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CHAPTER 1

Introduction

1.1 Overview

Space technology is relatively young compared to other modern technologies, such as aircraft technology. However, in only forty years this novel domain has achieved a tremendous level of complexity and sophistication. The reason for this is simply explained: most satellites, once in space, must rely heavily on the quality of their onboard instrumentation and on the design ingenuity of the scientists and engineers who produced them. Recent achievements of repairing satellites while in orbit testify to the complexity involved in space technology. The desire of humans to conquer space within the solar system will surely encourage new technological achievements that are not yet imagined.

The technical fields in which satellites are used are numerous – telecommunications, scientific research, meteorology, and others. According to the specific task for which they are designed, satellites are very different from one another. They may be in orbits as low as 200 km or as high as 40,000 km above the earth; other spacecraft leave the earth toward planets in the solar system. Satellites may be very heavy: an inhabited space station, for example, could weigh several tons or more. There also exist very light satellites, weighing 20 kg or less. Small satellites may be relatively cheap, of the order of a million dollars apiece. Despite their differences, satellites possess fundamental features that are common to all. The physical laws that govern their motion in space and their dynamics are the same for all spacecraft. Hence, the fundamental technologies that evolved from these laws are common to all.

A satellite's life begins with the specific booster transferring it to some initial orbit, called a *transfer orbit,* in which the satellite is already circling the earth. For a satellite that will stay near earth, the next stage will be to "ameliorate" the orbit; this means that the satellite must be maneuvered to reach the precise orbit for which the satellite was designed to fulfill its mission. Next, the satellite's software must check for the proper functioning of its instrumentation and its performance in space, as well as calibrate some of the instruments before they can be used to control the satellite. The final stage is the one for which the satellite was designed and manufactured.

These stages will be discussed in the next section. Understanding the meaning of each stage will help one to understand the infrastructure of the control system of any satellite. Throughout the text, the terms "satellite" and "spacecraft" (s/c for short) will be used interchangeably. The terms "geosynchronous" and "geostationary" will be used interchangeably to describe the orbit of a satellite whose period can be made exactly equal to the time it takes the earth to rotate once about its axis.

1.2 Illustrative Example

In this section, a geosynchronous communications satellite will be described in its different life stages. The U.S. *Intelsat V* and the European *DFS Kupernikus*

(Bittner et al. 1987) are good examples of a common, medium-sized satellite. Satellites of this type consist of the following main structural parts.

(1) A central body consisting of a cubelike structure with dimensions of about 1.5×2 m.
(2) Solar arrays extended in the N–S direction ($\mathbf{Y}_B$ axis), with panel dimensions of about 1.5×7 m.
(3) An antenna tower directed toward the earth ($\mathbf{Z}_B$ direction) carrying different communication payloads such as global and beacon horns, feed systems for communication, hemi/zone and spot reflectors, TM/TC (telemetry/telecommand) antenna, and others.
(4) Controllers (such as reaction thrusters) and attitude sensors (such as sun sensors) located over the central body and the solar panels.

1.2.1 *Attitude and Orbit Control System Hardware*

It is important to list the typical attitude and orbit control system (AOCS) hardware of a geostationary satellite in order to understand and perceive from the beginning the complexity of the problems encountered. This hardware may include:

(1) a reaction bipropellant thrust system, consisting of one 420-N thruster used for orbit transfer and two independent (one redundant) low-thrust systems consisting of eight 10-N thrusters each;
(2) two momentum wheels (one redundant) of 35 N-m-sec each;
(3) two infrared horizon sensors (one operating and one redundant);
(4) four fine sun sensors (two redundant);
(5) twelve coarse sun sensors for safety reasons (six redundant);
(6) two three-axis coarse rate gyros; and
(7) two three-axis integrating gyros.

An illustration of the partial control hardware of a typical geostationary communications satellite is given in Figure 1.1. Much of the control hardware is redundant in order to guarantee a reliable control system despite potential hardware failures.

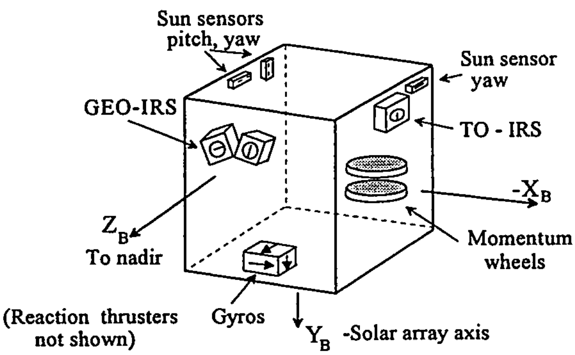

Figure 1.1 Principal arrangement of AOCS equipment; adapted from Bittner et al. (1987) by permission of IFAC.

Table 1.1 *Typical sequence stages up to the normal mode acquisition stage*
(adapted from Bittner et al. 1987 by permission of IFAC)

Event No.	Stage	Time	Event
1	Launch Stage	T_0	ARIANE lift-off, ignition of first stage
2		T_0+ 1009 s	Reorientation for payload separation
3		$T_{SEP}=T_0$+ 1122 s	S/C separation; start of S/C sequence
4	GTO Stage	T_{SEP}+ 9 min	Start of AOCS Sequences
5	Preparation to ABM Stage (Apogee Boost Motor)	T_{SEP}+ 10 m+ 20 s	Automatic beginning of Sun-acquisition,
6		T_{SEP}+ 11m to T_{SEP}+ 34 min	Sun pointing , x-axis pointing to the Sun, roll rate=0.5° /s
7		T_{SEP}+1h+5m	Solar panels deployment
8		$T_1=T_{SEP}$+37h+11m+34s	Apogee No.4 passage
9		T_1- 250m	Start gyro calibration
10		T_1- 160m	Gyrocalibration finished
11		T_1- 160m	Start GTO Earth acquisition
12		T_1- 60m	SS-bias functions loaded from ground
13	1st ABM	T_1- 24m	Apogee Boost Motor (ABM) ignition
14		T_1+ 25m	Apogee boost termination
15		T_1+ 28m	Ground comand of Sun acquisition mode
16 -30	2nd and 3rd ABM	T_{SEP}+129h+48m	Repetition of GTO Earth capture for second and fourth apogee boost maneuver until S/C in quasi-synchronous orbit.
31	Preparation to Normal Mode (Mission Stage)	$T_2=T_{SEP}$+129h+48m+ 18h	Start of Geosynchronous Orbit Earth acquisition
32		T_2	GEO Earth Acquisition command, finally Earth pointing takes place; y-axis perpendicular to orbital plane.
33		T_2+ 30m	Command for wheel run-up
34		T_2+ 45m	Wheel at nominal speed.
35		T_2+50m	Station keeping to reduce initial errors of acquisition loops
36		T_2+55m	Transition to normal mode as soon as angular and rate values within prescribed limits.

1.2.2 *Mission Sequence*

The mission events – from launch to in-orbit operation – may be summarized as follows. First is the launch into a geosynchronous transfer orbit (GTO), with perigee and apogee (low and high altitude) of 200 km and 35,786 km, respectively. This is followed by the transfer from GTO to geostationary orbit (GEO), where perigee and apogee both are 35,786 km and the orbit inclination and eccentricity are close to null. Next is the preparation and calibration of the AOCS before the useful GEO mission can start, followed by the actual GEO mission stage. Table 1.1 contains an outline of the typical major events and times related to the pre-mission stages. The significance of each event will become clear in the chapters to follow.

Figure 1.2 illustrates some of the principal stages in the geostationary transfer orbit. After separation from the launcher, the satellite is commanded into a sun acquisition mode with the $-\mathbf{X}_B$ axis pointing toward the sun. After completion of

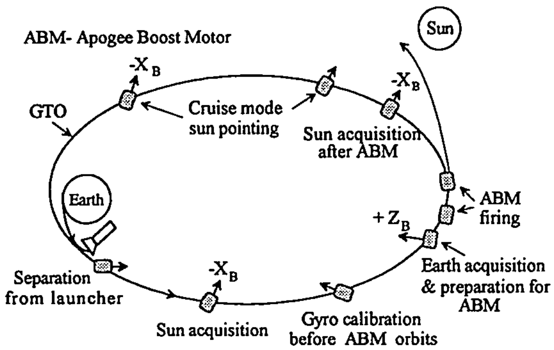

Figure 1.2 Sequence for injecting a satellite into the geostationary orbit.

this stage, the solar panels are partially or fully deployed. If fully deployed, they can be rotated about their axis of rotation toward the sun in order to maximize power absorption.

The satellite stays in this cruise mode until the first apogee boost motor (ABM) orbit is approached. In the first and the subsequent ABM orbits, several hours before the ABM firing at the apogee, the gyros' calibration maneuvers are initiated. Less than an hour before any ABM firing, earth acquisition is initiated with the $+\mathbf{Z}_B$ axis now pointed toward the earth, followed by preparation for the ABM firing stages. After ABM firings ranging from several to more than 30 minutes, the satellite is commanded to GTO cruise, sun-pointing. After the last ABM firing, the satellite is prepared for GEO operation. Some of the first maneuvers in GEO are shown in Figure 1.3. See also Bittner et al. (1989).

In the first GEO, earth acquisition is performed, meaning that the $+\mathbf{Z}_B$ axis of the satellite is directed toward the earth center of mass, thus allowing the normal GEO cruise. The momentum wheel is spun to its nominal angular velocity to provide momentum bias attitude control. In this stage, the satellite is brought to its nominal geographical longitude. The orbit is then corrected for any remaining inaccuracies in inclination and eccentricity (to be explained in Chapters 2 and 3).

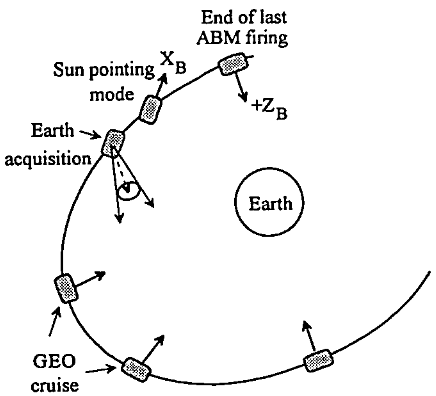

Figure 1.3 Principal stages in the first geostationary orbit (GEO).

When in the mission orbit, the following tasks are fulfilled in *normal mode:*

(1) pitch control by momentum wheel in torque mode control;
(2) roll/yaw control by the WHECON principle (to be explained in Chapter 8) – roll control by horizon sensor and yaw control by momentum bias (see Dougherty, Scott, and Rodden 1968); and
(3) momentum management of the wheel, which keeps the momentum of the wheel inside permitted bounds.

In addition, "station keeping" maintains the s/c within prescribed limits (of the order of $\pm 0.05°$) about the nominal longitude station position, and also within the same permitted deviation in the inclination of the mission orbit. Station keeping involves both north–south and east–west correction maneuvers.

1.3 Outline of the Book

The chapters of this book have been arranged to give the reader an integrated view of the subject of attitude and orbit control. Chapters 2 and 3 deal with the satellite *orbit* dynamics and control. The remaining chapters treat the *attitude* dynamics and control of satellites.

Chapter 2 develops the classical equations of motion of ideal Keplerian orbits. It then presents Gauss's and Lagrange's planetary equations, with which the perturbed orbit motion of a satellite can be analyzed. Chapter 3 covers basic orbital control concepts including control and station keeping of geostationary satellites.

Chapter 4 is devoted to the basic equations of rotational motion about some axis through its center of mass. The usual notions of angular momentum and rotational kinetic energy are introduced, defining the rotational state of a body. Next, Euler's moment equations are stated as a preliminary to analyzing the angular stability of a rotating body with or without the existence of internal energy dissipation. This chapter also develops the linearized angular equations of motion of a nonspinning spacecraft, which are necessary when designing a feedback attitude control system.

Chapter 5 deals with gravity gradient stabilization of a s/c. Gravity gradient control is a passive means of stabilizing the attitude of the satellite. In principle, gravity gradient attitude control is undamped. This chapter analyzes passive and active damping, and emphasizes the inaccuracies in attitude stabilization that arise in response to environmental conditions.

Chapter 6 deals with single- and dual-spin stabilization. The single-spin stabilization mode is frequently used to keep the direction of the thrust vector constant in space during the orbit change process. This chapter discusses the minimum spin rate needed to keep the thrust direction within permitted bounds, despite parasitic disturbing torques acting on the s/c. Also analyzed are active nutation control and despinning of the satellite at the end of the orbit change process together with the denutation stage. The mass of fuel consumed is evaluated analytically for both active nutation control and despin–denutation control. A design example is included.

The single-spin property is also used in the context of attitude stabilizing the spin axis of the s/c perpendicular to the orbit plane, thus allowing an attached communications payload to scan the earth continuously and so provide the communications link. Due to parasitic disturbing torques acting on the s/c, nutational motion is excited and must be constantly damped. Depending on the specific moments of inertia of the s/c, either passive or active damping is added to the attitude control of the satellite; various damping schemes are analyzed.

Dual-spin stabilization was developed in order to increase the efficiency of communication of spinning satellites by enabling the communications antenna to be continuously directed toward the earth. In this control concept, passive nutation damping can be achieved by energy dissipation. The stabilizing conditions are explicitly stated.

Chapter 7 is concerned with attitude stabilization and maneuvering of spacecraft stabilized in three axes. In these satellites, there is no constant angular momentum added to the s/c to keep the direction of one of its axes stabilized in space, so attitude control is achieved by simultaneously controlling the three body axes. For small attitude–angle maneuvering, the common Euler angles are a clear way to express the attitude of the satellite with reference to some defined frame in space. However, for larger attitude changes, the attitude kinematics are expressed much more effectively with the direction cosine matrix and the quaternion vector. The chapter begins with a thorough discussion of control laws for attitude control.

Momentum exchange devices are used to provide the control torques for accurate attitude control. These devices, called *reaction* and *momentum wheels,* are introduced and modeled for use in this and subsequent chapters. If an external inertial disturbance acts on an attitude-controlled satellite, then excess angular momentum is accumulated in the controlling wheels. A control scheme using *magnetic torqrods* to dump this momentum from the wheels is analyzed and simulated.

Attitude sensors and controllers are inherently noisy. When designing a control loop, these noises must be taken into consideration using statistical linear control theory. Tradeoffs in the design process due to such noises are stated. In order to augment the reliability and the control capability of a complete three-axis ACS, more than three reaction wheels are sometimes used. Chapter 7 analyzes optimal distribution of the computed control torques among the different wheels. Time-optimal attitude maneuvers about a single body axis are also analyzed. The last section of Chapter 7 deals with specifying technical characteristics of the reaction wheel based on mission requirements of the attitude control system (ACS).

Chapter 8 is concerned with momentum-biased satellites. A momentum wheel added to the satellite provides inertial stabilization to the three-axis stabilized s/c about one of its axes. The inertial stabilizing torque is achieved with the momentum produced in the momentum wheel. Unfortunately, environmental disturbances tend to destabilize the s/c by increasing the nutational motion, which thus must be actively controlled. There are three essential schemes for controlling nutational motion: magnetic damping, reaction propulsion damping, and – for high-altitude–orbit satellites such as geostationary satellites – solar torque control. These schemes are analyzed and compared.

Chapter 9 reviews the use of propulsion reaction hardware for attitude control. Only reaction thrusters can provide the high torques necessary in different attitude control tasks during orbit changes. The attitude stabilization scheme using reaction thrusters is stated and analyzed. Attitude maneuvering can likewise use reaction thrust torques. The achievable accuracies depend largely on the minimal impulse bit that a thruster can deliver. Also, since the torques delivered are with constant amplitude, the reaction pulses must be width- or frequency-modulated. Both modulation schemes are analyzed, and design examples are given.

Chapter 10 introduces the dynamics of structural modes and fuel sloshing dynamics. The chapter provides simplified analyses of solar panels and fuel sloshing, as well as rules-of-thumb for obtaining the simplified models so necessary in the initial design stage of an ACS. Also, given these initial models, the reader is shown how to approximate the maximum obtainable bandwidths of the system.

Appendix A is a short introduction to attitude transformations in space. It deals with Euler angle transformations, the direction cosine matrix, the quaternion vector, the relations among them, and attitude kinematics in general. Appendix B is a concise introduction to attitude measurement hardware. It is of the utmost importance to have a clear knowledge of such sensor characteristics, as their noise behavior influences achievable accuracies. The hardware treated includes horizon sensors (static or scanning), analog and digital sun sensors, star sensors, and angular rate sensors; characteristics data sheets are shown for various existing products. Appendix C describes a variety of control hardware, such as propulsion systems, magnetic torqrods, reaction wheels, and solar panels and flaps for achieving solar control torques.

1.4 Notation and Abbreviations

Vectors will be expressed by bold letters: $\mathbf{V}$, $\boldsymbol{\gamma}$. Matrices will be denoted by square brackets, and the name of the matrix inside the brackets in capital bold letters: $[\mathbf{A}]$. Scalar variables are expressed using italicized letters: V, γ. The scalar "dot" product of two vectors will be expressed by a solid dot: $\mathbf{a} \bullet \mathbf{b}$. The vector "cross" product will be denoted by a boldface cross: $\mathbf{a} \times \mathbf{b}$. Multiple products will likewise be denoted by solid dots and boldface crosses: $\mathbf{a} \bullet (\mathbf{b} \times \mathbf{c})$; $\mathbf{a} \times (\mathbf{b} \times \mathbf{c})$. The MKS system of units is used throughout the book.

The following abbreviations will be used: ACS ≡ attitude control system; AOCS ≡ attitude and orbit control system; cm ≡ center of mass; ES ≡ earth sensor; LP ≡ low pass; MW ≡ momentum wheel; RW ≡ reaction wheel; s/c ≡ spacecraft; ss ≡ steady state.

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CHAPTER 2

Orbit Dynamics

2.1 Basic Physical Principles

Orbital mechanics, as applied to artificial spacecraft, is based on celestial mechanics. In studying the motion of satellites, quite elementary principles are necessary. In fact, Kepler provided three basic empirical laws that describe motion in unperturbed planetary orbits. Newton formulated the more general physical laws governing the motion of a planet, laws that were consistent with Kepler's observations.

In this chapter, the dynamical equations of motion for ideal, unperturbed Keplerian orbits – and subsequently for realistic, perturbed orbits – will be analyzed. Kepler's laws of motion describe ideal orbits that do not exist in nature. Perturbing forces and physical anomalies cause spacecraft orbits to have strange properties; in most cases these cause difficulties for the space control engineer, but in other cases these properties may be of enormous help.

Keplerian orbits are treated in Sections 2.1–2.6. For further reading on this subject, Kaplan (1976) or Thomson (1986) may be consulted. Perturbed non-Keplerian orbits are treated in Sections 2.7–2.9 (see also Deutsch 1963, Alby 1983, and Battin 1990).

2.1.1 *The Laws of Kepler and Newton*

Kepler provided three empirical laws for planetary motion, based on Brahe's planetary observations. First, the orbit of each planet is an ellipse with the sun located at one focus. Second, the radius vector drawn from the sun to any planet sweeps out equal areas in equal time intervals (the law of areas). Third, planetary periods of revolution are proportional to the [mean distance to sun]$^{3/2}$.

Newton provided three laws of mechanics and one for gravitational attraction. Most analysis of celestial and spacecraft orbit dynamics is based on Newton's laws, formulated as follows.

(1) Every particle remains in a state of rest, or of uniform motion in a straight line with constant velocity, unless acted upon by an external force.

(2) The rate of change of linear momentum of a body is equal to the force $\mathbf{F}$ applied on the body, where $\mathbf{p} = m\mathbf{v}$ is the linear momentum and

$$\mathbf{F} = \frac{d\mathbf{p}}{dt} = \frac{d(m\mathbf{v})}{dt}. \tag{2.1.1}$$

In this equation, m is the mass of the body and $\mathbf{v}$ is the velocity vector. For a constant mass, this law takes the simplified form

$$\mathbf{F} = m\mathbf{a}, \tag{2.1.2}$$

where $\mathbf{a} = d\mathbf{v}/dt$ is the familiar linear acceleration.

(3) For any force $\mathbf{F}_{12}$ exerted by particle 1 on a particle 2, there must likewise exist a force $\mathbf{F}_{21}$ exerted by particle 2 on particle 1, equal in magnitude and opposite in direction:

$$\mathbf{F}_{12} = -\mathbf{F}_{21}. \tag{2.1.3}$$

(4) Any two particles attract each other with a force given by the expression

$$\mathbf{F} = \frac{Gm_1 m_2 \mathbf{r}}{r^3}, \tag{2.1.4}$$

where $\mathbf{r}$ is a vector of magnitude r along the line connecting the two particles with masses m_1 and m_2, and $G = 6.669 \times 10^{-11}$ m^3/kg-s^2 is the universal constant of gravitation. This is the famous *inverse square law of force;* the magnitude of the force is $F = Gm_1 m_2/r^2$.

2.1.2 *Work and Energy*

If a force $\mathbf{F}$ acting on a body causes its displacement by a distance $d\mathbf{r}$, then the incremental work done by the force on the body is defined as

$$dW = \mathbf{F} \cdot d\mathbf{r}, \tag{2.1.5}$$

where $\mathbf{F} \cdot d\mathbf{r}$ is a scalar "dot" product. This illustrates that only the component of $\mathbf{F}$ in the direction of $d\mathbf{r}$ is effective in doing the work. The total work done by the force on the body is equal to the line integral

$$W_{12} = \int_c \mathbf{F} \cdot d\mathbf{r} = \int_{\mathbf{r}_1}^{\mathbf{r}_2} \mathbf{F} \cdot d\mathbf{r} \tag{2.1.6}$$

(see Figure 2.1.1).

The work done on a body changes its kinetic and potential energies. With respect to kinetic energy, the total work done on a body by moving it along the line c from P_1 to P_2 in Figure 2.1.1 is given by

$$\begin{aligned} W_{12} &= \int_c \mathbf{F} \cdot d\mathbf{r} = \int_c m \frac{d\mathbf{v}}{dt} \cdot d\mathbf{r} = \int_{\mathbf{r}_1}^{\mathbf{r}_2} m \, d\mathbf{v} \cdot \mathbf{v} = \int_{\mathbf{r}_1}^{\mathbf{r}_2} \frac{m}{2} d(\mathbf{v}^2) \\ &= \frac{m}{2}(\mathbf{v}_2^2 - \mathbf{v}_1^2) = T_2 - T_1, \end{aligned} \tag{2.1.7}$$

which is the difference in kinetic energies at $\mathbf{r}_2$ and at $\mathbf{r}_1$; $T = (mv^2)/2$, and

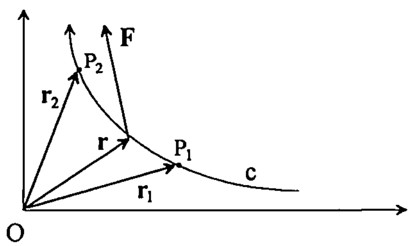

Figure 2.1.1 Line integral of force and work.

$$dW = dT. \tag{2.1.8}$$

With respect to potential energy, in conservative force fields there exists a scalar function U such that $\mathbf{F} = -\text{grad}\, U(\mathbf{r})$. In such fields, if the work is done from P_1 to P_2 then

$$\begin{aligned} W_{12} &= \int_{\mathbf{r}_1}^{\mathbf{r}_2} \mathbf{F} \cdot d\mathbf{r} = \int_{\mathbf{r}_1}^{\mathbf{r}_0} \mathbf{F} \cdot d\mathbf{r} + \int_{\mathbf{r}_0}^{\mathbf{r}_2} \mathbf{F} \cdot d\mathbf{r} \\ &= \int_{\mathbf{r}_1}^{\mathbf{r}_0} \mathbf{F} \cdot d\mathbf{r} - \int_{\mathbf{r}_2}^{\mathbf{r}_0} \mathbf{F} \cdot d\mathbf{r} = U(\mathbf{r}_1) - U(\mathbf{r}_2), \end{aligned} \tag{2.1.9}$$

where the scalar $U(\mathbf{r})$ is defined as the *potential energy* at $\mathbf{r}$. Hence

$$dW = -dU. \tag{2.1.10}$$

As is well known, the work done in a conservative force field is independent of the path taken by the force, and is a function only of the final position.

From Eq. 2.1.8 and Eq. 2.1.10 follows the law of conservation of energy:

$$dT + dU = 0 \quad \text{and} \quad T + U = \text{const} = E; \tag{2.1.11}$$

E is called the *total energy*. For a conservative force field, the total energy is constant. This is the principle of *conservation of energy.*

2.2 The Two-Body Problem

The two-body problem is an idealized situation in which only two bodies exist that are in relative motion in a force field described by the inverse square law (Eq. 2.1.4). In order to obtain simple analytical results for the motion of celestial bodies or spacecraft, it is assumed that additional bodies are situated far enough from the two-body system, thus no appreciable force is exerted on them from a third body.

In Figure 2.2.1, m_2 exerts an attraction force $\mathbf{F}_1 = m_1 \ddot{\mathbf{r}}_1$ on m_1, and m_1 exerts a force $\mathbf{F}_2 = m_2 \ddot{\mathbf{r}}_2$ on m_2:

$$\mathbf{F}_1 = m_1 \ddot{\mathbf{r}}_1 = G m_1 m_2 \frac{\mathbf{r}_2 - \mathbf{r}_1}{|\mathbf{r}_2 - \mathbf{r}_1|^3}; \tag{2.2.1}$$

$$\mathbf{F}_2 = m_2 \ddot{\mathbf{r}}_2 = G m_1 m_2 \frac{\mathbf{r}_1 - \mathbf{r}_2}{|\mathbf{r}_1 - \mathbf{r}_2|^3} = -\mathbf{F}_1. \tag{2.2.2}$$

From Eq. 2.2.1 and Eq. 2.2.2 we find that

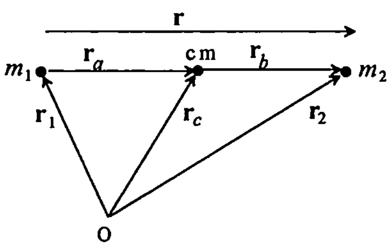

Figure 2.2.1 Displacement vectors in a two-body system.