area, given the flight path, ground track and safety corridor for Conestoga under a range of plausible vehicle failure scenarios and weather conditions.[36] However, because of redesign of the Conestoga, some of the safety assessments are being re-evaluated for launches from WFF.

The hazard models used by NORAD and AFSC to estimate far-field public risk exposure (i.e., for assessing the probability that a failed vehicle, re-entering second stage or debris will impact in CONUS and/or foreign countries and cause damage and casualties) were originally developed by the Aerospace Corporation.[34,35] These re-entry risks for second and upper stages and for for low-orbit payloads appear, typically, to be several orders of magnitude larger than launch and orbit insertion risks (see Ch.7, Vol.2) because they integrate world-wide casualty expectation. Impact probabilities and casualty expectations for a specific country are much smaller and proportional to their area and population contribution to the integral.

Overflight risks are also a modeling and operational planning concern for Range Safety: some trajectories may traverse Japan, Australia, Africa and South America (see Ch.10 also). Table 9-2 summarizes extant risk results, namely the probabilities of land impacts and projected casualties for typical ELV's on allowed azimuths for ESMC launches over water.[37] These flights must protect the "African Gate" during overflight (see also Ch.10). This performance gate defines the maximum cross-range deviations from the nominal overflight trajectory which may be tolerated without termination action. These are well within the destruct limits to better protect populated areas at risk in case of abnormal vehicle performance.

To place the criteria and goals for public risk exposure per space launch in perspective, it is instructive to compare them with other common, but voluntarily assumed or socially accepted transportation risks (see also Ch.5, Vol.2 and Ch.8). Ref. 29, published prior to the 1986 Challenger accident, estimated the casualty probability per flight for commercial air carriers to be 6.6×10^{-5} (based on 1972-74 data) vs. $1\text{-}3 \times 10^{-5}$ for the Space Shuttle (to compare respective risks from an STS failure with and without a destruct system on-board). For comparison, the 1982-84 transportation accident statistics give fatality rates per 100 million passenger-miles of .02 for inter-city buses, .04 for airlines and .07 for railroads. These values correspond to a casualty probabilities of $2\text{-}7 \times 10^{-10}$ per mile. This probability must be converted to units of interest to space operations (per launch event or per year) and then further normalized to the exposed population and the area at risk. Further, utility/benefit considerations must be brought to bear for a meaningful comparison of public transportation with space transportation risks.

TABLE 9-2. OVERFLIGHT LAND IMPACT PROBABILITIES & CASUALTY EXPECTATIONS AT ESMC (Ref 37)

Vehicle	Flight Az. (Deg)	P_i**	E_C*
Titan 34D/Transtage (1)	93	2.2×10^{-5}	2.1×10^{-8}
	97	1.7×10^{-5}	1.2×10^{-8}
	101	1.4×10^{-5}	0.7×10^{-8}
	105	1.1×10^{-5}	1.1×10^{-8}
	109	0.9×10^{-5}	1.5×10^{-8}
	112	0.7×10^{-5}	1.3×10^{-8}
Titan 34D/IUS (2)	40	unknown	1.6×10^{-6}
	44	"	0.4×10^{-6}
	48	"	0.2×10^{-6}
	52	"	0.7×10^{-6}
	56	"	0.3×10^{-6}
	60	"	0.1×10^{-6}
Space Shuttle (3)	39	"	3.5×10^{-7}
	61	"	7.5×10^{-8}
	90	"	1.8×10^{-7}
Atlas Centaur (4)	80	1.5×10^{-2}	9.6×10^{-6}
	90	0.66×10^{-2}	4.0×10^{-6}
	100	0.28×10^{-2}	0.7×10^{-6}
	110	0.14×10^{-2}	1.3×10^{-6}
Delta (5)	95	4×10^{-3}	3.7×10^{-6}
	108	8.1×10^{-4}	8.3×10^{-7}

Notes:

(1) 1982 study. Failure rate for stage thrusting during dwell time over Africa assumed to be 2.3×10^{-5} failures/sec. Ac = 860 sq. ft.

(2) 1978 study. Failure rate for stage thrusting during dwell time over Africa = 2.3×10^{5} failures/sec. A_c = 400 sq. ft.

(3) 1981 study. Failure rate assumed for overflight stage 2.9×10^{-7} failures/sec. As part of same study, NASA estimated catastrophic failure probability for solid rocket motor of 1×10^{-4}; the Range estimated 1×10^{-2}.

(4) Study from mid 1960's with failure probability for Centaur stage = 0.33

(5) Study from mid 1960's with failure probability for Agena stage = 0.108.

**P_i = Probability of land impact equal to the product of the dwell time over land with the failure probability of the vehicle stage thrusting during the dwell period.

*E_C = Casualty expectation equals product of P_i, the population density and the area exposed to re-entering fragments.

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10. A GENERIC RISK ASSESSMENT OF REPRESENTATIVE LAUNCH SCENARIOS

10.1 INTRODUCTION

Since the beginning of US space launch operations in the 1950's, there have been no launch operation accidents that have produced any general public casualties outside any of the Government Launch facilities. There has been some damage to some Range facilities and structures used to support the launches, but little damage to public property outside the perimeter of the launch sites. Considering the fact that there have been unavoidable failures during thirty years of new rocketry and spacecraft testing and streamlining of launch operations, it is evident that the Range Safety Control process and systems in place have prevented and controlled the risk from launch accidents that could have lead to potentially significant claims against the Government.

This proven track record of success for the Range Safety Control systems and practices at the National Ranges may cast doubt on the need to discuss the public risk exposure levels and the potential for third party liability claims. It is worthwhile, however, to discuss the consequences of ELV launch failures in the absence of the Range Safety Control system since proposed commercial space launches could originate at new launch sites (perhaps an island site or an ocean platform); use novel, untested or reconfigured tracking and control systems; and not require an FTS of high reliability on-board ELV's. This approach will permit an assessment of the extent of potential damage and/or casualties that can be avoided by the established Range Control Systems and safety practices (see also Ch.2, Vol.1, and Ch.9). While much of the qualitative hazards analysis of launch-related accidents has been given previously in Ch.5, Vol.2, the intent of this chapter is to provide a coherent, self-contained discussion of generic public risk associated with commercial launch operations for existing ELV's which weighs the consequences of each accident by its probability of occurrence in a Risk Matrix according to the methods and tools illustrated in Chs. 8 and 9.

10.2 RISKS DURING DIFFERENT PHASES OF A TYPICAL MISSION

10.2.1 Pre-Launch Hazards

During the preparation of a vehicle for launch, the chief hazards derive from the storage and handling of propellants and explosives. The Ground Safety procedures applied to stored explosives and propellants that can explode are similar to those used in the transportation and handling of these same materials off-site. The protective measures include quantity-distance requirements, so that parties uninvolved with the launch cannot be affected by any accident. In addition, other structural protection (e.g., hardened concrete) and emergency preparedness measures are used to contain toxic or corrosive materials within the boundaries of the Range in case of an accident on the pad (see also Ch.5, Vol.2).[1,12]

Accidents occurring prior to launch can result in on pad explosions, potential destruction of the vehicle and damage to facilities within range of the blast wave as well as dispersion of debris in the vicinity of the pad. The types of accidents depend upon the nature of the propellants, as discussed in Ch.5. In the case of cryogenic propellants, liquid oxygen alone will cause fires and explosive conditions; if used in association with liquid hydrogen, it can lead to very explosive conditions. Under somewhat ideal conditions, the TNT equivalence of a hydrogen-oxygen propellant explosion can be as much as 60 percent of their weight, while that of an RP-1-oxygen explosion can be 20 percent of the weight of the propellants (see Ch.5, Vol.2).[1]

An accident in handling storable hypergolic propellants could produce a toxic cloud, liable to move as a plume and disperse beyond the boundaries of the facility. The risk to the public will then depend upon the concentration of population in the path of this toxic plume and on the ability to evacuate or protect the population at risk until the cloud is dispersed. It is obviously advantageous if the winds generally blow away from populated areas. There are also specific safety requirements and risks associated with ground support equipment. The design and use of this equipment must incorporate safety considerations.

10.2.2 Launch Hazards

Generally, the on-board destruct system is not activated early in flight (during the first 10 seconds or so) until the failed vehicle clears the Range. This protects Range personnel and facilities from a command explosion. Failures during the very early portion of launch and ascent to orbit can be divided into two categories: propulsion and guidance/control. Lighting, wind and other meteorological hazards (e.g., temperature inversions) must be considered prior to launch countdown.

Propulsion failures produce a loss of thrust and the inability of the vehicle to ascend. Depending on its altitude and speed when thrust ceases, the vehicle can fall back intact or break up under aerodynamic stresses. If the vehicle falls back, the consequences are similar to those of an explosion on the ground. The exception is when intact solid rocket motors impact the ground at a velocity exceeding approximately 300 fps. In that case, the explosive yield may be significantly increased. If there are liquid fuels (hydrogen-oxygen), there is also potential for a large explosion, much higher overpressures and more damage to structures at the launch facility. It could also create higher overpressures off the facility which could break windows and possibly do minor structural damage to residential and commercial buildings (see Ch.5, Vol.2).

Solid rocket motor (SRM) failures can be due to a burn-through of the motor casing or damage or burn-through of the motor nozzle. In a motor burn- through there is a loss of chamber pressure and an opening is created in the side of the case, frequently resulting in structural breakup. The nozzle burn-through may affect both the magnitude and the direction of thrust. There is no way to halt the burning of a solid rocket once initiated. Hence, an SRM failure almost inevitably puts the entire launch vehicle and mission at risk. When there are several strapped-on SRM boosters, as is commonly the case, the probability of a failure of this type is increased, since any one of these failing can lead to mission loss.

The purpose of the Range Safety Control system is to destroy, halt or neutralize the thrust of an errant vehicle before its debris can be dispersed off-Range and become capable of causing damage or loss of life. Without a flight termination system (FTS), the debris could land on a population center and, depending upon the type of debris (inert or burning propellant), cause considerable damage. The destruct system generally is activated either on command or spontaneously (ISDS - the inadvertent separation destruct system is activated automatically in case of a stage separation failure) at or soon after the time of failure. In flight destruction limits vehicle debris dispersion and enables dispersion of propellants, thus reducing the possibility of secondary explosions upon ground impact. The destruct systems on vehicles having cryogenics are designed to minimize the mixing of the propellants, i.e., holes are opened on the opposite ends of the fuel tanks. This contrasts with vehicles with liquid storable propellants (e.g., Aerozine-50 and N_2O_4) where the destruct system is designed to promote the mixing and consumption of the propellant. Solid rocket destruct systems usually consist of linear shaped charges running along the length of the rocket which open up the side of the casing like a clam shell. This causes an abrupt loss of pressure and thrust. It may, however, produce many pieces of debris in the form of burning chunks of propellant and fragments of the motor casing and engines.

The Titan 34D accident on April 18, 1986, about 8 seconds after launch, is an example of a propulsion failure which caused considerable and costly damage to the VAFB facility.(2) In this case, the solid rocket case failed and the vehicle fragmented and spread burning propellant over the launch site. Typical debris velocities were 100 to 300 fps. This Titan 34D failure was the result of a burn-through of one of the rocket motor casings. The explosion, which occurred at an 800 ft. altitude, was not a detonation, where there is almost instant burning of propellant accompanied by a significant airblast, but a deflagration, where most of the propellant was not consumed in the explosion, but formed a cloud of flying burning debris. Some of the burning propellant still encased in a section of the rocket motor did appear to explode upon impact. The evidence was a flash of light recorded by a camera, although the camera was not directed at the point of impact. A series of small craters were also observed after the accident. It is believed that some of these these craters were formed by violent burning in the soft soil (sand) rather than by explosions. Films do show rebound of propellant chunks and shattering upon the rebound. This type of behavior was also observed in earlier Minuteman failures.

In addition to complete loss of control, there are three other early flight guidance and control failures that have been observed with launch vehicles over the life span of the space program: failure to pitch over, pitching over but flying in the wrong direction (i.e., failure to roll prior to the pitchover maneuver) and having the wrong trajectory programmed into the guidance computer. The likelihood of these circumstances depends upon the type of guidance and control used during the early portion of flight. The types are open or closed loop (i.e., no feedback corrections) and programmer or guidance controlled. In the case of vehicles which use programming and open-loop guidance during the first portion of flight, failure to roll and pitch is possible, although relatively unlikely, based on historical flight data. If the vehicle fails to pitchover, it rises vertically until it is destroyed. As it gains altitude, the destruct debris can spread over an increasingly larger area. Consequently, most Ranges watch for the pitchover and if it does not occur before a specified time, they destroy the vehicle before its debris pattern can pose significant risk to structures and people outside the launch facility or the region anticipated to be a hazard zone, where restrictions on airspace and ship traffic apply. Failure to halt the vehicle within this time can produce a significant risk to those not associated with launch operations.

With open-loop Stage 1 guidance, a launch in the wrong direction can occur due to improper programming or improper roll of the vehicle during its vertical rise. This circumstance, although considered improbable, can be very hazardous. If the Range does not halt the flight immediately, the vehicle could overfly populated regions. Then, even if the vehicle is normal in every other respect, it could drop jettisoned stages on populated areas, creating the potential for damage, injury and loss of life. The detection of improper launch azimuth is usually accomplished visually because radar tracking may not be effective very early in flight. Consequently, in making the decision to halt the flight, the Range must rely on visual observers to relay information about pitchover and azimuth, with possible time-delays.

With vehicles which are inertially guided from liftoff, failure in pitchover or roll is unlikely. It is possible, but extremely unlikely, that an inertially guided vehicle could have the wrong set of guidance constants, i.e., the wrong trajectory, stored in its guidance computer. To the observer this will appear the same as an improper roll (flight azimuth).

If a solid rocket loses thrust or has a change of direction of the thrust vector, the vehicle control system will try to compensate with the remaining engines. The result will be an aberrant corkscrewing behavior until the control system is totally overwhelmed, and then a tumble. With atmospheric forces present, the stages should break apart by this time.

Generally, rapid hard-over tumbles of failing vehicles do not cause the vehicle to move significantly cross-Range off the intended path of flight. It is the gradual turn that is of greater concern to the Range Safety Officer. If the vehicle turns slowly, it can move a significant distance cross-Range. This type of failure is rare and difficult to rationalize with most flight-tested ELV systems, but the unexpected must be anticipated. An example of the unexpected is the behavior of the solid rockets from the Space Shuttle after the failure of the Challenger.(3) They were supposed to tumble and not offer much of a dispersal hazard. Instead they turned very little and had to be destroyed before they could become a threat to a populated area.

Of greatest concern to Range Safety Control during the steep ascent phase, is the capability of the vehicle to wander off-course immediately following a malfunction. The Range Safety Control system must be able to respond before debris becomes a hazard. Consequently the design of the destruct lines must take into consideration: (1) the delay between decision and destruct; (2) the highest rate that the vehicle can move its IIP toward a protected area; (3) the effect of the winds; and (4) the contribution of any explosion to the scatter of debris.

During the early boost phase the vehicle experiences its greatest aerodynamic loads and heating. As the vehicle accelerates, the dynamic pressure ($1/2\ \rho v^2$) increases until the decrease in density (ρ) due to higher altitude overcomes the effect of increasing velocity (v). During the period of high airloads the vehicle is more vulnerable structurally and likely to break apart if it has a high angle of attack or begins to turn abruptly. The Space Shuttle, for example, with its complex configuration and lifting surfaces, is so sensitive during this period that the liquid propelled main engines are throttled down to keep the dynamic pressure within specified limits. One of the major fears during this phase is an abrupt change in wind velocity during ascent (a wind shear). This causes a rapid change in angle of attack and requires rapid and appropriate response by the control system.

The potential for damage to ground sites from a launch vehicle generally decreases with time into flight since fuel is consumed as the vehicle gains altitude (see Fig.5-6 in Ch.5, Vol.2). If it breaks up or is destroyed at a higher altitude, the liquid fuels are more likely to be dispersed and lead to lower concentrations on the ground. In addition, if there are solid propellants, they will have been partially consumed during the flight period prior to the failure and will continue to burn in free fall after the breakup.

Meteorological conditions contribute to the potential for off-site damage. Temperature inversions and wind shears can cause shock waves, which normally turn upward, to turn down and possibly focus at locations distant from the launch site.(4) This results in significantly higher overpressures locally, than the overpressures from shockwaves moving in a normal adiabatic atmosphere (an atmosphere where the temperature decreases with increasing altitude). Another meteorological influence is the wind, which can deflect falling debris towards populated areas.

Very early in flight, when the vehicle is still close to the ground, there is less opportunity for debris to be scattered. The debris fall within a footprint which is affected by the range of ballistic coefficients of the pieces, the wind speed and direction, velocity contributions due to explosion and random lift (see also Ch.2, Vol.1 and Ch.7, Vol. 2). To understand the make-up of the debris footprint, first observe the "centerline" as shown in Figure 10-1.[5] This centerline represents the spread of debris impact and drag effects when there is no uncertainty due to wind, lift, etc.

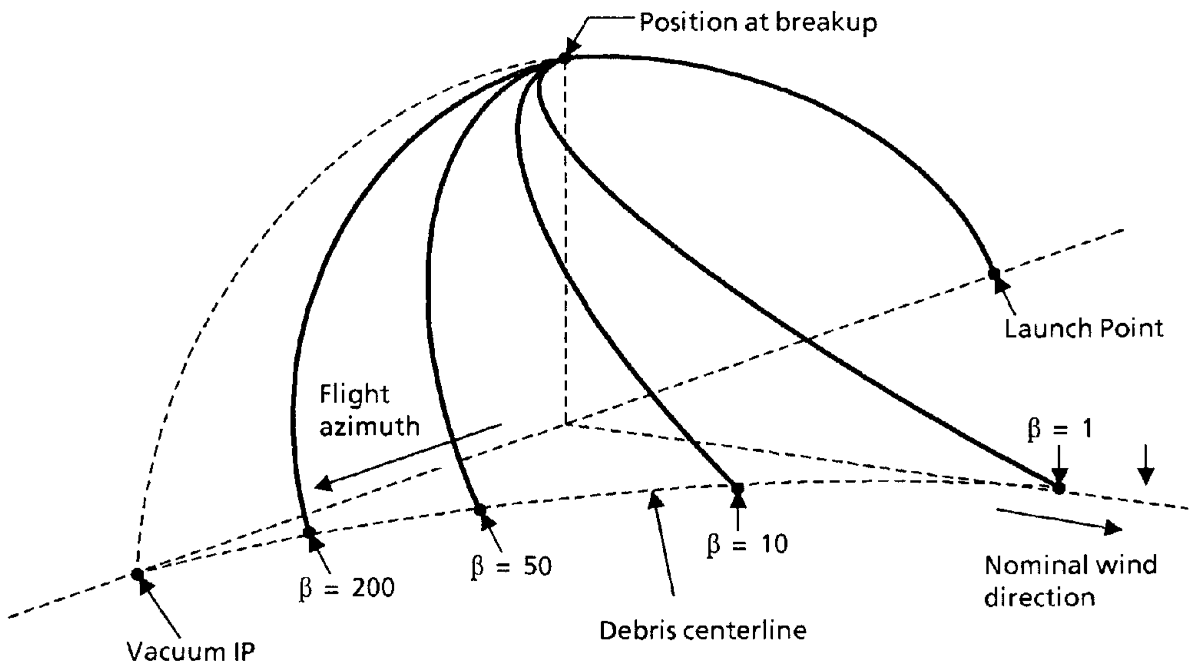


FIGURE 10-1. BEHAVIOR OF DEBRIS AFTER VEHICLE BREAK UP

Debris which are very dense and have a high ballistic coefficient (β) are not as affected by drag and will tend to land closer to the vacuum IIP. High ballistic coefficients can be associated with pumps, other compact metal equipment, etc. Panels or pieces of motor and rocket skin offer a high drag relative to their mass (a low ballistic coefficient) and consequently slow down much more rapidly in the atmosphere. After slowing down they tend to fall and drift with the wind. This effect is also shown in the figure. A piece of debris with a very low ballistic coefficient ($\beta = 1$) is shown to stop its forward flight almost immediately and drift to impact in the direction of the wind. Pieces having intermediate value ballistic coefficients show a combination of effects and fall along a centerline. From a lethality standpoint, the pieces having a higher ballistic coefficient impact at a higher velocity and can cause more damage (depending upon their size). The debris will not necessarily impact along the centerline. The velocity impulses at breakup, the wind and tumbling behavior all contribute to uncertainties about the impact point. This is illustrated in Figure 10-2.

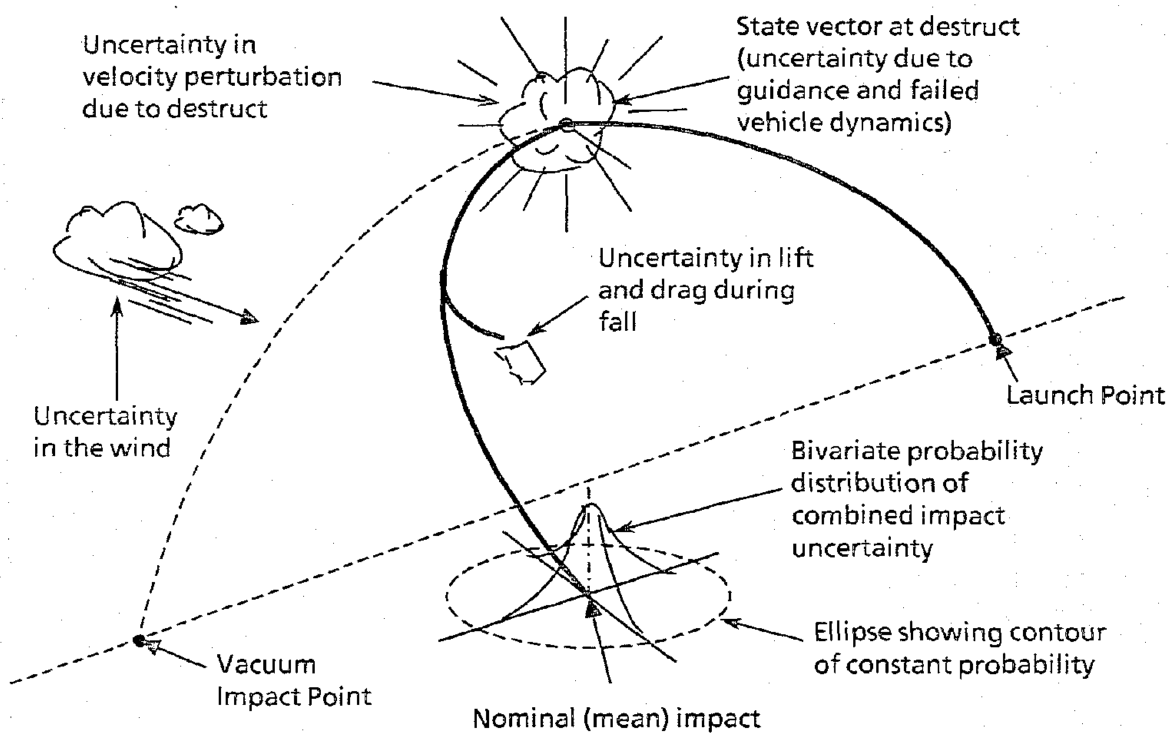


FIGURE 10-2. DEBRIS IMPACT DISPERSION

When all of the factors affecting debris transport and dispersal to impact are considered at once, the effect is a pattern as shown in Figure 10-3. The boundaries of the debris dispersion footprint are not precise but rather represent a contour which contains, say, 95 percent of the debris. Thus, when considering the hazard to structures or people on the ground, one must consider the hazard area for debris impacts in the terms of a pattern which is dynamic. It grows rapidly as the vehicle gains altitude, as illustrated in Figure 10-4 for a vehicle launched from Vandenberg Air Force Base. Note the geography and the fact that part of the debris pattern dwells over land for a significant period of time. The time interval that the debris impact pattern dwells over land depends upon the direction and strength of the wind. If the wind, as in this case, is blowing very hard from the southwest, the low ballistic coefficient portion of the pattern will tend to stay over the land. If the wind is blowing from the northeast, the pattern will move very rapidly out to sea. This demonstrates the very important role of wind in evaluating risks of a launch. Depending on prevailing meteorological conditions, including clouds, visibility, atmospheric electricity, temperature and wind conditions, a launch may be postponed until adverse conditions subside.

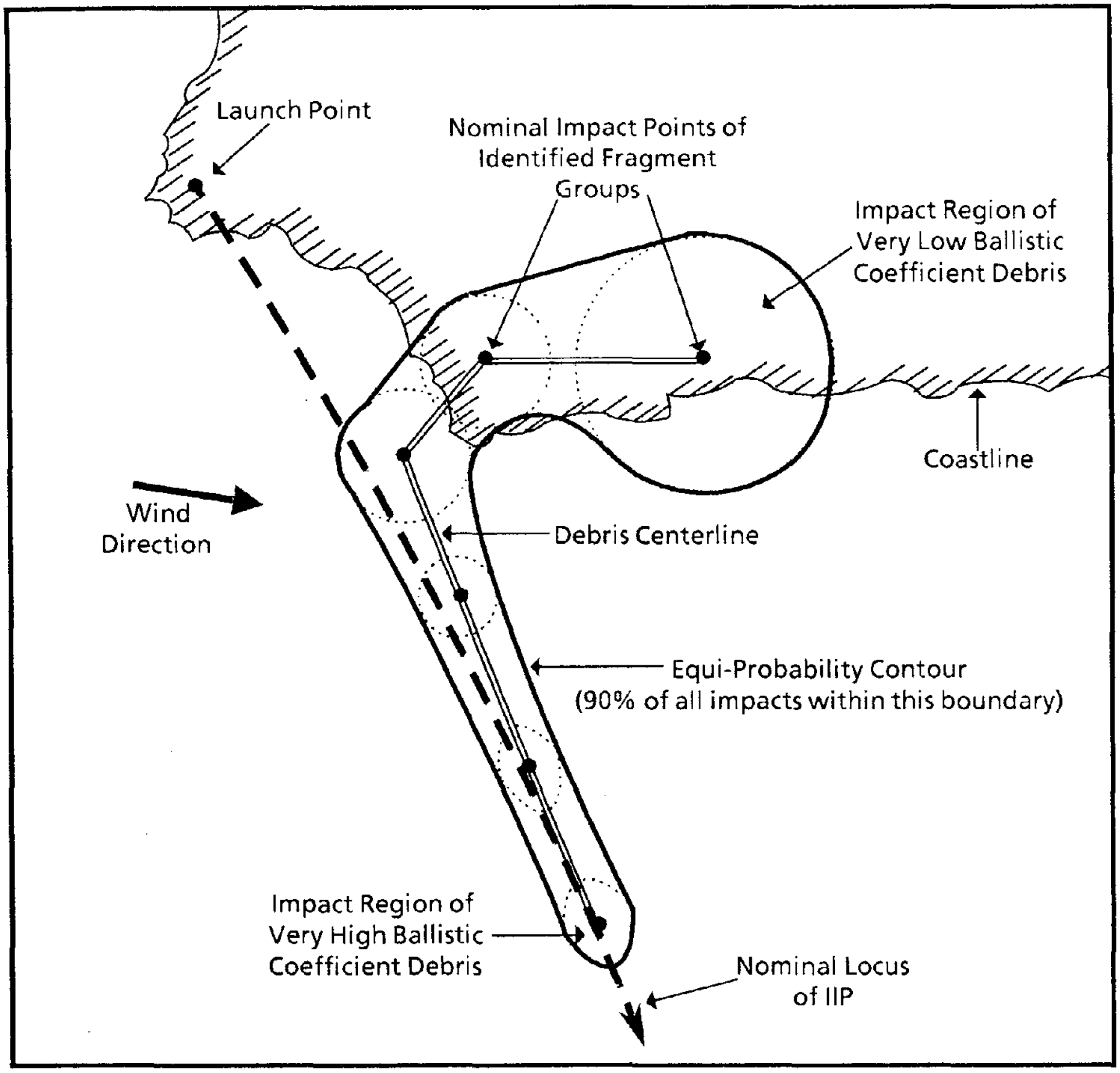


FIGURE 10-3. A TYPICAL DEBRIS DISPERSION AT IMPACT PATTERN

The bulge in the center of the growing debris pattern in Figure 10-4 is due to debris which have velocities imparted to them from an explosion (spontaneous or destruct action). The upper-right hand portion of the debris pattern consists of debris which have a high drag to weight ratio, slow down quickly and are carried by the wind, which, in this case, is blowing from the west. Notice how the debris pattern stretches as the vehicle increases in altitude. This effect continues until the vehicle reaches an altitude where aerodynamic drag no longer has an effect on dispersion.

For all launches, the boosters, sustainers and other expendable equipment are always jettisoned and fall back to the Earth. Therefore, in planning a mission, care must be taken to keep these objects from impacting on land, offshore oil platforms, aircraft and shipping lanes. The impact locations are normally quite predictable, so risks can be avoided or minimized.

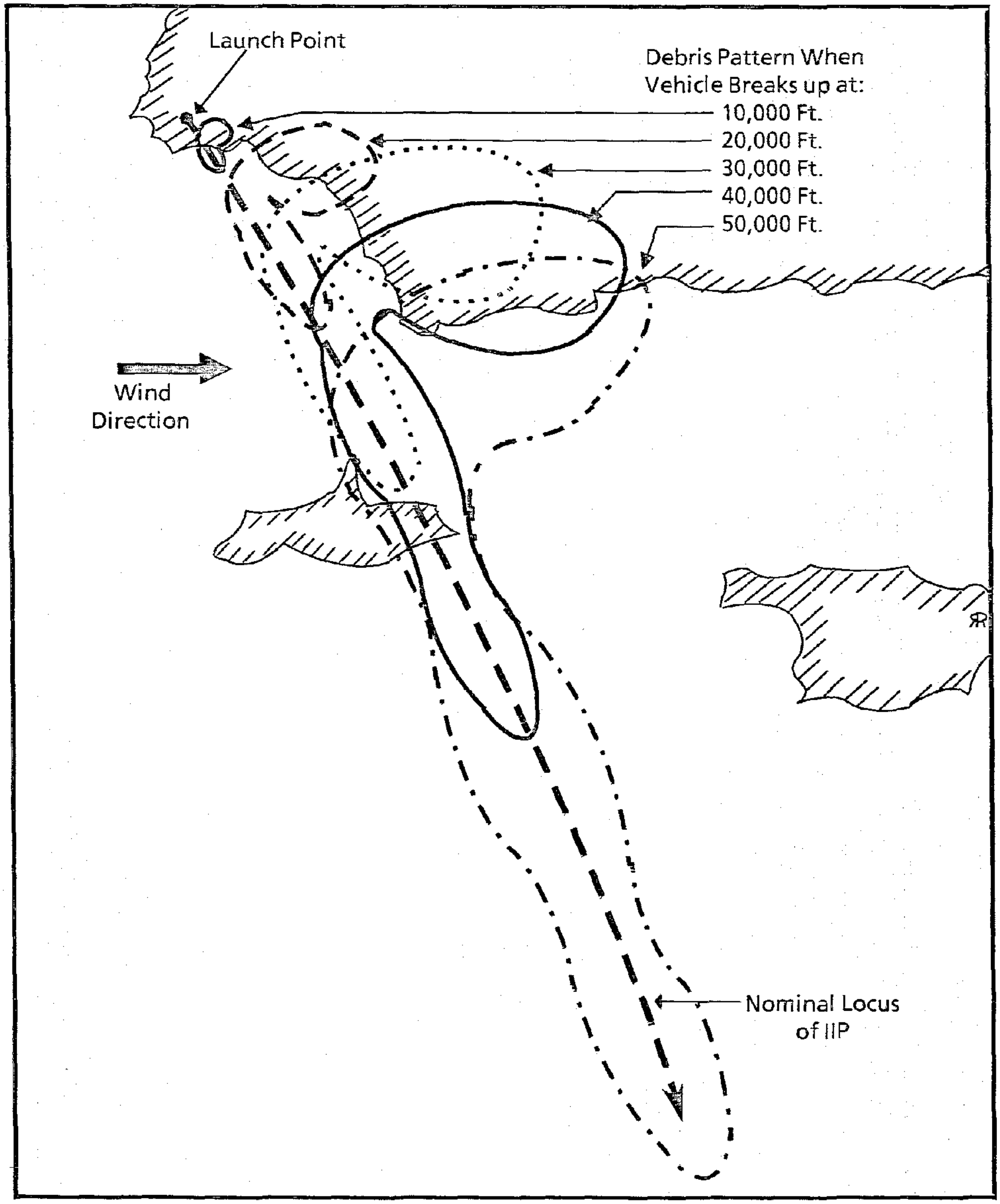


FIGURE 10-4. GROWTH OF THE DESTRUCT DEBRIS IMPACT DISPERSION PATTERN

As mentioned earlier, during the entire history of the space and missile programs at VAFB and Cape Canaveral/Cape Kennedy, no errant launch vehicle has ever been allowed to wander over a populated area near the launch site and deposit debris upon it. As a consequence there have been no claims, damages or casualties. This is a convincing argument in the support of continued safe launch and mission planning and approval procedures, reinforced by a reliable Range Safety Control system.

10.2.3 Pre-Orbital Hazards

After jettison of the booster stage and, in some cases, the solid rockets, the remaining core vehicle usually contains only liquid propellants and is at a fairly high altitude. If a failure occurs and no destruct action takes place, the vehicle may fall and remain largely intact till ground impact. Depending upon the initial altitude, the airloads during the fall may become sufficient to contribute to the vehicle breakup. If this occurs, the propellants will most likely be dispersed and the only hazard will be from impacting "inert" debris. In the unlikely event that the tanks do remain intact, some explosion may occur at impact. If the propellants are hypergolic, as in the case of the Titan, there may be considerable burning and a cloud appearing in the impact area. In this latter case, the damage from debris impact will probably be less than the hazard from the toxic propellants. When an altitude is reached where the vehicle stages can no longer remain intact because of airloads and heating, the only hazard will be due to impacting debris.

If a destruct or thrust termination system is used to halt ascent, as is usually the case, the propellants will be dispersed and should offer very little threat to people on the ground. A product of the destruct action will be inert debris, which could present a hazard at ground impact (for fire, explosion and toxic hazards, see Ch.5, Vol 2).

During the boost trajectory of almost any space vehicle from any US National Range, the IIP will at some time pass over occupied land. For Titan 3 launches due east from Cape Canaveral, the IIP will begin to pass over Africa at t = 475 seconds, and leave Africa 3 seconds later. For some southerly launches from Vandenberg Air Force Base, the IIP can pass over southern Argentina and Chile. Activation of the destruct system is of no value at this point because it poses risks of land impact. It is often better to let the failing vehicle continue with the hope that it will clear the land area and impact in the ocean. The threat from either launch condition is relatively small because in both cases the IIP is traveling very fast over land areas (hundreds of miles per second). If, for example, the failure rate of the Titan 3 were uniformly 0.000075 failures per second (historical launch failure probability of .036 divided by 480 sec. of burn operation) and the time required for the IIP to cross Africa is 3.2 seconds (see Figure 10-5), then the probability of failing and causing debris to fall on Africa is 3.2 times 7.5×10^{-5} or 2.4×10^{-4} (one chance in approx. 4200). If the combined cross section of debris which survive to land impact is on the order of 1000 sq. ft.,and the average density of population which can be harmed by the debris is 50 per square statute mile (according to Ref. 5, this figure is higher than the average of the population densities of Zambia, Angola and Zimbabwe), then the average number of casualties per launch due to an African impact is:

$$E_c = (\text{failure rate}) \times (\text{dwell time over land}) \times (\text{debris "casualty area"}) \times (\text{population density})$$

$$= 7.5 \times 10^{-5} \times 3.2 \times 1000 \times (50/5280^2) = 4 \times 10^{-7}$$

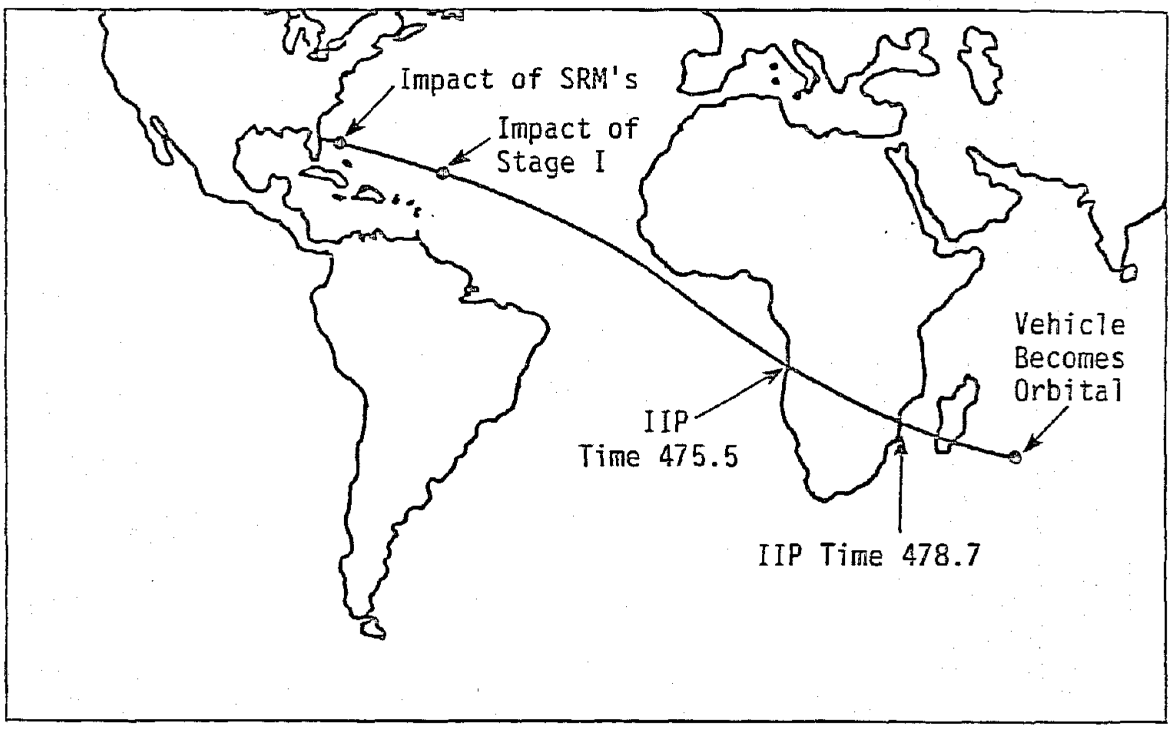


FIGURE 10-5. LOCUS OF IIP FOR A TYPICAL TITAN III LAUNCH FROM CAPE CANAVERAL (ETR)

This corresponds to less than one chance in a million of a casualty per launch. Whereas Range Safety Control systems can act very positively to restrict and prevent debris from falling on populated areas earlier in flight, there is no effective risk control when the flight plan calls for a direct land overflight, such as the one discussed above. Consequently the casualty expectation of 8×10^{-7} is the same with or without a flight termination system on-board the ELV.

The potential for damage from the impact is based on the area of falling debris (in this case estimated to be 1000 $ft.^2$) and the likelihood of impacting a structure of value. With a population density of less than 50 per square mile, the density of such structures is rather low. As an example, assume the surviving debris consist of four pieces, each having a cross-section of 250 $ft.^2$, and the average structure is 600 $ft.^2$ with, on the average, one person per structure. (This is an attempt to account for both residential and commercial structures very conservatively.) A structure will be hit if any edge is hit by the debris. This is pictured in Figure 10-6. The effective area of impact is therefore a combination of the structure area and the debris cross-sectional area. In this case the effective impact area becomes approximately 3400 $ft.^2$. The probability of any impact on a structure becomes:

P_i = (failure rate) x (dwell time) x (effective impact area) x (structural density) x (no. of fragments)

$= 7.5 \times 10^{-5} \times 3.2 \times 3400 \times (50/5280^2) \times 4 = 5.5 \times 10^{-6}$.

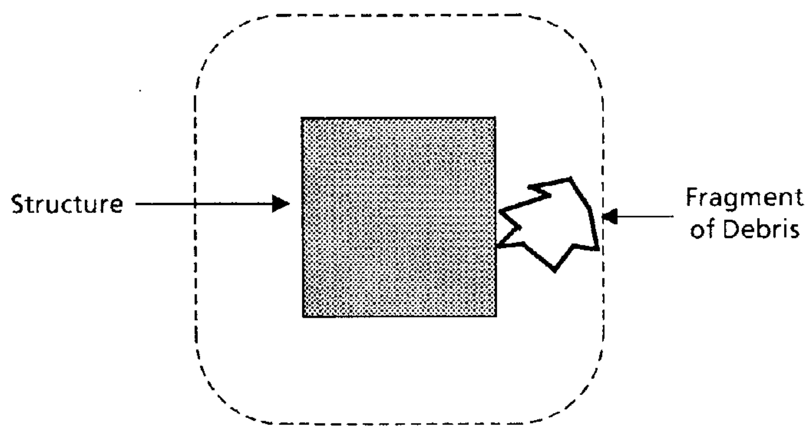


FIGURE 10-6. MODEL CALCULATION OF THE EFFECTIVE AREA OF IMPACT

Thus, in this example the probability of hitting and damaging a structure is approximately 1 in 100,000. If a monetary value or range thereof, were assigned to the structures at risk, then the expected loss could be tied to both the severity and extent of damage (the consequence) and to the very low probability of its occurrence.

A similar analysis can be performed for launches from Vandenberg Air Force Base (see Figure 10-7) when the IIP passes over the southern portion of South America. According to Ref. 7 and to Figure 10-8, an ELV would have to violate current azimuth restrictions in order to overfly South America (although some flights may overfly Antarctica or Australia at much greater altitudes). The dwell or transit time over Chile and Argentina will be no more than 1.4 seconds. If all other parameters of the casualty expectation and impact probability equations are assumed to be the same, then the E_c and the P_i will be less than those over Africa by the ratio of 1.4/3.2. Thus, very approximately, the casualty expectation for overflight over the southern region of South America will be 1.75×10^{-7} and the impact probability on a dwelling or commercial structure will be 2.3×10^{-6}.

On-orbit collision hazards, once the satellite has been properly inserted into final orbit, have been discussed in detail in Ch. 7. Similarly, orbital decay and re-entry hazards for satellites and spent rocket stages have been addressed in Ch. 8. Although they contribute to the overall space mission-related hazards, they will not be discussed any further here.

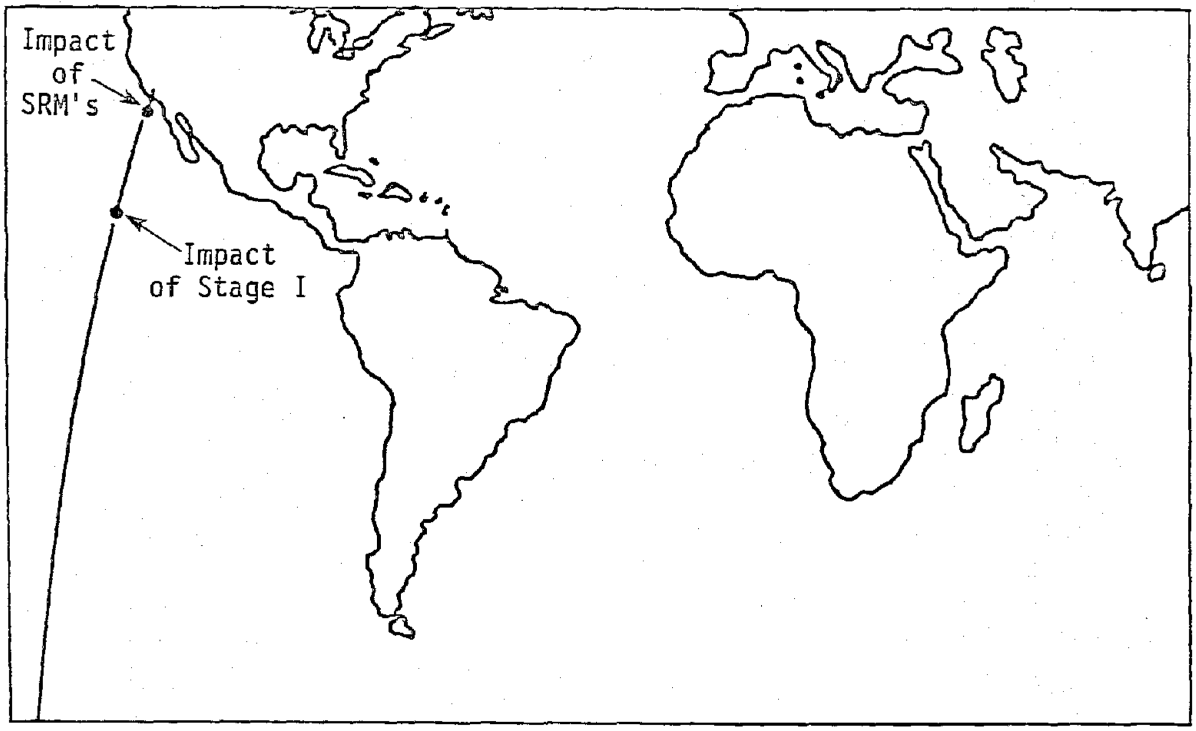


FIGURE 10-7. LOCUS OF IIP FOR A TYPICAL TITAN III LAUNCH FROM VANDENBERG AIR FORCE BASE (WTR)

10.3 LAUNCH SITE RISK CONSTRAINTS

The location of the launch facility has a significant impact on the options for launch missions. Launches to the east always benefit from the west to east rotation of the Earth. Consequently, equatorial orbits (0^o inclination) are best achieved by launching from facilities which are near the equator and have a broad ocean area to the east of the launch site. Figures 10-8 and 10-9 show the acceptable and restricted azimuths for launches from the USAF Eastern and Western Test Ranges.[6] It becomes apparent that ETR is best suited for launches into equatorial orbits and WTR is best suited for achieving polar orbits.

Launches at ETR can also have inclinations other than 0^o. If a vehicle is launched at an azimuth of 45^o from true north, an orbit with an inclination angle of approximately 47^o will result. A satellite in an orbit inclined at 47^o would cover a groundtrack over the region of the Earth between $\pm 47^o$ latitude. From a risk standpoint, as the launch azimuth decreases, the locus of IIP moves closer to the East coast of the US. and Canada. There is also considerably more overflight of countries in the Eastern Hemisphere, with potential political and international repercussions for a space launch accident.

The lowest risk to populated areas is almost always associated with missions where the launch azimuth is perpendicular to the coastline and the wind blows in the direction of the launch. This

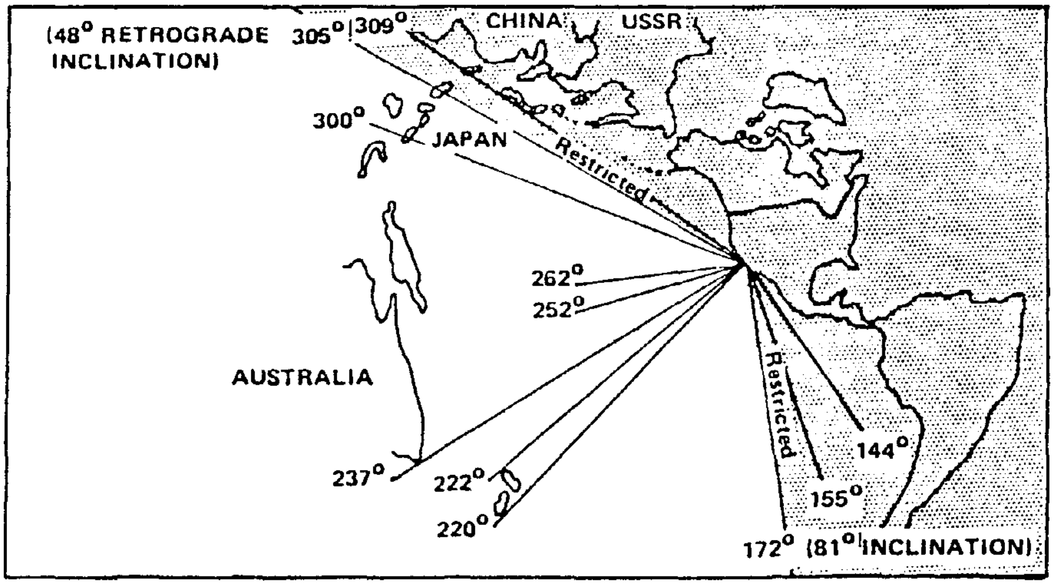


FIGURE 10-8. WTR GEOGRAPHIC LAUNCH AZIMUTH CONSTRAINTS (REF. 7)

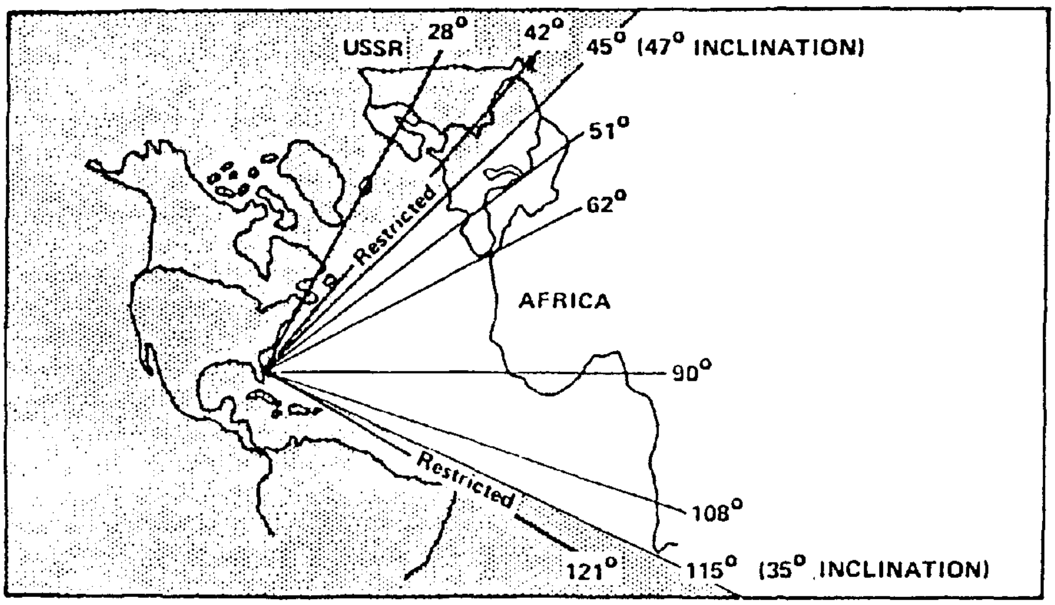


FIGURE 10-9. ETR GEOGRAPHIC LAUNCH AZIMUTH CONSTRAINTS (REF. 7)

situation is experienced with many launches at the Eastern Test Range (from Cape Kennedy or Cape Canaveral). Launches into polar orbit from Vandenberg Air Force Base have a southerly launch azimuth, which is perpendicular to the coast at the launch site, but then moves parallel to the coast as the California coastline becomes more aligned north to south. Prevailing winds in the region of the Vandenberg launch site tend to be more onshore and this must be accounted for in establishing destruct lines for Range Safety Control.

10.4 VARIATION OF RISK DUE TO MISSION PROFILE, LAUNCH VEHICLE AND PAYLOAD

10.4.1 Relative Risks of Missions

Missions can be broadly categorized in terms of their orbital parameters: inclination, eccentricity, perigee and apogee altitude. The risks associated with different final orbit inclinations are those associated with the initial launch azimuth necessary to support the sequence of boost and transfer operations needed to achieve the desired final orbit inclination. The risks associated with launch azimuth and site constraints are discussed in Section 10.3. Satellites will re-enter within a few years due to orbital decay from Low Earth Orbits (LEO), but will not from geosynchronous orbits (GEO) (See Ch. 8). Thus geosynchronous orbits offer considerably less risk from the re-entry hazard. The ELV launching a satellite into a geosynchronous orbit must carry more propellant in the initial orbiting vehicle and more stages. The additional propellant in the upper stage (up to a factor of 3) may increase the hazard by a proportionate fraction (percent) for launch accidents on or near the ground. Moreover, insertion of a payload into GEO involves more orbital maneuvers, more stages and a greater fuel load, hence greater overall risk of failing hardware and mission failure. For example, payload delivery to GEO orbit, as shown in Figure 10-12, involves firing an apogee kick motor (AKM) and a perigee kick motor (PKM). However, even if the mission fails to insert the payload into the correct final orbit, public hazards may not increase unless a highly elliptical transfer orbit leads to early uncontrolled re-entry of upper stages and payload or an on-orbit explosion creates collision hazards for GEO and LEO operational satellites.

However, for accidents at high altitude when the vehicle is near orbital, the vehicle with a geosynchronous orbit destination will have less inert debris and the propellant will probably be consumed before ground impact. Hence, in this case, the Low Earth Orbit vehicle will have a larger casualty area and offer a somewhat greater overall risk. In general, the changes in risk level due to the mission profile are relatively small, with the exception of missions requiring restricted azimuths or riskier staging and orbital maneuvers for achieving the mission objective.

10.4.2 Hazardous Characteristics of Typical ELV's

Two ELV's, Atlas/Centaur and Titan III, are the primary subjects of this discussion, although the Delta is also discussed briefly. They offer a broad range of payload lift capacity, they are the largest of the currently available vehicles and they present a variety of propulsion types and representative associated hazards. Furthermore, a hazard analysis for two plausible accident scenarios, based on a typical Delta vehicle and flight profile as a function of time after launch and down-range and altitude evolution, was presented earlier in Figs. 5-5, 5-6 of Ch.5, Vol.2.

10.4.2.1 Titan - The basic Titan III is illustrated in Ch. 5, Figure 5-4. Its central core vehicle consists of two liquid fuel stages, a Transtage and a payload. Two solid rockets (zero stage) are attached to the first core stage and these fire at liftoff and continue until their fuel is consumed. The first core stage is ignited near the end of the solid rocket burn (about 108 seconds after lift-off). After the solid rocket fuel is depleted and the first stage ignites, the empty solid motors are jettisoned (approximately 116 seconds after liftoff). The first stage continues to burn until approximately 273 seconds after liftoff, when its fuel is depleted and the stage is jettisoned. The fairing around the payload is also jettisoned at this time to reduce the weight that will have to be accelerated by the core second stage engine. The fairing is used to reduce the drag and protect the payload during ascent in the atmosphere. At the time of jettison, the vehicle is at an altitude of 400,000 feet (130 km) and is essentially out of the atmosphere. The second core stage fires up immediately and thrusts for 216 seconds. The Transtage has a restartable rocket motor used for orbital maneuvers. Various upper stages can be added for mission and payload flexibility.

During a normal mission, the only risks offered by the Titan are from vehicle hardware which is jettisoned. The impact locations and the approximate locus of IIP for launches from Cape Canaveral are shown in the map in Figure 10-5. The Stage 1 engine covers are not shown there, but are dropped off during the zero-stage solid rocket motor phase of flight. This particular launch trajectory is intended to have a minimum inclination angle in order to support transfer to a geosynchronous orbit.

The impact locations and the approximate locus of IIP for a Titan launch from Vandenberg Air Force Base are shown in the map in Figure 10-7. The requirements for "polar" orbits may not actually need fly over of the poles, but rather very high inclination angles, such as 70°. In addition, launches with inclination angles lower than 90° from VAFB can have larger payloads. Consequently, launches from VAFB may have a range of launch azimuths, as indicated in Figure 10-7, depending on the minimum orbital plane inclination angle.

The liquid fuels which propel the core vehicle and Transtage of the Titan are non-cryogenic and ~~storable: Aerozine-50 and nitrogen tetroxide used in the core vehicle are highly toxic, if released by~~ accidental venting or a spill (see Appendix B and Ch.5, Vol.2). Pre-launch and launch hazards are controlled by handling and storage regulations and by specifying optimal weather conditions for launch which permit toxic vapors and plume dispersal in case of an accident. If the vehicle is destroyed, these hypergolic propellants do not react as energetically as cryogenic propellants. The spontaneous ignition does not allow them to mix before igniting and, consequently, they burn, but have no significant explosion. However, there was an exception: On March 16, 1982, a Titan II, which is basically the first two core stages of the Titan 3, blew up in its silo at Little Rock Air Force Base near Damascus, Arkansas. A very significant explosion resulted which destroyed the entire facility. The magnitude of the explosion was ascribed to the confinement provided by the silo, which did not permit the propellants to scatter while burning. On the other hand, tests of the destruct system of the Titan have generally indicated that the unconfined burning propellants have very little explosive energy.

The more pressing problem with Titan liquid propellants is their toxicity and corrosivity. The destruction of the vehicle may produce a white and reddish-brown (Aerozine-50 and N_2O_4) cloud which is very toxic and also very harmful to vegetation.

In addition to the liquid propellants, the Titan has strap-on solid propellant motors (similar to the Space Shuttle). The emissions from these engines also contain contaminants which, in high concentrations, can be detrimental to agriculture. The main hazard associated with the solid rockets is their explosiveness, the resulting overpressure and the spread of burning debris. Unlike liquid rockets, solid rockets, once ignited, cannot be shut down without being destroyed. Destruct action will always produce a conflagration and dispersion of burning debris. An impact test of an intact Titan solid rocket booster in 1967 indicated that the resulting explosion would be equivalent to TNT having a weight of 7.5 percent of the weight of the propellant in the rocket.(7) Some individuals in the explosive safety field believe, that under the right circumstances, this equivalent yield could be doubled. Others have the opinion that, without impact at a significant velocity, the stage will have no TNT equivalence (see also Ch.5, Vol.2, for a discussion of yield uncertainties).

10.4.2.2 Atlas/Centaur - The Atlas/Centaur is illustrated in Figure 5-7. It is basically a two-stage vehicle consisting of an Atlas first stage and a Centaur upper stage. The Atlas is a liquid oxygen (cryogenic) and RP-1 (hydrocarbon) powered vehicle while the Centaur upper stage is powered by liquid oxygen and liquid hydrogen. Neither vehicle offers a toxic threat, but both are volatile, particularly the hydrogen/oxygen Centaur stage. The primary hazards are blast overpressure and debris from a potential explosion.

At lift-off, the Atlas has thrust provided by three rocket engines. After 155 seconds of flight, the two outer engines (called the boosters) are shut down and jettisoned on rails (3 seconds later). The remaining sustainer engine, which is designed to be more efficient at higher altitudes, continues until all of the fuel has been consumed. During sustainer operation, equipment which served a purpose during the operation within the atmosphere is also jettisoned. Once the sustainer engine is shut down, the Atlas stage is jettisoned, the Centaur engines are ignited and the flight continues. The Centaur has two burn periods, the first to place the Centaur and payload into orbit and the second to put the Centaur and payload into a transfer orbit. The Centaur is separated from the payload while in the transfer orbit. A solid propellant rocket (Apogee Kick Motor or AKM) on the payload may provide the final thrust to place the payload in the geosynchronous orbit; other payloads may use a liquid fueled motor for final GEO emplacement.

The same two missions which were discussed for the Titan are considered, one producing a low polar orbit and the other producing a high equatorial orbit (geosynchronous). The Atlas/Centaur is a smaller vehicle than Titan and can place about 40 percent of the Titan payload in a geosynchronous orbit. Figures 10-10 and 10-11 show the IIP loci for Atlas/Centaur missions from ESMC and WSMC during the pre-orbital phase. During a normal mission, the only hazards associated with the Atlas/Centaur launch are from the jettisoned spent stages, whose impact locations are shown in the figures.

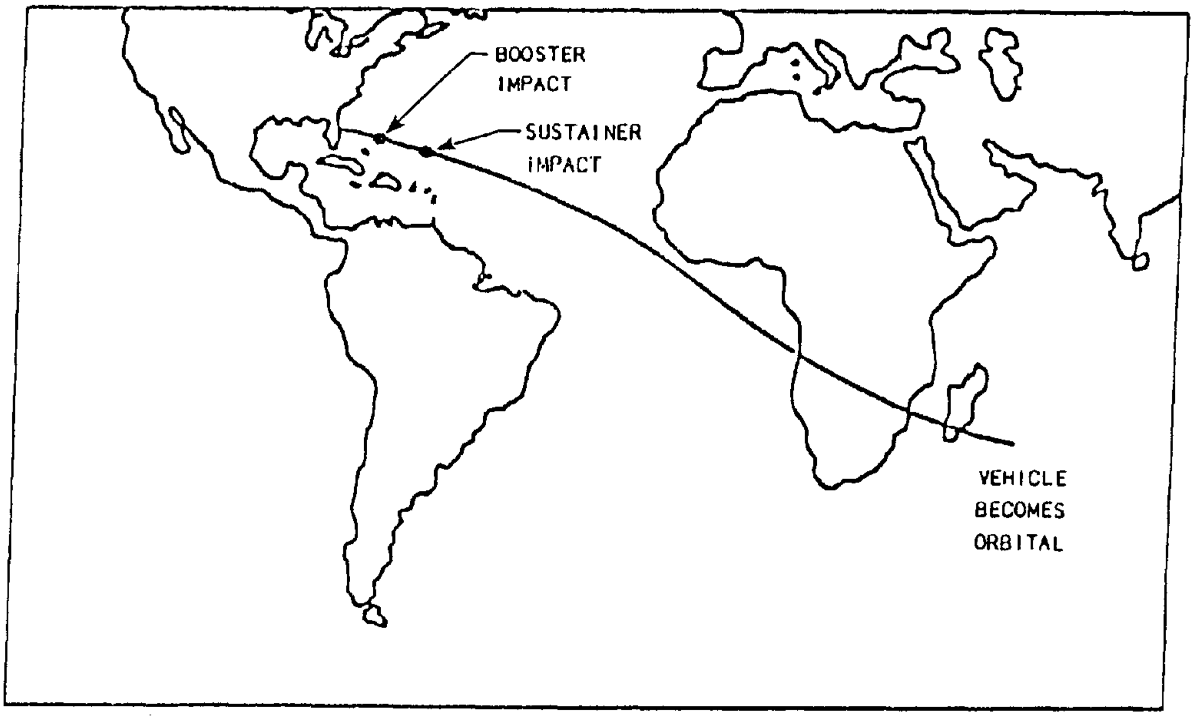


FIGURE 10-10. LOCUS OF IIP FOR A TYPICAL ATLAS/CENTAUR LANCH FROM CAPE CANAVERAL (ESMC)

The sequence of orbital events for an Atlas/Centaur FLTSATCOM mission is shown in Figure 10-12.[3] This is a mission very similar to any other Atlas/Centaur geosynchronous mission, although in this particular case, there is no initial parking orbit. The vehicle, after becoming orbital, continues to accelerate directly into the transfer orbit. Note from Figure 10-12 that the Apogee Kick Motor burn also provides the plane change necessary to achieve an equatorial geosynchronous final orbit.

The hazard potential for the Atlas/Centaur launch will decrease with time into mission as the vehicle and payload gain altitude and propellant is consumed (see Figs. 5-5 and 5-6 in Ch.5, illustrating the risk vs. time for a Delta vehicle). The RP-1 propellant will not be absorbed into the atmosphere, but it

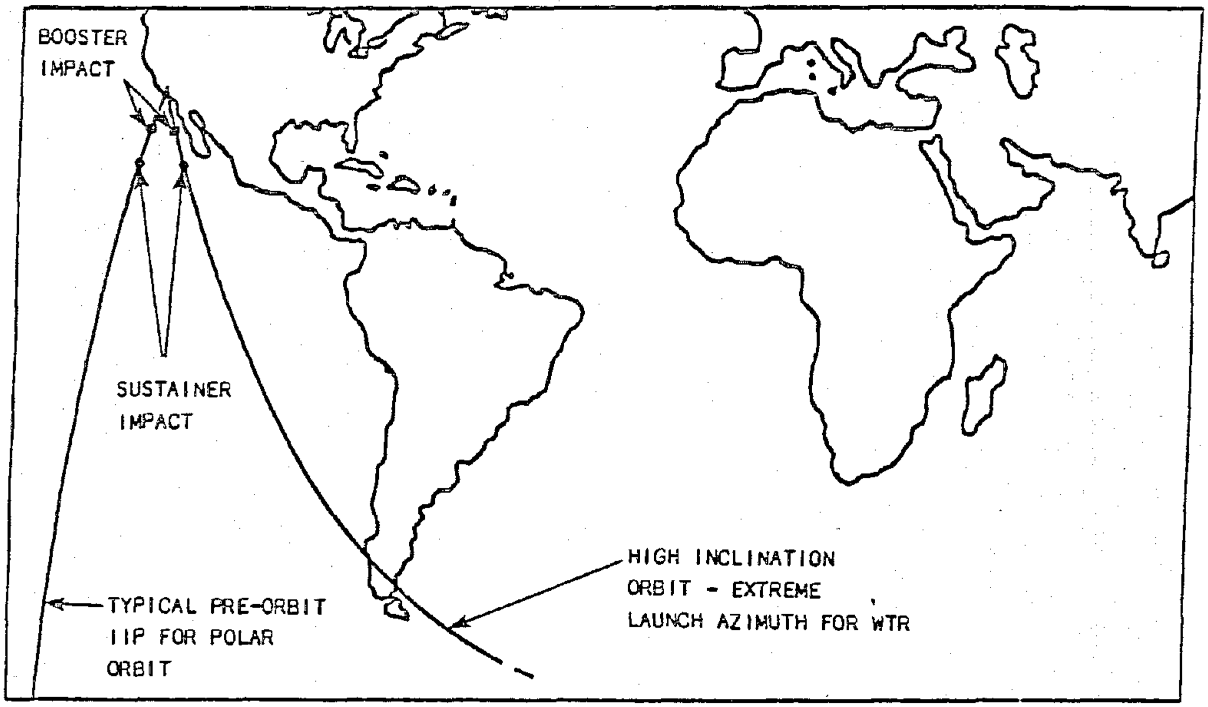


FIGURE 10-11. LOCUS OF IIP FOR A TYPICAL ATLAS/CENTAUR LAUNCH FROM VANDENBERG AIR FORCE BASE (WSMC)

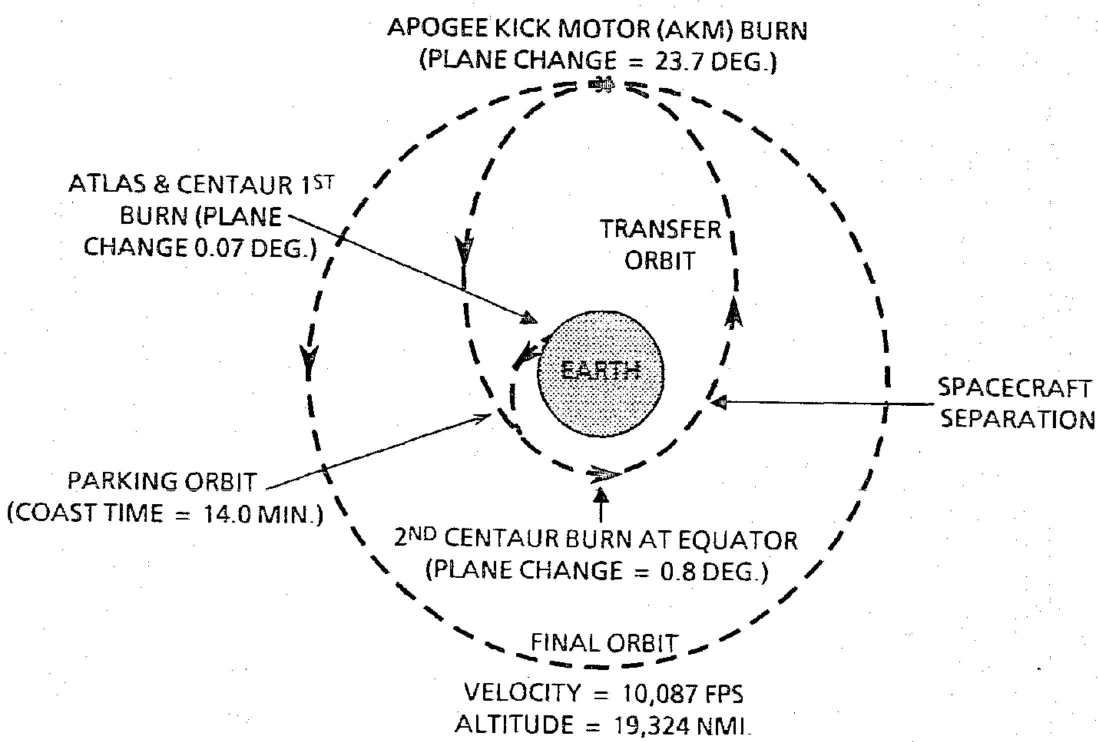


FIGURE 10-12. SEQUENCE OF EVENTS FOR THE ATLAS/CENTAUR ON A GEOSYNCHRONOUS MISSION

will become more widely dispersed as the vehicle reaches a higher altitude. Note that RP-1 fuel is not toxic or corrosive in the same sense as hypergolic liquid propellants.

Fewer pieces of debris are expected from an Atlas/Centaur destruct than for a Titan. This is because of its smaller size and it uses only liquid rocket engines. However, the structure of the Atlas is more fragile than that of the Titan and will most likely break into more pieces than the Titan core vehicle. The very thin Atlas skin pieces will probably scatter more in the wind than the Titan pieces and, consequently, the low ballistic coefficient portion of the Atlas debris pattern will show greater dispersion. In this case, greater dispersion does not mean greater risk to ground objects since Atlas debris are lighter and smaller.

If a failure occurs during the Centaur sustainer burn phase of the flight and no destruct action takes place, the vehicle may remain somewhat intact, depending upon its altitude at that time and on the nature of the failure. Normally, the airloads during the fall will cause vehicle breakup. If this occurs, the propellants will be dispersed and the only hazard will be from impacting "inert" debris. If the tanks were to remain intact, some explosion might occur at ground impacts. However, it is very unlikely that the tanks will remain intact under high airloads given their structural vulnerability. The principal hazard anticipated is damage from impacting debris. If the vehicle is destroyed by a destruct command, there will be more numerous pieces of debris, but the vehicle will not have been allowed to wander over a possibly populated area.

For launches of geosynchronous satellites from Cape Canaveral, the IIP will move over Africa late in pre-orbital flight, as described for the Titan in Section 10.2.3. The previous discussion of debris impact hazards to Africa and South America is also applicable to Atlas/Centaur, except that it will have less massive debris and the risks may be reduced by as much as a factor of two.

10.4.2.3 Delta - The Delta launch vehicle offers the variety of propellants and components of both the Titan and the Atlas/Centaur vehicles. The Delta has strap-on solid propellant boosters (Castor 4 for Stage 0), a core booster stage (Stage 1) which uses cryogenic liquid oxygen and RP-1, an upper stage (Stage 2) which uses liquid storable propellants (Aerozine-50 and N_2O_4) and a Stage 3 which has a solid rocket motor. The Delta has been launched in a variety of configurations with different numbers of solid rocket boosters and different upper stages. For example, the enhanced Delta configuration, illustrated in Ch.4, Vol.1, has the capability to place 5,500 lbs. of payload into a Low Earth Orbit and 2,800 lbs. of payload in a Geosynchronous Transfer Orbit. The hazards from a typical Delta launch failure have been discussed qualitatively and illustrated quantitatively in Ch.5, Vol.2.

From a comparative risk standpoint, most of the elements of the Delta are on a smaller scale, but there are more of them: there is considerably less hypergolic propellant than on the Titan (see Ch.4 and App. B); there are solid boosters as on the Titan, but they are much smaller and more numerous; there is also less cryogenic propellant in the vehicle than the Atlas/Centaur and there is no explosive and combustible liquid hydrogen fuel. A strap-down inertial guidance system provides guidance throughout booster and upper stage flight. The Delta was considered the most reliable ELV by NASA with an overall failure rate of 6.7 percent, due to 12 failures out of 181 launches; only four launch failures required destruct action. Only six failures led to re-entry of various stages and payload and only one of the six led to ground impact, but no damage was reported (see Table 3-5, Cap. 3, Vol. 1). A discussion of ELV reliability and the implications for public safety from the historical launch statistics were also discussed in Ch.3, Vol.1) The most recent launch accident (Delta 178, on May 3, 1986 at Cape Canaveral) occurred 71 seconds after launch when the main engine was prematurely shut-off by an electrical short, the vehicle tumbled out of control and had to be destroyed by Range Safety (see Ref.to Mishap Report, Ch.9). The NOAA weather satellite GOES-G payload was destroyed; no damage or injury resulted from debris.

10.4.3 Payload Contributions to Launch and Mission Risk

The payload can contribute to overall launch and mission hazards in several ways:

(1) The payload can initiate a malfunction in the launch vehicle by causing a failure (e.g., electrical short or surge) or an explosion during launch which could affect the rest of the vehicle. Generally, the payload is unlikely to cause a launch vehicle failure.

(2) The payload could contribute to the amount of the hazardous material resulting from the accident. Normally this would be in the form of propellant, but if a nuclear heat source is considered, the debris from an accident could present a significant radioactive hazard (see Chs. 7 and 8).

(3) The payload could re-enter and impact on land along with other destruct debris, in case of a launch failure that requires destruct action.

Any payload-related hazards to the public will have to be identified, examined, quantified and managed to tolerable levels as part of the DOT/ OCST licensing safety audit (see Ch.1, Vol.1).

10.5 BENEFITS OF RANGE SAFETY CONTROL

10.5.1 Range Safety Control System Reliability

Range Safety Control systems have played a very important role in the success of the space program. Combined with an outstanding Risk Prevention and Control program, their success has been such that there have been no casualties resulting from in-flight launch vehicle failures. As mentioned in Ch. 4, this is due to both mission planning and to the design standards and performance reliability of the Flight Termination Systems (FTS). The USAF design goal for FTS hardware reliability is .999 at a 95% confidence level for WSMC and ESMC, whereas the WSMR design goal for sub-orbital ELV's is .997 to the same confidence level (see Ch.8 and Ch.9 discussions of reliability vs. safety). Performance testing and verification of the FTS reliability depends on the number of such failures, environmental stress during testing or accident and on other accident specifics. The reliability that has been achieved is due in part to the redundancies built into both the ground and airborne components of the systems. There are no published figures on the operational reliability of Range Safety systems, but with hundreds of vehicles destroyed with no system failures, one could conclude that the probability of system failure is less than 1 in 1000.

10.5.2 Loss and Casualty Potential When Range Safety Controls Are Not Used

The following is intended to discuss worst case loss situations for space launches, assuming that vehicles are launched and fail over communities and that Range Safety Controls (chiefly a Flight Termination System provided on-board the ELV, as described in Ch.2, Vol.1) are not in place. A computer model, Community Damage (COMDAM), was developed for this special purpose. The concept for this model is shown in Figure 10-13. The model is deterministic, not probabilistic (see Ch. 8), i.e., given a catastrophic ELV failure and the absence of a destruct system, it examines the nature and severity of possible consequences of interest, namely a conditional casualty expectation. In reality, implementation of Range Safety restricts launch azimuths as well as decreasing the likelihood of any accident that could have public impacts (see Ch.9).

The launch vehicle is assumed to overfly and fail above a community located in the vicinity of the Range. This model might apply to evaluating damage from debris impacting in the vicinity of a Range, say, to Santa Barbara or the Channel Islands near WSMC, or to Miami Beach near ESMC, or to Albuquerque near WSMR. These scenarios are obviously unrealistic because launch vehicles are neither allowed to overfly populated areas nor allowed to proceed without certified Flight Termination Systems. On the other hand, COMDAM may afford insight into the potential of unconstrained launch operations for accidental casualty and property loss.

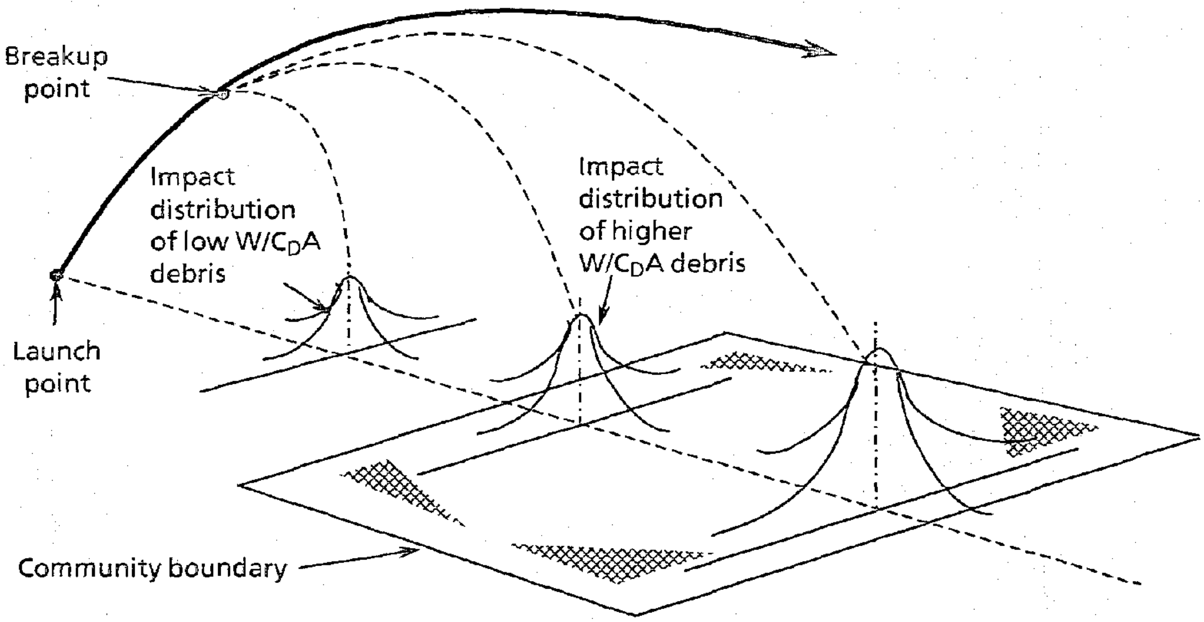


FIGURE 10-13. COMMUNITY DAMAGE MODEL OF DEBRIS LANDING ON A COMMUNITY - COMDAM

For simplicity, the hypothetical community at risk is laid out as a square, with several types of structures spaced evenly over the area within the community boundaries. The ELV is assumed to fail and break into pieces spontaneously due to aerodynamic stress. These fragments must be classified according to their ballistic coefficient and explosiveness (if solid propellant). The debris can be dispersed by scattering (lift/drag) effects and velocity impulses which may be imparted to the debris at the time of an explosive in-flight failure. If a piece of debris impacts the ground and explodes, the overpressure (P) and impulse (I) are computed on all of the adjoining structures (see also Ch.5, Vol.2). The explosive damage to each structure is computed using the formula $D = a(P^b)(I^c)$, where D is the percent damage and the coefficients a, b and c are unique for each different structure class and were developed from data gathered from explosive accidents.[9,10] If the structure is calculated to be more than sixty percent damaged, it is assumed that it must be totally replaced and, thus, equivalent to being 100% damaged. The dollar loss is obtained by multiplying percent damage times the average building value.

For damage due to inert (non-explosive) debris, kinetic energy thresholds are set. If the kinetic energy of an impact fragment did not exceed a pre-specified level, it is assumed not to penetrate the structure and cause any damage. If it did exceed the threshold, the damage to the structure is assumed to be the ratio of the area of the fragment to the projected area of the structure. Casualty expectations, E_C, were computed using the model developed in Ref. 13.

The flow diagram for this specifically adopted analytical procedure is shown in Figure 10-14. These algorithms and logic can be programmed and used to estimate the approximate expected losses and casualties similar to those discussed above. One of the reasons for developing such an unrealistic worst-case consequence model was to show several effects, such as:

1) the change in total losses as a function of the time of launch vehicle failure:

2) the effect of the distance from the point of launch on the population center at risk; and

3) the influence of exploding debris.

The COMDAM numbers must be treated as approximate at best, and illustrative only, since no specific community has been considered and the consequences of accidents can vary significantly even under essentially the same conditions. The financial (dollar loss) consequence estimates consider only damage, and not business interruption costs.

It should be noted that the above model accounts for structural damage produced by:

1 - direct impact of inert fragments

2 - blasts triggered by the explosion of burning fragments upon impact with ground.

Damage mechanisms not included in the model are:

a - fires initiated by burning fragments upon impact with ground (e.g., brush fires, gas main explosions and fires).

b - vapor clouds produced by burnt/unburned propellants .

c - blast and fire ball produced in the air at the instant of vehicle breakup.

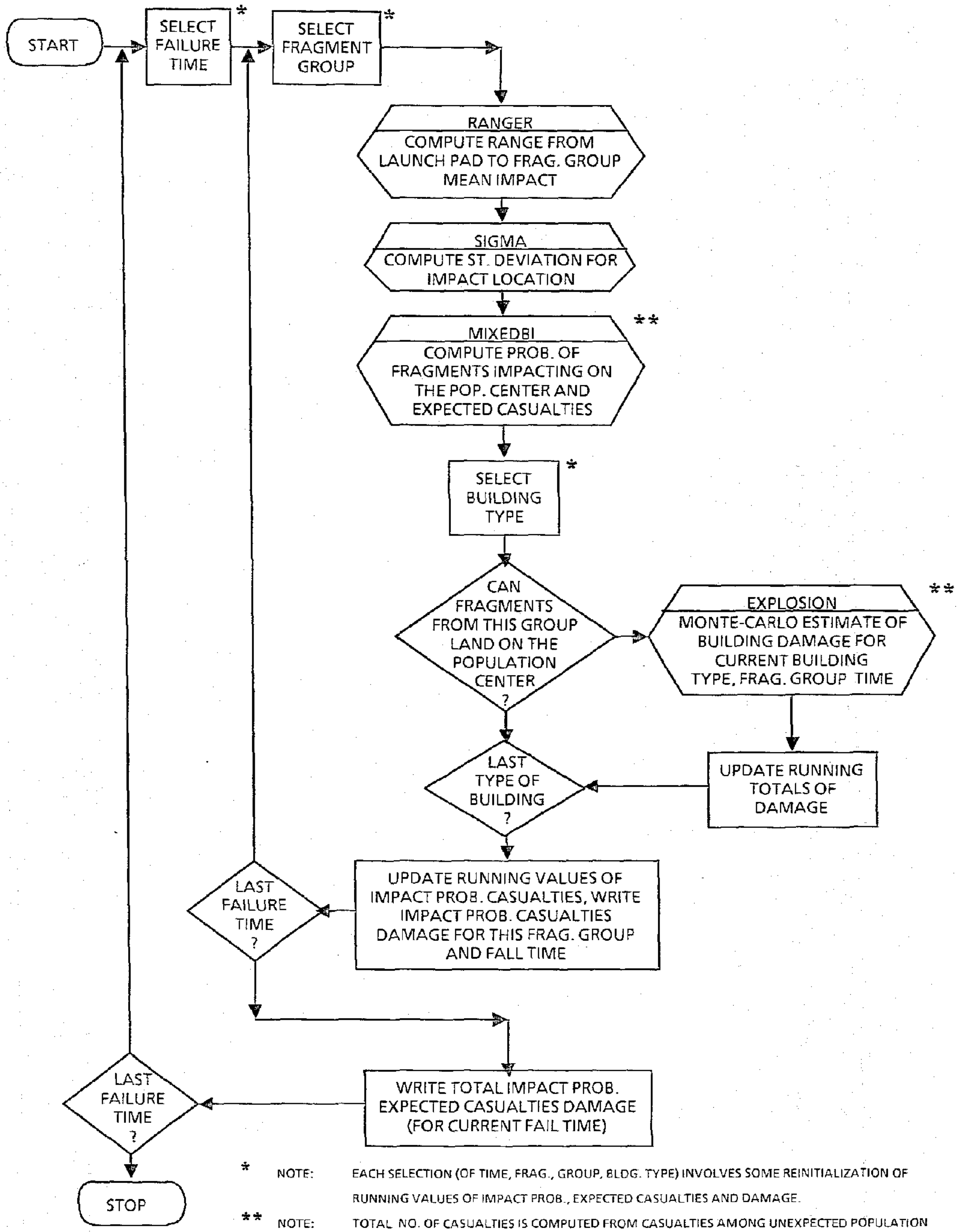
This COMDAM model does not predict what would occur realistically, but rather what is the worst that could happen. With the addition of launch azimuth restrictions enforced to avoid land overflight, the provision of a highly reliable FTS on-board the ELV and an effective ground-based Range Safety Control network, such public damage and casualties as a consequence of launch accidents become highly unlikely.

10.5.3 Comparison of Risk Acceptability

MIL-STD-882B provides only qualitative definitions of the severity and frequency of accidents for the purpose of risk assessment.[12] These definitions are reproduced in Tables 10-1 and 10-2, since they could be used to demonstrate the relative acceptability of risks from launch vehicles both with and without Range Safety Controls in place.

Although these qualitative definitions apply to military systems including space system certification, acceptance and failure risk analysis, they can also be applied to hazard assessment for commercial launches.

Tables 10-3 and 10-4 give two examples from MIL-STD-882B for risk acceptability, in the form of a



* NOTE: EACH SELECTION (OF TIME, FRAG., GROUP, BLDG. TYPE) INVOLVES SOME REINITIALIZATION OF RUNNING VALUES OF IMPACT PROB., EXPECTED CASUALTIES AND DAMAGE.

** NOTE: TOTAL NO. OF CASUALTIES IS COMPUTED FROM CASUALTIES AMONG UNEXPECTED POPULATION (COMPUTED BY SUBROUTINE MIXEDBI) SUMMED WITH CASUALTIES TO PERSONS IN BLDG'S (BASED ON THE PERCENT OF BUILDING DESTROYED)

FIGURE 10-14. GENERAL FLOW DIAGRAM OF COMMUNITY DAMAGE MODEL, COMDAM

TABLE 10-1. HAZARD SEVERITY DEFINITIONS (MIL-STD-882B)

Description	Category	Mishap Definition
Catastrophic	I	Death or system loss.
Critical	II	Severe injury, severe occupational illness or major system damage.
Marginal	III	Minor injury, minor occupational illness or minor system damage.
Negligible	IV	Less than minor injury, occupational illness or system damage.

TABLE 10-2. HAZARD PROBABILITY DEFINITIONS (MIL-STD-882B)

Description (1)	Level	Specific individual item	Fleet or inventory (2)
Frequent	A	Likely to occur frequently.	Continually experienced.
Probable	B	Will occur several times in life of an item.	Will occur frequently.
Occasional	C	Likely to occur sometime in life of an item.	Will occur several times.
Remote	D	Unlikely, but possible to occur in life of an item.	Unlikely, but can reasonably be expected to occur.
Improbable	E	So unlikely it can be assumed occurrence may not be experienced.	Unlikely to occur, but possible

hazard risk assessment matrix.[12]

The next step is to find the risk associated with ELV launches in the hazard frequency/acceptability format exhibited in the previous four tables. When a vehicle (e.g., Titan, Atlas/Centaur or Delta) is not under Range Safety Control, there is potential for catastrophe if the vehicle fails fairly early in flight near or over a community. Since all prospective commercial launch vehicles have a historical launch failure frequency of more than 4 percent (range from 4 to 14 percent) (see Ch. 3, Vol 1), this must be considered an "occasional event." With the Range Safety Control System in place, there is potential for catastrophe only when this system fails to perform its function. Given the proven reliability of modern Range Safety Control systems, the occurrence of a accidental failure with major public safety impacts must be considered improbable or remote.

TABLE 10-3. FIRST EXAMPLE, HAZARD/RISK ASSESSMENT MATRIX (MIL-STD-882B)

Frequency of occurrence	Hazard Categories			
	I Catastrophic	II Critical	III Marginal	IV Negligible
(A) Frequent	1A	2A	3A	4A
(B) Probable	1B	2B	3B	4B
(C) Occasional	1C	2C	3C	4C
(D) Remote	1D	2D	3D	4D
(E) Improbable	1E	2E	3E	4E

Hazard Risk Index	Suggested Criteria
1A, 1B, 1C, 2A, 2B, 3A	Unacceptable.
1D, 2C, 2D, 3B, 3C	Undesirable (Management Authority Decision Required).
1E, 2E, 3D, 3E, 4A, 4B	Acceptable with review by management authority.
4C, 4D, 4E	Acceptable without review.

TABLE 10-4. SECOND EXAMPLE, HAZARD/RISK ASSESSMENT MATRIX (MIL-STD-882B)

Frequency of occurrence	Hazard Categories			
	I Catastrophic	II Critical	III Marginal	IV Negligible
(A) Frequent	1	3	7	13
(B) Probable	2	5	9	16
(C) Occasional	4	6	11	18
(D) Remote	8	10	14	19
(E) Improbable	12	15	17	20

Hazard Risk Index	Suggested Criteria
1 - 5	Unacceptable.
6 - 9	Undesirable (Management Authority Decision Required).
10 - 17	Acceptable with review by management authority.
18 - 20	Acceptable without review.

As the vehicle progresses from launch toward achieving orbit, the associated risk to the public is reduced, as discussed in Section 10.2.3. At this stage the Range Safety System provides little or no benefit, because the debris produced from high altitude destruct action will be similar to that without destruct and there is no way to restrict the impact location of the debris. Consequently, both with and without a Range Safety Control System, the risk to the public is approximately the same in the pre-orbital and orbital stage, a marginal hazard with a remote probability of occurrence. In returning from orbit (uncontrolled re-entry), there is no possibility of Range Safety Control and the public risk is again marginal, with a remote probability of debris causing any casualties.

These conclusions about the relative public risks associated with ELV launches are summarized in Table 10-5 using the definitions of hazard, frequency and acceptability as specified in MIL-STD-882B.[12]

TABLE 10-5. RELATIVE RISKS FOR VARIOUS FLIGHT PHASES WITH AND WITHOUT RANGE SAFETY SYSTEMS

Flight Phase	Without Range Safety Control			With Range Safety Control		
	Hazard level	Frequency	Acceptability	Hazard Level	Frequency	Acceptability
Early Launch	Potentially catastrophic	Occasional	Unacceptable	Potentially catastrophic	Improbable	Acceptable
Pre-orbital	Marginal	Remote	Acceptable	No benefit	No benefit	No benefit
Return from orbit (uncontrolled)	Marginal	Remote	Acceptable	No Possible control	No Possible control	No Possible control

The conclusion is that a Range Safety Control Systems must be in place so that normal, though relatively low probability, launch failures become tolerable and permissible from the point-of-view of public safety.

Figure 10-15, reproduced from Ref 14, is a Public Launch Hazard Event Tree based on ESMC launch experience, but it also applies conceptually the the other National Ranges. It shows that a long chain of failure events must take place to expose the public to launch or overflight hazards. Conditional probabilities and branching of events are also indicated. This type of analysis will be applied to evaluate the safety risks associated with specific ELV's, launch sites and missions.

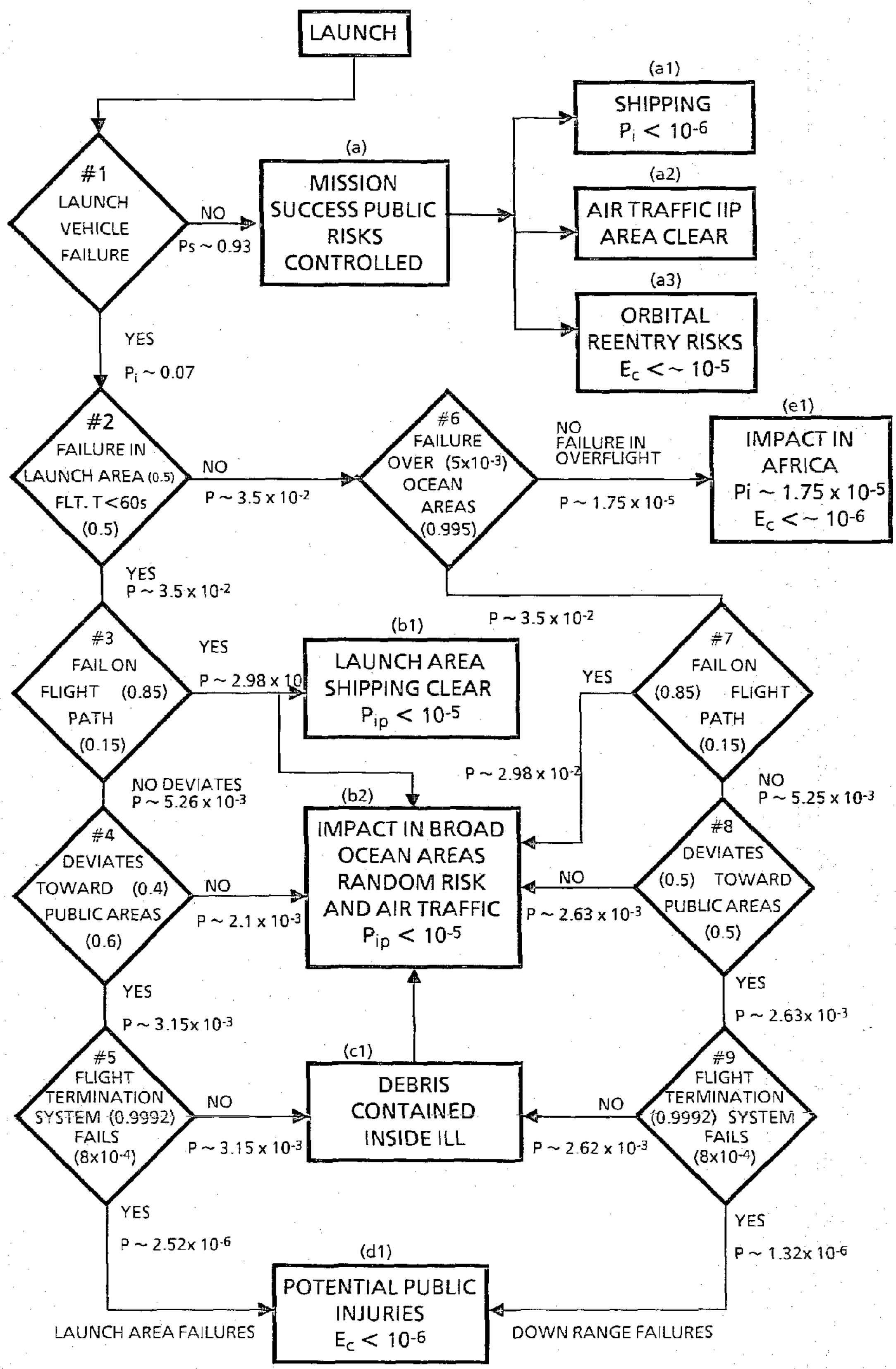


FIGURE 10-15 PUBLIC LAUNCH HAZARD EVENT TREE

REFERENCES TO CHAPTER 10

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2. a) "Titan Analysis Unable to Show Cause of Insulation Separation from Casing," Av. Week and Space Technology, July 7, 1986.

   b) "Titan Accident Disrupts Military Space Program," Science 232, 702, May 9, 1986;

   c) "Air Force Determines Potential Cause of Titan Explosion," Science 233, 78 July 18, 1986.

   d) "Titan Solid Booster Failure Caused Vandenberg Accident," Av. Week and Space Technology, p. 24, May 5, 1986.

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9. "Range Safety and Range Planning Trajectory Data for the AC-68/F-8 Atlas G/Centaur FLTSATCOM Follow-On Mission," Report No. GDSS-SP-87-010, General Dynamics Space Systems Division, San Diego, California, May 1987.

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12. "Military Standard, System Safety Program Requirements," MIL-STD-882B, US Dept. of Defense, 30 March 1983.

13. Baeker, J.B., J.D. Collins and J.M. Haber, "Launch Risk Analysis," Journal of Spacecraft and Rockets, 14, pp. 733-738, Dec. 1977.

14. "Baseline Assessment - ESMC," RTI Draft Report for DOT/OCST, May, 1988.

APPENDIX A

GLOSSARY AND DEFINITION OF TERMS

Many documents have been referenced to obtain the definitions of terms that are used in this document. In most cases, the definitions from the referenced documents have been used directly, while others have been modified to more fully apply to the text herein, and where appropriate, some have been developed by the authors.

The referenced documents are as follows:

1. AFETRM 127-1, Sept. 1972
2. MIL-STD-882, March 30, 1984
3. WSMCR 127-1, May 15, 1985
4. ESMCR 127-1, July 30, 1984
5. NASA GHB 1771.1, Sept. 14, 1984
6. Federal Register, Vol. 51, No. 38, Part 401.5
7. Public Law 98-575, Oct. 30, 1984
8. UMTA System Safety Glossary, June,1986
9. The Aerospace Age Dictionary, 1965
10. The Dictionary of Space Technology, by J.A. Angelo Jr., 1982
11. CPIA 394, Sept. 1984

ACCIDENT (MISHAP) - An unplanned and undesirable event that results in injury; death (casualty) or damage to facilities, equipment, the launch vehicle or public property.

ANALYSIS - Technical procedure, following a prescribed pattern.

ASSESSMENT - Consideration of the results of an analysis in a broader context to determine and evaluate their significance.

AEROZINE-50 (A-50) - A liquid propellant fuel; a mixture of 50% (by weight) hydrazine and 50% asymmetrical dimethylhydrazine.

AVERAGE FAILURE RATE - Frequency of failure averaged over the time interval of operation (or the number of duty cycles) for a component, system or subsystem.

BLAST - Brief and rapid movement of air or fluid away from a center of outward pressure, as in an explosion; the pressure accompanying this movement.

CASUALTY EXPECTATION - The probability of a casualty for a probable (or credible) accident scenario under consideration

CREDIBLE CONDITION - A condition that can occur and is reasonably likely to occur.

CREDIBLE ACCIDENT - A probable, possible and/or plausible accident scenario, or sequence of failure events which can lead to the occurrence of accidents.

CREDIBLE FAILURE - A failure mode which can be foreseen as possible and probable.

CRITICAL DIAMETER - The diameter of a confined or unconfined material below which an explosive reaction will not propagate when subjected to induced shock.

CRITICAL FUNCTION - As applied to nuclear and space launch systems, those functions which apply directly to, or control, mission success or failure (e.g., functions that enable, pre-arm, arm, unlock, release or guide).

CRYOGEN - A liquid which boils at temperatures of less than about 114°K (-254.4°F) at atmospheric pressure, e.g., hydrogen, helium, nitrogen, oxygen, air or methane.

DAMAGE - A loss, negative outcome or undesirable impact of an accident. May refer to equipment, property, monetary or production loss.

DEFLAGRATE - Burn at a rapid rate, but below the speed of sound in the unreacted medium.

DELPHI ANALYSIS - A method of risk assessment which requires experts' opinions and consensus-building; term derives from the ancient Greek Delphi oracle which could predict the future.

DETONATION - An exothermic reaction that propagates with such speed that the rate of advance of the reaction zone into the unreacted material exceeds the velocity of sound in the unreacted material. The rate of advance of the reaction zone is termed detonation velocity. When this rate of advance attains a value that will continue without diminution through the unreacted material, it is termed the stable detonation velocity. When the detonation velocity is equal to or greater than the stable detonation velocity of the explosive, the reaction is termed a "high order" detonation. When it is lower, the reaction is termed a "low order" detonation.

DEVIATION - An alternate method of compliance with the intent of satisfying specific requirements. A procedure differing from established norms and practices.

DYNAMIC PRESSURE - The air pressure which results from the mass air flow (or wind) behind the shock front of a blast wave. It is equal to the product of half the density of the air through which the blast wave passed and the square of the particle (or wind) velocity behind the shock front as it impinges on the object or structure.

EQUIVALENT WEIGHT (EW) - The amount of a standard explosive which, when detonated, will produce a blast effect comparable to that which results at the same distance from the detonation or explosion of a given amount of material whose performance is being evaluated. It is usually expressed as a percentage of the total weight of all reactive materials contained in the item or system. It is conventional to use TNT for comparison.

EVENT - A specific occurrence that is defined by a time and location.

EXPECTED LOSS - The probable loss or damage/casualty level for the accident scenario under consideration.

EXPENDABLE LAUNCH VEHICLE (ELV) - A launch vehicle (configuration of rocket motors) intended to be used only once, because the majority of its components are expected to be destroyed or discarded after the launch, during orbit insertion and/or re-entry.

EXPLOSION - A rapid expansion of matter into a volume greater than its original one, accompanied (in air) by loud sounds.

EXPLOSIVE - Any chemical compound or mechanical mixture which, when subjected to heat, impact, friction, detonation or other suitable initiation, undergoes a very rapid chemical change with the production of large volumes of highly heated gases which exert pressures in the surrounding medium. The term applies to materials that either detonate or deflagrate.

FAILURE - A condition of a component, subsystem or system in which the intended design or specified operation is not met.

FAILURE ANALYSIS - The process by which the cause, effect, responsibility and cost of an accident is determined and reported. A method to identify the types of faults or malfunctions that may occur and lead to accidents.

FAILURE MODE - A specific failure for a critical component, subsystem or system which can be foreseen or identified.

FAILURE MODE AND EFFECT ANALYSIS (FMEA) - An inductive procedure in which potential malfunctions are identified and then analyzed as to their possible effects.

FAULT TREE ANALYSIS (FTA) - A deductive analysis procedure which graphically presents all possible sequences of failures and chains of events which can result in the final undesired event (accident) at the top of the tree; used to determine possible and most probable causes.

FIREBALL - A more or less spherical ball of flames produced by the instantaneous release, evaporation and ignition of propellants. Generally, the fireball expands and rises in the atmosphere until the propellant is consumed.

FIREBRAND - A projected burning or hot fragment whose thermal energy is transferred to a receptor.

FLAMMABLE LIMITS - The upper and lower vapor concentrations of fuel to air which will ignite and burn (i.e., deflagrate) in the presence of external ignition sources; often referred to as the explosive range, although they are not identical.

FLASH EVAPORATION - The changing of a liquid propellant into a gas when the external pressure is released during the rupture of a vessel.

FLIGHT - That period of time beginning with engine ignition and continuing until earth impact for suborbital or orbital trajectories, or indefinitely for deep space trajectories.

FLIGHT AZIMUTH - The angular direction of the launch and flight trajectory of a launch vehicle measured in degrees from true north.

FLIGHT CORRIDOR - Two-dimensional area on Earth's surface (ground track) above which a launch vehicle can fly safely.

FLIGHT PATH - The path traversed through the atmosphere or through space by a launch vehicle or spacecraft.

FLIGHT PLAN - Description of the proposed launch and its events, including description and definition of payload orbit.

FLIGHT SAFETY - Protection of the public health and safety and safety of property during the flight of the launch vehicle and its payload.

FLIGHT TERMINATION SYSTEM (FTS) - Explosive or other disabling equipment installed in the ELV stages plus associated ground equipment for tracking and terminating the flight should it become necessary in order to protect people and property on the ground from a malfunctioning ELV. Also called Flight Safety Control System. A Thrust Termination System is a special type of FTS which shuts down the propulsion system.

GEO - Geosynchronous or Geostationary Earth orbit; equatorial, high altitude Earth orbits in which a satellite rotates with Earth's spin period, thus appearing stationary with respect to its sub-Earth point.

GROUND TRACK - The projection of a spacecraft launch, flight and orbital trajectory onto the surface of the Earth, traced by the motion of its sub-Earth point.

HAZARD - Any existing or potential condition that can cause injury or death, that leads to risk of damage to or loss of equipment or property. Also; A source of potential damage or harm, in case of an accident.

HAZARD ANALYSIS - An analysis performed to identify hazardous conditions for the purpose of their elimination or control.

HAZARD MANAGEMENT - An element of the system safety management function that evaluates the safety effects of potential hazards by considering acceptance, control or elimination of such hazards.

HAZARDOUS CONDITION - A situation where, because of the nature of the equipment, facilities, personnel, environment or operation being performed, there is a potential for an accident. For example, hazardous conditions may exist:

1. During propellant transfer to or from the ELV, whenever work is in progress on a rocket containing propellant and whenever a solid propellant motor is in a propulsive state.
2. During installation, electrical connection, testing and handling of ordinance items also, while ordinance items are electrically connected in the missile.
3. Whenever vehicle pressurization systems fail to satisfy safety factors.
4. Whenever any toxic or flammable materials are used for any purpose in ELV handling areas.
5. Any time that electrical storms are within five miles of the launch complex.

HAZARDOUS EVENT - An accidental occurrence that endangers people or property

HAZARDOUS EVENT PROBABILITY - The likelihood, expressed in quantitative terms, that a hazardous event will occur. Both units of frequency, (1/ time) and probability (dimensionless), can be used. See also next entry.

HAZARD PROBABILITY - The probability that a hazard will occur during the planned life or operation of the system. Hazard probability may also be expressed in qualitative terms using a relative ranking system, such as:

A. Frequent
B. Probable
C. Occasional
D. Remote
E. Improbable
F. Impossible

HAZARD SEVERITY - A qualitative measure of the potential consequences that could be caused by a specific hazard in case of an accident. An example of a hazard severity ranking system is:

A. Catastrophic
B. Critical
C. Marginal
D. Negligible

HYPERGOLIC - Term applied to the self ignition of a fuel and an oxidizer upon mixing with each other without a spark or other external aid.

IGNITION TEMPERATURE - The mean temperature at which a combustible material can be ignited and will continue to burn when the ignition source is removed. The ignition temperature for any one substance will vary with its particle size, confinement, moisture content and ambient temperature.

IMPACT AREA - An area surrounding an approved impact point for vehicle stages under normal operation or for destructed vehicle debris. The extent and configuration of the area is based upon the vehicle or stage dispersion characteristics.

IMPACT LIMIT LINE - A predetermined line defining a limit beyond which a failed ELV or its jettisoned spent stages will not be allowed to impact on the ground, in order to protect people or property.

IMPULSE - Blast wave parameter denoting the integral of pressure over pulse duration. It may be positive or negative depending on whether the pressure is above or below ambient.

LAUNCH - Release a powered rocket/spacecraft from a specially designed launch pad or platform.

LAUNCH ABORT - Premature and abrupt termination of a launch operation because of a potential or diagnosed failure of the launch system or noncompliance with the launch safety requirements.

LAUNCH ACTIVITY - The preparation, test or execution of launch; the operation of a launch site or both.

LAUNCH AZIMUTH - The horizontal angular direction initially taken by a launch vehicle at lift-off, measured clockwise in degrees from true north (see flight azimuth).

LAUNCH COMPLEX - The facility, usually fenced, which contains the ELV launch facilities including: the launch pad and servicing structures, the blockhouse or control building, propellant transfer equipment, support buildings (e.g., vehicle assembly building, VAB) required to support a launch.

LAUNCH CONTROL CENTER (LCC) - The facility from which launch operations are conducted and monitored.

LAUNCH CONTROL OFFICER - The individual who supervises and coordinates activities in the launch complex during prelaunch and post-launch. Also called Range Safety Officer (RSO).

LAUNCH OPERATION - Site, personnel, procedures, equipment and vehicles, which are collectively used for launch preparation or launch of a launch vehicle.

LAUNCH PROPERTY - Propellants, launch vehicles and components thereof and other physical items constructed for, or used in, the preparation or launch of a launch vehicle.

LAUNCH RANGE - A finite area along the path of a launch vehicle beginning at a launch site and ending at a point where the vehicle impacts on Earth, achieves orbit or reaches escape velocity. Includes instrumentation throughout that area used to monitor the flight of the launch vehicle for safety and other purposes.

LAUNCH SAFETY - Protection of personnel, safety of property on the ground and of the public health and safety during and after a launch operation.

LAUNCH SERVICES - Activities involved in the preparation of a launch vehicle and its payload (including assembly, test, integration and environmental protection) for launch and the conduct of a launch.

LAUNCH SITE - The geographical location from which a launch takes place, as defined in any license issued or transferred by DOT. Includes all facilities located on a launch site which are necessary to conduct a launch. See also Launch Complex.

LAUNCH SITE OPERATOR - A sponsoring or contractor organization (government or commercial) which has the demonstrated capability to satisfactorily conduct a launch operation safely from a particular launch site.

LAUNCH VEHICLE - Any rocket propulsion or similarly capable vehicle constructed for the purpose of inserting a payload in a ballistic or orbital trajectory.

LICENSEE - The person or organization authorized by a license to conduct specified commercial launch activities and who is responsible for conducting such activities in conformance with applicable DOT regulatory requirements.

LIQUEFIED GASES - Substances which are gases at ambient conditions of temperature and pressure that have been converted to liquids under controlled pressure and temperature.

LOW EARTH ORBIT (LEO) - Orbital altitudes up to about 1,000 km. (see Ch. 6, Vol.2).

LOWER FLAMMABLE LIMIT (LFL) - The lowest concentration, by percent of volume, of a gas or vapor in the atmosphere at normal temperatures and pressures at which the gas or vapor will ignite and sustain combustion.

MISHAP - An unplanned event or series of undesirable events that result in death, injury, damage or loss of equipment and/or property. (See also ACCIDENT)

MISSION - The objective to be accomplished by a proposed launch and the general plan for achieving that objective, namely launch azimuth, site, orbital parameters, vehicle configuration, design, etc.

ORBIT INCLINATION - The angle between the plane of a particular orbit and the equator.

ORBITAL INJECTION - The sequence of operations, in time and space, whereby a vehicle achieves a combination of velocity and position so that its payload is placed into the desired Earth orbit.

ORBITAL VELOCITY - The velocity at which the centrifugal force created by the launch vehicle's motion around the Earth equals the Earth's gravity; at this point the vehicle will orbit the Earth until some other force is applied.

OBLATENESS - The deviation of the Earth's shape from a perfect sphere (flattened poles, bulging equator).

OVERPRESSURE - Blast wave parameter denoting the peak pressure rise over ambient.

PASCAL - Unit of pressure. 1kPa = 1000 Pa. 1 atmosphere = 101 kPa

POOL FIRE - A fuel film formed on the ground and burning in a turbulent diffusion flame located above the film.

PRELIMINARY HAZARD ANALYSIS (PHA) - A qualitative listing and ranking of hazards of interest.

PROPELLANTS - Balanced mixtures of fuel and oxidizer designed to produce large volumes of hot gases at controlled, predetermined rates, once the burning reaction is initiated.

PSI - Pounds per square inch, a unit of pressure. 1 atmosphere = 14.7 psi.

RESIDUAL RISK - Risk exposure levels which cannot be further reduced or eliminated by risk mitigation (management) strategies and must be accepted.

RISK - The potential for an undesirable consequence to arise from an accident occurring during a hazardous activity. Technically, Risk (R) is the product of the probability (p) or frequency (f) of occurrence and its consequence (C) (the severity of its impact).

RISK ANALYSIS - A detailed examination of systems and operations which involves both the estimation of the expected frequency or probability of adverse events and the severity (magnitude) of their consequence expressed in units of interest (property damage, casualties, down time, production or business losses). Risk analysis requires; first the identification and characterization of hazards (qualitative analysis); then a quantification and ranking of hazards in terms of the likelihood of their occurrence, severity of their consequence or their expected risk figure .

RISK ASSESSMENT- Evaluation of analytical results of Risk Analysis in a broader context.

RISK SCREENING - The ordered ranking of hazards so that acceptable risk thresholds can be defined and intolerable risk levels that require reduction and management resources can be identified.

RISK MANAGEMENT - The process used to form decisions that control risk (reduce, eliminate or accept) based on system safety analysis. The set of policy and operational control options that must be introduced in order to avoid, reduce and eliminate risks. Risk management may focus on either prevention and diminished probability of occurrence of hazardous events or on controlling the impacts of such events by emergency preparedness and response planning. Risk management options are usually selected based on cost-benefit analyses.

SAFETY ASSESSMENT REPORT (SAR) - A comprehensive evaluation of the safety risks being assumed prior to test or operation of the system. It identifies all safety features of the system, as well as the design and procedural hazards present and specific controls to be adopted.

SAFETY - A reasonable degree of freedom from those conditions that can cause injury, death to personnel, damage or loss of equipment or property; freedom from danger.

SAFETY CRITICAL - A designation placed on a system, subsystem, element, component, device or function denoting that satisfactory operation of such is mandatory to ensure a safe operation. Such a designation dictates incorporation of special safety design considerations and features. Any condition, event, operation, process, equipment or system with a potential for major injury or damage.

SAFETY OPERATIONS - Collectively the personnel, equipment, facilities, documented plans, procedures and any other resource needed for safe preparation and launch of a launch vehicle and its payload.

SHOCK WAVE - A relatively thin region of discontinuity which can propagate through fluids and solids and across which properties (pressure, velocity, density and temperature) change very rapidly.

SOLID PROPELLANTS - Solid propellants act as monopropellants. Homogeneous propellants are true solid monopropellants; each molecule contains both fuel and oxygen (e.g., nitrocellulose-containing compounds). Composite propellants are physical (not chemical) mixtures of a finely ground oxidizer in a matrix of plastic, resinous or elastomeric fuel (e.g., ammonium perchlorate in a resin binder).

SYSTEM - A composite, at any level of complexity, of personnel, procedures, materials, tools, equipment, facilities and software. The elements of this composite entity are used together in the intended operational or support environment to perform a given task or achieve a specific production, support or mission requirement.

SYSTEM SAFETY - The application of engineering and management principles, criteria and techniques to optimize safety within the constraints of operational effectiveness, time and cost throughout all phases of the system life cycle.

SYSTEM SAFETY MANAGEMENT - The element that defines the system safety program requirements and ensures the planning, implementation and accomplishment of system safety tasks and activities.

SUBORBITAL LAUNCH - A launch during which the vehicle does not achieve orbital velocity and, therefore, falls back to the Earth's surface following a ballistic trajectory after the completion of powered flight.

SUBORBITAL TRAJECTORY - The ballistic path a launch vehicle follows during a suborbital launch.

THERMAL RADIATION - Thermal energy emitted by hot surfaces or gases by virtue of their temperatures.

THRESHOLD LIMIT VALUE (TLV) - The lowest concentration level of a toxic substance at which toxic effects may develop.

TNT EQUIVALENT YIELD - Energy release in an explosion inferred from measurements of the characteristics of blast waves generated by the explosion.

TRAJECTORY - A series of points in three dimensional space relative to time that describes the exact position of the vehicle at any time with respect to Earth's surface.

UPPER FLAMMABLE LIMIT (UFL) - The highest concentration, by percent of volume, of a gas or vapor in the atmosphere at normal temperatures and pressures at which the gas or vapor will ignite and sustain combustion.

VOLATILE - A substance that has a high vapor pressure (i.e., it will readily vaporize) at a low temperature.

APPENDIX B

FUEL PROPERTIES AND CHARACTERISTICS

(from CPIA Publication 394,
"Hazards of Chemical Rockets and Propellants", by John Hopkins University,
Applied Physics Laboratory, Laurel, MD, Sept. 1984)

NAME:	LH_2 - Liquid Hyrogen
MILITARY HAZARD CLASSIFICATION:	Group III
DOT CLASSIFICATION:	Flammable Liquified gas
QUANTITY PER VEHICLE:	3,400 lb (Centaur)
APPLICATION:	Centaur

PROPERTIES AND CHARACTERISTICS

COMPOSITION:	99.79% para-hydrogen and 0.21% ortho-hydrogen.
APPEARANCE:	High purity Liquid Hydrogen is transparent and colorless.
STABILITY:	Liquid Hydrogen is chemically stable. Physically stable only when stored under suitable conditions.
FREEZING POINT:	-435°F
BOILING POINT:	-423°F
DENSITY:	0.59 lb/gal. at -423°F
CRITICAL PRESSURE:	188 PSIA
CRITICAL TEMPERATURE:	-400°F
ODOR:	None

HAZARDS

PHYSIOLOGICAL:	Human contact with liquid hydrogen or uninsulated lines can result in severe frost bite. Hydrogen gas acts as a simple asphyxiant that can be breathed in high concentrations without producing systematic effects. However, if the concentration is high enough to significantly reduce the amount of oxygen in the air, the effects of oxygen deprivation will be produced.
EXPLOSION:	Unconfined hydrogen-air mixtures generally burn rapidly without detonation. However, in confined areas or when ignition is caused by a shock source or small explosive charge, the mixture can detonate. An explosion hazard can exist if liquid hydrogen is contaminated with solid oxygen or solidified oxygen enriched air.
THRESHOLD LIMIT VALUE:	None

NAME:	LOX-Liquid Oxygen
MILITARY HAZARD CLASSIFICATION:	II
MILITARY STORAGE COMPATIBILITY:	LIQ-A
DOT CLASSIFICATION:	Non-Flammable Liquid
QUANTITY PER VEHICLE:	146,300 lb (an additional 15,300 lb)
APPLICATION:	First Stage Oxidizer

PROPERTIES AND CHARACTERISTICS

COMPOSITION:	99.5% oxygen
APPEARANCE:	Light blue transparent liquid. Boils vigorously at ambient conditions.
STABILITY:	Liquid oxygen is chemically stable, is not shock sensitive and will not decompose.
FREEZING POINT:	-361°F
BOILING POINT:	297°F
DENSITY:	9.53 lb/gal. at -297.4°F
CRITICAL PRESSURE:	737 PSIA
CRITICAL TEMPERATURE:	-181°F
ODOR:	None

HAZARDS

PHYSIOLOGICAL:	Human contact with liquid oxygen or uninsulated lines can result in severe frost bite. Oxygen gas will not cause toxic effects. Gaseous oxygen from the liquid is absorbed by clothing and any ignition source may cause flare burning.
EXPLOSION:	When mixed with liquid oxygen, all materials that burn represent explosive hazards.
THRESHOLD LIMIT VALUE:	None

NAME:	TEA (Triethyl aluminum) TEB (Triethyl boron)
MILITARY HAZARD CLASSIFICATION:	III
MILITARY STORAGE COMPATIBILITY:	LIQ-C
DOT CLASSIFICATION:	Flammable Liquid
QUANTITY PER VEHICLE:	0.17 lb
APPLICATION:	TEA in first stage main engine TEA/TEB in vernier engines

PROPERTIES AND CHARACTERISTICS

COMPOSITION:	100% TEA in main engine 15% TEA, 85% TEB in vernier engines	
APPEARANCE:	Colorless liquid	
STABILITY:	TEA reacts violently with water and organic and inorganic acids. TEB reacts violently with oxygen.	
	TEA	TEB
FREEZING POINT:	-52°F	-134°F
BOILING POINT:	+381°F	+203°F
DENSITY:	52 lb/cu. ft	43 lb/cu. ft at 70°F
FLASH POINT:	Ignites spontaneously in air at room temperature.	
ODOR:	Combustion products have pungent ammonia-like odor.	

HAZARDS

PHYSIOLOGICAL:	TEA and TEB will destroy living tissue on contact. Combustion products are highly toxic.
FLAMMABILITY:	TEA and TEB ignites spontaneously in air at room temperature.
THRESHOLD LIMIT VALUE:	Zero

NAME:	Nitrogen Tetroxide
MILITARY HAZARD CLASSIFICATION:	I
MILITARY STORAGE COMPATIBILITY:	LIQ-A
DOT CLASSIFICATION:	Poison Liquid A
QUANTITY PER VEHICLE:	6228 lb
APPLICATION:	Second stage oxidizer

PROPERTIES AND CHARACTERISTICS

COMPOSITION:	99.5% N_2O_4
APPEARANCE:	Reddish-brown liquid with yellowish to reddish-brown fumes.
STABILITY:	N_2O_4 is very stable at room temperature. At +302°F it begins to dissociate into nitric oxide and oxygen, but upon cooling it reforms into N_2O_4.
FREEZING POINT:	+ 11.8°F
BOILING POINT:	+ 70.1°F
DENSITY:	12.1 lb/cu. gal. at 68°F
CRITICAL TEMPERATURE:	1469 psia
CRITICAL PRESSURE:	+ 316.8°F
FLASH POINT:	None
ODOR:	Characteristic irritating, pungent and acid-like odor.

HAZARDS

PHYSIOLOGICAL:	N_2O_4 liquid is corrosive and can cause severe burns of the skin and eyes unless it is immediately removed. Inhalation of N_2O_4 vapors is normally the most serious hazard.
SYMPTOMS OF POISONING:	irritation of the eyes and throat, cough, tightness of the chest, and nausea - are slight and may not be noticed. Then hours afterward, severe symptoms begin; their onset may be sudden and precipitated by exertion. Coughing, a feeling of constriction in the chest, and difficult breathing are typical.
FLAMMABILITY:	N_2O_4 is a corrosive agent whose corrosiveness is enhanced in the presence of water. It is not sensitive to shock, heat, or detonation. It is not flammable in air but will support combustion.
THRESHOLD LIMIT VALUE:	3 ppm for NO_2 2.5 ppm for N_2O_4 At no time will personnel be subjected to any concentration greater than TLV.

NAME:	RP-1
MILITARY HAZARD CLASSIFICATION:	I
MILITARY STORAGE COMPATIBILITY:	LIQ-C
DOT CLASSIFICATION:	Flammable Liquid
QUANTITY PER VEHICLE:	67,000 lb (an additional 11,000 lb.)
APPLICATION:	RP-1 is a thermally stable kerosene having a very high energy content. It is used for first stage fuel.

PROPERTIES AND CHARACTERISTICS

COMPOSITION:	Hydrocarbon
APPEARANCE:	Clear liquid ranging in color from water-white to a pale yellow.
STABILITY:	A mixture of RP-1 and liquid oxygen forms a gel which may explode upon being subjected to impact or shock.
FREEZING POINT:	-40°F Max.
BOILING POINT:	350° to 525°F
DENSITY:	49.95 to 50.82 lb/ft^3 at 60°F
FLASH POINT:	110°F
ODOR:	Strong, kerosene-like

HAZARDS

PHYSIOLOGICAL:	Inhaling vapors may cause headache, dizziness or nausea. Continuous contact with the skin can cause irritation.
EXPLOSION:	A mixture of vapor and air is dangerous and should be considered as an explosive mixture.
THRESHOLD LIMIT VALUE:	500 PPM in air. At no time will personnel be subjected to any concentration greater than the threshold limit value (TLV).

NAME:	Aerozine 50
MILITARY HAZARD CLASSIFICATION:	III
MILITARY STORAGE COMPATIBILITY:	LIQ-C
DOT CLASSIFICATION:	Flammable Liquid
QUANTITY PER VEHICLE:	3892 lb
APPLICATION:	Second stage fuel

PROPERTIES AND CHARACTERISTICS

COMPOSITION:	Mixture of 50% UDMH and 50% hydrazine
APPEARANCE:	Clear, colorless liquid
STABILITY:	A-50 is thermally stable and is not shock or friction sensitive.
FREEZING POINT:	+ 18.8°F
BOILING POINT:	+ 158.2°F
DENSITY:	56.1 lb/cu. ft at 77°F
FLASH POINT:	+ 104°F
CRITICAL TEMPERATURE:	+ 634°F
CRITICAL PRESSURE:	1696 psia
ODOR:	Ammonia gas

HAZARDS

PHYSIOLOGICAL:	The liquid can be absorbed through the skin; the vapors can be inhaled. Exposure may cause irritation of the mucous membranes of the eyes, respiratory passages, lungs,and gastro-intestinal tract. Direct skin contact can cause severe burns. MMH and UDMH are convulsant agents, irritants to the respiratory tract and eyes and may irritate the skin. They are absorbed by the skin, oral and inhalation routes. Hydrazine fuels form carcinogenic nitrosamine compounds. Also, ACGIH has listed the hydrazines as "Suspected Human Carcinogens."
EXPLOSIVE:	Liquid is flammable and reacts violently with acids and oxidizing agents.
THRESHOLD LIMIT VALUE:	0.5 ppm in air. At no time will personnel be subjected to any concentration greater than the TLV.

NAME:	Oronite Extreme Pressure Additive
MILITARY HAZARD CLASSIFICATION:	None
MILITARY STORAGE COMPATIBILITY:	None
DOT CLASSIFICATION:	Flammable Liquid
QUANTITY PER VEHICLE:	5.96 lb
APPLICATION:	First-stage booster engine lubricant.

PROPERTIES AND CHARACTERISTICS

COMPOSITION:	Phosphorus, zinc, sulphur, calcium
APPEARANCE:	Transparent, light orange oil
STABILITY:	Stable at controlled storage temperature below +100°F
FREEZING POINT:	+17°F
BOILING POINT:	Not Available
DENSITY:	67.8 lb/cu. ft at 60°F
FLASH POINT:	+340°F
ODOR:	Foul, sulphur-like smell

HAZARDS

PHYSIOLOGICAL:	None. Inhaling vapors is unpleasant.
EXPLOSION:	A mixture of additive and liquid oxygen forms a gel which may explode upon being subjected to impact or shock; however, such contact does not normally occur. A mixture of additive and fuel is normal in the lubrication system and is not hazardous.
THRESHOLD LIMIT VALUE:	None

COMMENTS

This sheet has been provided to the reader for comments. To do so, cut out this sheet along the indicated line, comment on the reverse side, fold along the dotted lines so that the postmark faces out and staple together. Please be sure to include your name, telephone number and the name of the organization you are with.

We thank you for your comments.