11.1 Attitude Determination and Control

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The *attitude determination and control subsystem (ADCS)* stabilizes the vehicle and orients it in desired directions during the mission despite the external disturbance torques acting on it. This requires that the vehicle determine its attitude, using sensors, and control it, using actuators. The ADCS often is tightly coupled to other subsystems on board, especially the propulsion (Chap. 17) and navigation (Sec. 11.7) functions. Additional information on attitude determination and control can be found in Wertz [1978, 2001], Kaplan [1976], Agrawal [1986], Hughes [1986], Griffin and French [1990], Chobotov [1991], and Fortescue and Stark [1992].

We begin by discussing several useful concepts and definitions, including mass properties, disturbance torques, angular momentum, and reference vectors. The mass properties of a spacecraft are key in determining the size of control and disturbance torques. We typically need to know the location of the center of mass or gravity (cg) as well as the elements of the inertia matrix: the moments and products of inertia about chosen reference axes. (See Sec. 11.6 for examples of moment of inertia calculations.) The direction of the principal axes—those axes for which the inertia matrix is diagonal and the products of inertia are zero—are also of interest. Finally, we need to know how these properties change with time, as fuel or other consumables are used, or as appendages are moved or deployed.

A body in space is subject to small but persistent disturbance torques (e.g., 10^{-4} N·m) from a variety of sources. These torques are categorized as *cyclic*, varying in a sinusoidal manner during an orbit, or *secular*, accumulating with time, and not averaging out over an orbit. These torques would quickly reorient the vehicle unless resisted in some way. An ADCS system resists these torques either passively, by exploiting inherent inertia or magnetic properties to make the "disturbances" stabilizing and their effects tolerable, or actively, by sensing the resulting motion and applying corrective torques.

Angular momentum plays an important role in space, where torques typically are small and spacecraft are unconstrained. For a body initially at rest, an external torque will cause the body to angularly accelerate proportionally to the torque—resulting in an increasing angular velocity. Conversely, if the body is initially spinning about an axis perpendicular to the applied torque, then the body spin axis will precess, moving with a constant angular velocity proportional to the torque. Thus, spinning bodies act like gyroscopes, inherently resisting disturbance torques in 2 axes by responding with constant, rather than increasing, angular velocity. This property of spinning bodies, called *gyroscopic stiffness*, can be used to reduce the effect of small, cyclic disturbance torques. This is true whether the entire body spins or just a portion of it, such as a momentum wheel or spinning rotor.

Conservation of vehicle angular momentum requires that only external torques change the system net angular momentum. Thus, external disturbances must be resisted by external control torques (e.g., thrusters or magnetic torquers) or the resulting momentum buildup must be stored internally (e.g., by reaction wheels) without reorienting the vehicle beyond its allowable limits. The momentum buildup due to secular disturbances ultimately must be reduced by applying compensating external control torques.

Often, in addition to rejecting disturbances, the ADCS must reorient the vehicle (in *slew* maneuvers) to repoint the payload, solar arrays, or antennas. These periodic repointing requirements may drive the design to larger actuators than would be required for disturbance rejection alone.

To orient the vehicle correctly, external references must be used to determine the vehicle's absolute attitude. These references include the Sun, the Earth's IR horizon, the local magnetic field direction, and the stars. In addition, inertial sensors (gyroscopes) also can be carried to provide a short-term attitude reference between external updates. External references (e.g., Sun angles) are usually measured as body-centered angular distances to a vector. Each such vector measurement provides only two of the three independent parameters needed to specify the orientation of the spacecraft. This results in the need for multiple sensor types on board most spacecraft.

Table 11-1 lists the steps for designing an ADCS for spacecraft. The FireSat spacecraft, shown in Fig. 11-1, will be used to illustrate this process. The process must be iterative, with mission requirements and vehicle mass properties closely related to the ADCS approach. Also, a rough estimate of disturbance torques (see Chap. 10) is necessary before the type of control is selected (step 2), even though the type of control will help determine the real disturbance environment (step 3).

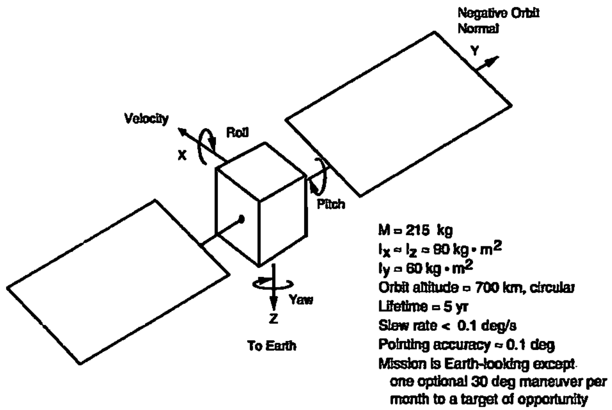

Fig. 11-1. Hypothetical FireSat Spacecraft. We use this simplified example to discuss key concepts throughout this section. See Fig. 5-1 for illustration of roll-pitch-yaw coordinates.

11.1.1 Control Modes and Requirements

Tables 11-2 and 11-3 describe typical spacecraft control modes and requirements. The ADCS requirements are closely tied to mission needs and other subsystem characteristics, as shown in Fig. 11-2. These requirements may vary considerably with mission phase or modes, challenging the designer to develop a single hardware suite for different objectives.

For many spacecraft, the ADCS must control vehicle attitude during firing of large liquid or solid rocket motors, which may be used during orbit insertion or for orbit changes. Large motors create large disturbance torques, which can drive the design to larger actuators than are needed once on station.

TABLE 11-1. Control System Design Process. An iterative process is used for designing the ADCS as part of the overall spacecraft system.

Step	Inputs	Outputs	FireSat Example
1a. Define control modes 1b. Define or derive system-level requirements by control mode	Mission requirements, mission profile, type of insertion for launch vehicle	List of different control modes during mission (See Table 11-2) Requirements and constraints (See Table 11-3)	Orbit injection: none—provided by launch vehicle Normal: nadir pointing, < 0.1 deg; autonomous determination (Earth-relative) Optional slew: One 30 deg maneuver per month to a target of opportunity
2. Select type of spacecraft control by attitude control mode (Sec. 11.1.2)	Payload, thermal and power needs Orbit, pointing direction Disturbance environment	Method for stabilizing and control: 3-axis, spinning, or gravity gradient	Momentum bias stabilization with a pitch wheel, electromagnets for momentum dumping, and optionally, thrusters for slewing (shared with ΔV system in navigation)
3. Quantify disturbance environment (Sec. 11.1.3)	Spacecraft geometry, orbit, solar/magnetic models, mission profile	Values for forces from gravity gradient, magnetic aerodynamics, solar pressure, internal disturbances, and powered flight effects on control (cg offsets, slosh)	Gravity gradient: 1.8×10^{-6} N·m normal pointing; 4.4×10^{-5} N·m during target-of-opportunity mode Magnetic: 4.5×10^{-5} N·m Solar: 6.6×10^{-6} N·m Aerodynamic: 3.4×10^{-6} N·m
4. Select and size ADCS hardware (Sec. 11.1.4)	Spacecraft geometry, pointing accuracy, orbit conditions, mission requirements, lifetime, orbit, pointing direction, slew rates	Sensor suite: Earth, Sun, inertial, or other sensing devices Control actuators, e.g., reaction wheels, thrusters, or magnetic torquers Data processing electronics, if any, or processing requirements for other subsystems or ground computer	1 Momentum wheel, Momentum: 40 N·m·s 2 Horizon sensors, Scanning, 0.1 deg accuracy 3 Electromagnets, Dipole moment: 10 A·m² 4 Sun sensors, 0.1 deg accuracy 1 3-axis magnetometer, 1 deg accuracy
5. Define determination and control algorithms	All of above	Algorithms, parameters, and logic for each determination and control mode	Determination: Horizon data filtered for pitch and roll. Magnetometer and Sun sensors used for yaw. Control: Proportional-plus-derivative for pitch, Coupled roll-yaw control with electromagnets
6. Iterate and document	All of above	Refined requirements and design Subsystem specification	

Once the spacecraft is on station, the payload pointing requirements usually dominate. These may require Earth-relative or inertial attitudes, and fixed or spinning fields of view. In addition, we must define the need for and frequency of attitude slew maneuvers. Such maneuvers may be necessary to:

TABLE 11-2. Typical Attitude Control Modes. Performance requirements are frequently tailored to these different control operating modes.

Mode	Description
Orbit Insertion	Period during and after boost while spacecraft is brought to final orbit. Options include no spacecraft control, simple spin stabilization of solid rocket motor, and full spacecraft control using liquid propulsion system.
Acquisition	Initial determination of attitude and stabilization of vehicle. Also may be used to recover from power upsets or emergencies.
Normal, On-Station	Used for the vast majority of the mission. Requirements for this mode should drive system design.
Slew	Reorienting the vehicle when required.
Contingency or Safe	Used in emergencies if regular mode fails or is disabled. May use less power or sacrifice normal operation to meet power or thermal constraints.
Special	Requirements may be different for special targets or time periods, such as eclipses.

TABLE 11-3. Typical Attitude Determination and Control Performance Requirements. Requirements need to be specified for each mode. The following lists the areas of performance frequently specified.

Area	Definition*	Examples/Comments
	DETERMINATION	
Accuracy	How well a vehicle's orientation with respect to an absolute reference is known	0.25 deg, 3 σ, all axes; may be real-time or post-processed on the ground
Range	Range of angular motion over which accuracy must be met	Any attitude within 30 deg of nadir
	CONTROL	
Accuracy	How well the vehicle attitude can be controlled with respect to a commanded direction	0.25 deg, 3 σ; includes determination and control errors, may be taken with respect to an inertial or Earth-fixed reference
Range	Range of angular motion over which control performance must be met	All attitudes, within 50 deg of nadir, within 20 deg of Sun
Jitter	A specified angle bound or angular rate limit on short-term, high-frequency motion	0.1 deg over 1 min, 1 deg/s,1 to 20 Hz; usually specified to keep spacecraft motion from blurring sensor data
Drift	A limit on slow, low-frequency vehicle motion. Usually expressed as angle/time.	1 deg/hr, 5 deg max. Used when vehicle may drift off target with infrequent resets (especially if actual direction is known)
Settling Time	Specifies allowed time to recover from maneuvers or upsets.	2 deg max motion, decaying to < 0.1 deg in 1 min; may be used to limit overshoot, ringing, or nutation

* Definitions vary with procuring and designing agencies, especially in details (e.g., 1 or 3 σ, amount of averaging or filtering allowed). It is always best to define exactly what is required.

- Repoint the payload's sensing systems to targets of opportunity
- Maneuver the attitude control system's sensors to celestial targets for attitude determination
- Track stationary or moving targets
- Acquire the desired satellite attitude initially or after a failure

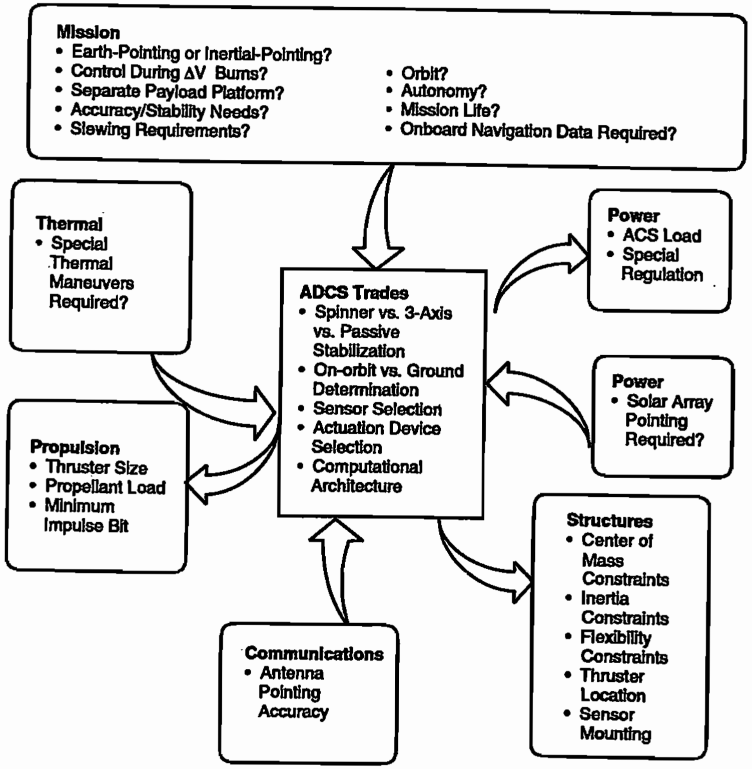

Fig. 11-2. The Impact of Mission Requirements and Other Subsystems on the ADCS Subsystem. Direction of arrows shows requirements flow from one subsystem to another.

In most cases, we do not need to rotate the spacecraft quickly. But retargeting time may be critical for some applications. In either case, slewing mainly influences the choice and size of actuators. For example, the vehicle's maximum slew rate determines the thrusters' size or the reaction wheel's maximum torque. High-rate maneuvers may require other actuation systems, such as a second set of high-thrust reaction jets or perhaps control moment gyros.

For FireSat, we assume that the launch vehicle places us in our final orbit, with no need for ADCS control during orbit insertion. The normal pointing requirement is 0.1 deg, nadir-oriented. Attitude determination must be autonomous, providing Earth-relative knowledge better than 0.1 deg (to support the pointing requirement) while the vehicle is within 30 deg of nadir. In addition to these basic requirements, we will consider an optional requirement for occasional repointing of the spacecraft to a region of interest. We want to examine how such a requirement would influence the design, increasing the complexity and capability of the ADCS. For this option, we will assume the requirement to repoint the vehicle once every 30 days. It must repoint, or slew, up to 30 deg in under 10 min, and hold the relative nadir orientation for 90 min.

11.1.2 Selection of Spacecraft Control Type

Once we have defined the subsystem requirements, we are ready to select a method of controlling the spacecraft. Table 11-4 lists several different methods of control, along with typical characteristics of each.

TABLE 11-4. Attitude Control Methods and Their Capabilities. As requirements become tighter, more complex control systems become necessary.

Type	Pointing Options	Attitude Maneuverability	Typical Accuracy	Lifetime Limits
Gravity-gradient	Earth local vertical only	Very limited	±5 deg (2 axes)	None
Gravity-gradient and Momentum Bias Wheel	Earth local vertical only	Very limited	±5 deg (3 axes)	Life of wheel bearings
Passive Magnetic	North/south only	Very limited	±5 deg (2 axes)	None
Pure Spin Stabilization	Inertially fixed any direction Repoint with precession maneuvers	High propellant usage to move stiff momentum vector	±0.1 deg to ±1 deg in 2 axes (proportional to spin rate)	Thruster propellant (if applies)*
Dual-Spin Stabilization	Limited only by articulation on despun platform	Momentum vector same as above Despun platform constrained by its own geometry	Same as above for spin section Despun dictated by payload reference and pointing	Thruster propellant (if applies)* Despin bearings
Bias Momentum (1 wheel)	Best suited for local vertical pointing	Momentum vector of the bias wheel prefers to stay normal to orbit plane, constraining yaw maneuver	±0.1 deg to ±1 deg	Propellant (if applies)* Life of sensor and wheel bearings
Zero Momentum (thruster only)	No constraints	No constraints High rates possible	±0.1 deg to ±5 deg	Propellant
Zero Momentum (3 wheels)	No constraints	No constraints	±0.001 deg to ±1 deg	Propellant (if applies)* Life of sensor and wheel bearings
Zero Momentum CMG	No constraints	No constraints High rates possible	±0.001 deg to ±1 deg	Propellant (if applies)* Life of sensor and wheel bearings

*Thrusters may be used for slewing and momentum dumping at all altitudes. Magnetic torquers may be used from LEO to GEO.

Passive Control Techniques. *Gravity-gradient control* uses the inertial properties of a vehicle to keep it pointed toward the Earth. This relies on the fact that an elongated object in a gravity field tends to align its longitudinal axis through the Earth's center. The torques which cause this alignment decrease with the cube of the orbit radius, and are symmetric around the nadir vector, thus not influencing the yaw of a spacecraft around the nadir vector. This tendency is used on simple spacecraft in near-Earth orbits without yaw orientation requirements, often with deployed booms to achieve the desired inertias.

Frequently, we add dampers to gravity-gradient spacecraft to reduce *libration*—small oscillations around the nadir vector caused by disturbances. Gravity-gradient spacecraft are particularly sensitive to thermal shocks on long deployed booms when entering or leaving eclipses. They also need a method of ensuring attitude capture with the correct end of the spacecraft pointed at nadir—the gravity-gradient torques make either end along the minimum inertia axis equally stable.

In the simplest gravity-gradient spacecraft, only two orientation axes are controlled. The orientation around the nadir vector is unconstrained. To control this third axis, a small, constant-speed momentum wheel is sometimes added along the intended pitch axis (i.e., an axis perpendicular to the nadir and velocity vectors). This "yaw" wheel is stable when it aligns with the orbit normal, and small energy dissipation mechanisms on board cause the spacecraft to seek this minimum energy, stable orientation without active control.

A third type of purely passive control uses permanent magnets on board the spacecraft to force alignment along the Earth's magnetic field. This is most effective in near-equatorial orbits where the field orientation stays almost constant for an Earth-pointing vehicle.

Spin Control Techniques. *Spin stabilization* is a passive control technique in which the entire spacecraft rotates so that its angular momentum vector remains approximately fixed in inertial space. Spin-stabilized spacecraft (or *spinners*), employ the gyroscopic stability discussed earlier to passively resist disturbance torques about two axes. The spinning motion is stable (in its minimum energy state) if the vehicle is spinning about the axis having the largest moment of inertia. Energy dissipation mechanisms on board, such as fuel slosh and structural damping) will cause any vehicle to head toward this state if uncontrolled. Thus disk-shaped spinners are passively stable while pencil-shaped vehicles are not. Spinners can be simple, survive for long periods without attention, provide a thermally benign environment for components, and provide a scanning motion for sensors. The principal disadvantages of spin stabilization are (1) that the vehicle mass properties must be controlled to ensure the desired spin direction and stability and (2) that the angular momentum vector requires more fuel to reorient than a vehicle with no net angular momentum, reducing the usefulness of this technique for payloads that must be repointed frequently.

It takes extra fuel to reorient a spinner because of the gyroscopic stiffness which also helps it resist disturbances. In reorienting a spinning body with angular momentum, h, a constant torque, T, will produce an angular velocity, ω, perpendicular to the applied torque and angular momentum vector, of magnitude $\omega = T/h$. Thus, the higher the stored momentum is, the more torque must be applied for a given ω. For a maneuver through an angle θ, the torque-time product—an indication of fuel required for the maneuver—is a constant equal to $h\theta$. Conversely, for a nonspinning vehicle with no initial angular velocity, a small torque can be used to start it rotating, with an opposite torque to stop it. The fuel used for any angle maneuver can be infinitesimally small if a slow maneuver is acceptable.

A useful variation of spin control is called *dual-spin stabilization*, where the spacecraft has two sections spinning at different rates about the same axis. Normally, one section, the rotor, spins rapidly to provide angular momentum, while the second section, the stator or platform, is despun to keep one axis pointed toward the Earth or Sun. By combining inertially fixed and rotating sections on the same vehicle, dual spinners can accommodate a variety of payloads in a simple vehicle. Also, by adding energy dissipation devices to the platform, a dual spinner can be passively stable spinning about the axis with the smallest moment of inertia. This permits more pencil-shaped spacecraft, which fit better in launch vehicle fairings. The disadvantage of dual-spin stabilization is the added complexity of the platform bearing and slip rings between the sections. This complexity can increase cost and reduce reliability compared to simple spin stabilization.

Spinning spacecraft, both simple and dual, exhibit several distinct types of motion which often are confused. *Precession* is the motion of the angular momentum vector caused by external torques such as thruster firings. *Wobble* is the apparent motion of the body when it is spinning with the angular momentum vector aligned along a principal axis of inertia which is offset from a body reference axis—for example, the intended spin axis. This looks like motion of the intended spin axis around the angular momentum vector at the spin rate.

Nutation is the torque-free motion of the spacecraft body when the angular momentum vector is not perfectly aligned along a principal axis of inertia. For rod-shaped objects, this motion is a slow rotation (compared to spin rate) of the spin axis around the angular momentum vector. For these objects, spinning about a minimum inertia axis, additional energy dissipation will cause increased nutation. For disk-shaped objects, spinning around a maximum inertia axis, nutation appears as a higher-than-spin-rate tumbling. Energy dissipation for these objects (e.g., with a passive nutation damper) reduces nutation, resulting in a clean spin.

Nutation is caused by disturbances such as thruster impulses, and can be seen as varying signals in body-mounted inertial and external sensors. Wobble is caused by imbalance and appears as constant offsets in body-mounted sensors. Such constant offsets are rarely discernible unless multiple sensors are available.

Spin stability normally requires active control, such as mass expulsion or magnetic coils, to periodically adjust the spacecraft's attitude and spin rate to counteract disturbance torques. In addition, we may need to damp the nutation caused by disturbances, precession commands, or fuel slosh. Aggravating this nutation is the effect of structural flexure and fuel slosh, which is present in any space vehicle to one degree or another. Once the excitation stops, nutation decreases as these same factors dissipate the energy. But this natural damping can take hours. We can neutralize this source of error in minutes with nutation dampers (see Sec. 11.1.2). We can also reduce the amount of nutation from these sources by increasing the spin rate, thus increasing the stiffness of the spinning vehicle. If the spin rate is 20 rpm, and the nutation angle is 3 deg, then at 60 rpm the nutation angle would decrease by a factor of three. We seldom use spin rates above 90 rpm because of the large centrifugal forces and their effect on structural design and weight. In thrusting and pointing applications, spin rates under 20 rpm may allow excessive nutation and are not used. However, noncritical applications, such as thermal control, are frequently insensitive to nutation and may employ very low spin rates.

Three-axis Control Techniques. Spacecraft stabilized in 3 axes are more common today than those using spin or gravity gradient. They maneuver and can be stable and accurate, depending on their sensors and actuators. But they are also more expensive and more complex. The control torques about the axes of 3-axis systems come from combinations of momentum wheels, reaction wheels, control moment gyros, thrusters, or magnetic torquers. Broadly, however, these systems take two forms: one uses momentum bias by placing a momentum wheel along the pitch axis; the other is called zero momentum with a reaction wheel on each axis. Either option usually needs thrusters or magnetic torquers as well as the wheels.

In a *zero-momentum* system, reaction wheels respond to disturbances on the vehicle. For example, a vehicle-pointing error creates a signal which speeds up the wheel, initially at zero. This torque corrects the vehicle and leaves the wheel spinning at low speed, until another pointing error speeds the wheel further or slows it down again. If the disturbance is cyclic during each orbit, the wheel may not approach saturation speed for several orbits. Secular disturbances, however, cause the wheel to drift toward saturation. We then must apply an external torque, usually with a thruster or magnetic torquer, to force the wheel speed back to zero. This process, called *desaturation, momentum unloading*, or *momentum dumping*, can be done automatically or by command from the ground.

When high torque is required for large vehicles or fast slews, a variation of 3-axis control is possible using *control moment gyros,* or *CMGs*. These devices work like momentum wheels on gimbals. (See Sec. 11.1.4 for a further discussion of CMGs.) The control of CMGs is complex, but their available torque for a given weight and power can make them attractive.

As a final type of zero momentum 3-axis control, simple all-thruster systems are used for short durations when high torque is needed, such as orbit insertion or during ΔV burns from large motors. These thrusters then may be used for different purposes such as momentum dumping during other mission modes.

Momentum bias systems often have just one wheel with its spin axis mounted along the pitch axis, normal to the orbit plane. The wheel is run at a nearly constant, high speed to provide gyroscopic stiffness to the vehicle, just as in spin stabilization, with similar nutation dynamics. Around the pitch axis, however, the spacecraft can control attitude by torquing the wheel, slightly increasing or decreasing its speed. Periodically, the pitch wheel must be desaturated (brought back to its nominal speed), as in zero-momentum systems, using thrusters or magnets.

The dynamics of nadir-oriented momentum-bias vehicles exhibit a phenomenon known as *roll-yaw coupling*. To see this coupling, consider an inertially-fixed angular momentum vector at some angle with respect to the orbit plane. If the angle is initially a positive roll error, then 1/4 orbit later it appears purely about the yaw axis as a negative yaw error. As the vehicle continues around the orbit, the angle goes through negative roll and positive yaw before realigning as positive roll. This coupling, which is due to the apparent motion of the Earth and, therefore, the Earth-fixed coordinate frame as seen from the spacecraft, can be exploited to control roll and yaw over a quarter orbit using only a roll sensor.

Effects of Requirements on Control Type. With the above knowledge of control types, we can proceed to select a type which best meets mission requirements. Tables 11-5 through 11-7 describe the effects of orbit insertion, payload pointing, and payload slew requirements on the selection process.

A common control approach during orbit insertion is to use the short-term spin stability of the spacecraft-orbit-insertion motor combination. Once on station, the motor may be jettisoned, the spacecraft despun using jets or a yo-yo device, and a different control technique used.

Payload pointing will influence the ADCS control method, the class of sensors, and the number and kind of actuation devices. Occasionally, pointing accuracies are so stringent that a separate, articulated platform is necessary. An articulated platform can perform scanning operations much easier than the host vehicle, with better accuracy and stability.

TABLE 11-5. Orbit Transition Maneuvers and Their Effect. Using thrusters to change orbits creates special challenges for the ADCS.

Requirement	Effect on Spacecraft	Effect on ADCS
Large impulse to complete orbit insertion (thousands of m/s)	Solid motor or large bipropellant stage Large thrusters or a gimbaled engine or spin stabilization for attitude control during burns	Inertial measurement unit for accurate reference and velocity measurement Different actuators, sensors, and control laws for burn vs. coasting phases Need for navigation or guidance
On-orbit plane changes to meet payload needs or vehicle operations (hundreds of m/s)	More thrusters, but may be enough if coasting phase uses thrusters	Separate control law for thrusting Actuators sized for thrusting disturbances Onboard attitude reference for thrusting phase
Orbit maintenance trim maneuvers (<100 m/s)	One set of thrusters	Thrusting control law Onboard attitude reference

TABLE 11-6. Effect of Payload Pointing Directions on ADCS Design. The payload pointing requirements are usually the most important factors for determining the type of actuators and sensors.

Requirement	Effect on Spacecraft	Effect on ADCS
Earth-pointing • Nadir (Earth) pointing • Scanning • Off-nadir pointing	• Gravity-gradient fine for low accuracies (>1 deg) only • 3-axis stabilization acceptable with Earth local vertical reference	***If gravity-gradient*** • Booms, dampers, Sun sensors, magnetometer or horizon sensors for attitude determination • Momentum wheel for yaw control ***If 3-axis*** • Horizon sensor for local vertical reference (pitch and roll) • Sun or star sensor for third-axis reference and attitude determination • Reaction wheels, momentum wheels, or control moment gyros for accurate pointing and propellant conservation • Reaction control system for coarse control and momentum dumping • Magnetic torquers can also dump momentum • Inertial measurement unit for maneuvers and attitude determination
Inertial pointing • Sun • Celestial targets • Payload targets of opportunity	• Spin stabilization fine for medium accuracies with few attitude maneuvers • Gravity gradient does not apply • 3-axis control is most versatile for frequent reorientations	***If spin*** • Payload pointing and attitude sensor operations limited without despun platform • Needs thrusters to reorient momentum vector • Requires nutation damping ***If 3-axis*** • Typically, sensors include Sun sensors, star tracker, and inertial measurement unit • Reaction wheels and thrusters are typical actuators • May require articulated payload (e.g., scan platform)

TABLE 11-7. Slewing Requirements That Affect Control Actuator Selection. Spacecraft slew agility can demand larger actuators for intermittent use.

Slewing	Effect on Spacecraft	Effect on ADCS
None	Spacecraft constrained to one attitude—highly improbable	• Reaction wheels, if planned, can be smaller • If magnetic torque can dump momentum, may not need thrusters
Nominal rates—0.05 deg/s (maintain local vertical) to 0.5 deg/s	Minimal	• Thrusters very likely • Reaction wheels adequate by themselves only for a few special cases
High rates—> 0.5 deg/s	• Structural impact on appendages • Weight and cost increase	• Control moment gyros very likely or two thruster force levels—one for stationkeeping and one for high-rate maneuvers

Trade studies on pointing requirements must consider accuracy in determining attitude and controlling vehicle pointing. We must identify the most stringent requirements. Table 11-8 summarizes effects of accuracy requirements on the spacecraft's ADCS subsystem approach. Section 5.4 discusses how to develop pointing budgets.

FireSat Control Selection. For FireSat, we consider two options for orbit insertion control. First, the launch vehicle may directly inject the spacecraft into its mission orbit. This common option simplifies the spacecraft design, since no special insertion mode is needed. An alternate approach, useful for small spacecraft such as FireSat, is to use a monopropellant system on board the spacecraft to fly itself up from a low parking orbit to its final altitude. For small insertion motors, reaction wheel torque or momentum bias stabilization may be sufficient to control the vehicle during this burn. For larger motors, ΔV thruster modulation or dedicated ADCS thrusters become attractive.

Once on-station, the spacecraft must point its sensors at nadir most of the time and slightly off-nadir for brief periods. Since the payload needs to be despun and the spacecraft frequently reoriented, spin stabilization is not the best choice. Gravity-gradient and passive magnetic control cannot meet the 0.1 deg pointing requirement or the 30 deg slews. This leaves 3-axis control and momentum-bias stabilization as viable options for the on-station control as well.

Depending on other factors, either approach might work, and we will baseline momentum bias control with its simpler hardware requirements. In this case, we will use a single pitch wheel for momentum and electromagnets for momentum dumping and roll and yaw control.

For the optional off-nadir pointing requirement, 3-axis control with reaction wheels might be more appropriate. Also, 3-axis control often can be exploited to simplify the solar array design, by using one of the unconstrained payload axes (yaw, in this case) to replace a solar array drive axis. Thus, the reduced array size possible with 2 deg of freedom can be achieved with one array axis drive and one spacecraft rotation.

11.1.3 Quantify the Disturbance Environment

In this step, we determine the size of the external torques the ADCS must tolerate. Only three or four sources of torque matter for the typical Earth-orbiting spacecraft. They are gravity-gradient effects, magnetic-field torques on the vehicle, impingement

TABLE 11-8. Effect of Control Accuracy on Sensor Selection and ADCS Design. Accurate pointing requires better, higher cost, sensors, and actuators.

Required Accuracy (3σ)	Effect on Spacecraft	Effect on ADCS
> 5 deg	• Permits major cost savings • Permits gravity-gradient (GG) stabilization	***Without attitude determination*** • No sensors required for GG stabilization • Boom motor, GG damper, and a bias momentum wheel are only required actuators ***With attitude determination*** • Sun sensors & magnetometer adequate for attitude determination at ≥ 2 deg • Higher accuracies may require star trackers or horizon sensors
1 deg to 5 deg	• GG not feasible • Spin stabilization feasible if stiff, inertially fixed attitude is acceptable • Payload needs may require despun platform on spinner • 3-axis stabilization will work	• Sun sensors and horizon sensors may be adequate for sensors, especially a spinner • Accuracy for 3-axis stabilization can be met with RCS deadband control but reaction wheels will save propellant for long missions • Thrusters and damper adequate for spinner actuators • Magnetic torquers (and magnetometer) useful
0.1 deg to 1 deg	• 3-axis and momentum-bias stabilization feasible • Dual-spin stabilization also feasible	• Need for accurate attitude reference leads to star tracker or horizon sensors & possibly gyros • Reaction wheels typical with thrusters for momentum unloading and coarse control • Magnetic torquers feasible on light vehicles (magnetometer also required)
< 0.1 deg	• 3-axis stabilization is necessary • May require articulated & vibration-isolated payload platform with separate sensors	• Same as above for 0.1 deg to 1 deg but needs star sensor and better class of gyros • Control laws and computational needs are more complex • Flexible body performance very important

by solar-radiation, and, for low-altitude orbits, aerodynamic torques. Section 8.1 discusses the Earth environment in detail, and Chap. 10 and Singer [1964] provide a discussion of disturbances. Tables 11-9A and 11-9B summarize the four major disturbances, provide equations to estimate their size for the worst case, and calculate values for the FireSat example.

Disturbances can be affected by the spacecraft orientation, mass properties, and design symmetry. For the normal FireSat orientation, the largest torque is due to the residual magnetism in the spacecraft. If, however, we use the optional 30-deg off-nadir pointing, the gravity-gradient torque increases over an order of magnitude, to become as large as the magnetic torque. Note that we use 1 deg in the gravity-gradient calculations, rather than the 0.1 deg pointing accuracy. This is to account for our uncertain knowledge of the principal axes. If the principal axes are off by several degrees, that angle may dominate in the disturbance calculations. We also note that a less symmetric solar array arrangement would have increased both the aerodynamic and solar torques, making them closer to the magnetic torque in this example.

TABLE 11-9A. Simplified Equations for Estimating Worst-Case Disturbance Torques. Disturbance torques affect actuator size and momentum storage requirements.

Disturbance	Type	Influenced Primarily by	Formula
Gravity-gradient	Constant torque for Earth-oriented vehicle, cyclic for inertially oriented vehicle	• Spacecraft inertias • Orbit altitude	$T_g = \frac{3\mu}{2R^3}\lvert I_z - I_y \rvert \sin(2\theta)$ where T_g is the max gravity torque; μ is the Earth's gravity constant (3.986×10^{14} m³/s²); R is orbit radius (m), θ is the maximum deviation of the Z-axis from local vertical in radians, and I_z and I_y are moments of inertia about z and y (or x, if smaller) axes in kg·m².
Solar Radiation	Cyclic torque on Earth-oriented vehicle, constant for solar-oriented vehicle or platform	• Spacecraft geometry • Spacecraft surface reflectivity • Spacecraft geometry and cg location	Solar radiation pressure, T_{sp}, is highly dependent on the type of surface being illuminated. A surface is either transparent, absorbent, or a reflector, but most surfaces are a combination of the three. Reflectors are classed as diffuse or specular. In general, solar arrays are absorbers and the spacecraft body is a reflector. The worst case solar radiation torque is $T_{sp} = F(c_{ps} - cg)$ where $F = \frac{F_s}{c} A_s (1+q)\cos I$ and F_s is the solar constant, 1,367 W/m², c is the speed of light, 3×10^8 m/s, A_s is the surface area, c_{ps} is the location of the center of solar pressure, cg is the center of gravity, q is the reflectance factor (ranging from 0 to 1, we use 0.6), and I is the angle of incidence of the Sun.
Magnetic Field	Cyclic	• Orbit altitude • Residual spacecraft magnetic dipole • Orbit inclination	$T_m = DB$ where T_m is the magnetic torque on the spacecraft; D is the residual dipole of the vehicle in amp·turn·m² (A·m²), and B is the Earth's magnetic field in tesla. B can be approximated as $2M/R^3$ for a polar orbit to half that at the equator. M is the magnetic moment of the Earth, 7.96×10^{15} tesla·m³, and R is the radius from dipole (Earth) center to spacecraft in m.
Aerodynamic	Constant for Earth-oriented vehicles, variable for inertially oriented vehicle	• Orbit altitude • Spacecraft geometry and cg location	Atmospheric density for low orbits varies significantly with solar activity. $T_a = F(c_{pa} - cg) = FL$ where $F = 0.5\,[\rho C_d A V^2]$; F being the force; C_d the drag coefficient (usually between 2 and 2.5); ρ the atmospheric density; A, the surface area; V, the spacecraft velocity; c_{pa} the center of aerodynamic pressure; and cg the center of gravity.

TABLE 11-9B. Example of Worst Case Disturbance Torque Estimates for FireSat. Magnetic and aerodynamic disturbances are the largest for this small spacecraft.

Disturbance	FireSat Example
Gravity-gradient	For R = (6,378 + 700) km = 7,078 km; I_z = 90 kg·m², I_y = 60 kg·m² and θ = 1 deg (normal mode) or 30 deg (optional target-of-opportunity mode): normal: $T_g = \frac{(3)(3.986 \times 10^{14}\ \mathrm{m^3/s^2})(30\ \mathrm{kg \cdot m^2})\sin(2\ \mathrm{deg})}{(2)(7.078 \times 10^6\ \mathrm{m})^3}$ $= 1.8 \times 10^{-6}$ N·m optional target-of-opportunity: $T_g = 4.4 \times 10^{-5}$ N·m
Solar Radiation	For a 2 m by 1.5 m spacecraft cross-section, a center-of-solar-pressure to center-of-mass difference of 0.3 m, incidence angle of 0 deg and coefficient of reflectivity of 0.6. T_{sp} = (1,367 W/m²) (2 m × 1.5 m) (0.3 m) (1 + 0.6) (cos 0 deg) / (3 × 10⁸ m/s) $= 6.6 \times 10^{-6}$ N·m
Magnetic Field	For R = 7,078 km, a spacecraft magnetic dipole of 1 A·m²* and the worst-case polar magnetic field, M = 2 (7.96 × 10¹⁵ tesla·m³)/(7.078 × 10⁶ m)³ = 4.5 × 10⁻⁵ tesla (= 0.45 gauss) $T_m = 1 \times 4.5 \times 10^{-5} = 4.5 \times 10^{-5}$ N·m
Aerodynamics	For illustration purposes we assume a 3 m² surface, offset from the center of mass by 0.2 m. In a 700-km orbit the velocity is ≈ 7,504 m/s, the atmospheric density (ρ) is ≈ 10^{-13} kg/m³. For C_d, the drag coefficient, use 2.0. F = 1/2 [(10⁻¹³ kg/m³) (2)(3 m²) (7,504 m/s)²] = 1.7×10^{-5} N $T = FL = 1.7 \times 10^{-5}$ N (0.2 m) = 3.4×10^{-6} N·m This is small. At a 100-km orbit, however, $\rho = 10^{-9}$ kg/m³. This results in $T = 3.3 \times 10^{-2}$ N·m, which is significant for our small spacecraft.

* Residual magnetic dipoles can range anywhere from 0.1 to > 20 A·m² depending on the spacecraft's size and whether any onboard compensation is provided. On a small-sized, uncompensated vehicle, 1 A·m² is typical (1 A·m² = 1,000 pole·cm).

The other disturbances on the control system are internal to the spacecraft. Fortunately, we have some control over them. If we find that one is much larger than the rest, we can respecify it to tighter values. This change would reduce its significance but most likely add to its cost or weight. Table 11-10 summarizes the common internal disturbances. Misalignments in the center of gravity and in thrusters will show up during thrusting only and are corrected in a closed-loop control system. The slosh and operating machinery torques are of greater concern but depend on specific hardware. If a spacecraft component has fluid tanks or rotating machinery, the system designer should investigate disturbance effects and ways to compensate for the disturbance, if required. Standard techniques include slosh baffles or counter-rotating elements.

TABLE 11-10. Principal Internal Disturbance Torques. Spacecraft designers can minimize internal disturbances through careful planning and precise manufacturing which may increase costs.

Disturbances	Effect on Vehicle	Typical Values
Uncertainty in Center of Gravity (cg)	Unbalanced torques during firing of coupled thrusters Unwanted torques during translation thrusting	1 to 3 cm
Thruster Misalignment	Same as cg uncertainty	0.1 to 0.5 deg
Mismatch of Thruster Outputs	Similar to cg uncertainty	± 5%
Rotating Machinery (pumps, tape recorders)	Torques that perturb both stability and accuracy	Dependent on spacecraft design; may be compensated by counter-rotating elements.
Liquid Sloshing	Torques due to fluid motion and variation in center-of-mass location	Dependent on specific design; may be controlled by bladders or baffles.
Dynamics of Flexible Bodies	Oscillatory resonance at bending frequencies, limiting control bandwidth	Depends on spacecraft structure.
Thermal Shocks on Flexible Appendages	Attitude disturbances when entering/leaving eclipse	Depends on spacecraft structure. Worst for gravity gradient systems with long inertia booms.

11.1.4 Select and Size ADCS Hardware

We are now ready to evaluate and select the individual ADCS components.

Actuators. We first discuss the actuators, as summarized in Table 11-11, beginning with reaction and momentum wheels. *Reaction wheels* are essentially torque motors with high-inertia rotors. They can spin in either direction, and provide one axis of control for each wheel. *Momentum wheels* are reaction wheels with a nominal spin rate above zero to provide a nearly constant angular momentum. This momentum provides gyroscopic stiffness to two axes, while the motor torque may be controlled to precisely point around the third axis.

In sizing wheels, it is important to distinguish between cyclic and secular disturbances, and between angular momentum storage and torque authority. For 3-axis control systems, cyclic torques build up cyclic angular momentum in reaction wheels, as the wheels provide compensating torques to keep the vehicle from moving. We typically size the *angular momentum capacity* of a reaction wheel (limited by its saturation speed) to handle the cyclic storage during an orbit without the need for frequent momentum dumping. Thus, the average disturbance torque for 1/4 or 1/2 orbit determines the minimum storage capability. The secular torques and our total storage capacity then define how frequently angular momentum must be dumped.

The *torque capability* of the wheels usually is determined by slew requirements or the need for control authority above the peak disturbance torque in order for the wheels to maintain pointing accuracy.

For 3-axis control, at least three wheels are required with their spin axes not coplanar. Often, a fourth redundant wheel is carried in case one of the three primaries

TABLE 11-11. Typical ADCS Actuators. Actuator weight and power usually scale with performance.

Actuator	Typical Performance Range	Weight (kg)	Power (W)
Thrusters			
Hot Gas (Hydrazine)	0.5 to 9,000 N*	Variable†	N/A†
Cold Gas	< 5 N*	Variable†	N/A†
Reaction and Momentum Wheels	0.4 to 400 N·m·s for momentum wheels at 1,200 to 5,000 rpm; max torques from 0.01 to 1 N·m	2 to 20	10 to 110
Control Moment Gyros (CMG)	25 to 500 N·m of torque	> 10	90 to 150
Magnetic Torquers	1 to 4,000 A·m²‡	0.4 to 50	0.6 to 16

* Multiply by moment arm (typically 1 to 2 m) to get torque.
† Chap. 17 discusses weight and power for thruster systems in more detail.
‡ For 700-km orbit and maximum Earth field of 0.4 gauss, the maximum torques would be 4.5×10^{-5} N·m to 0.18 N·m (see Table 11-9B).

fails. If the wheels are not orthogonal (and the redundant one never is), additional torque and momentum authority may be necessary to compensate for the unfavorable geometry. It is also common to use wheels larger than the minimum required in order to use a standard component.

For spin-stabilized or momentum-bias systems, the cyclic torques will cause cyclic rates, while the secular torques cause gradual divergence. We typically design the stored angular momentum, determined by spin rate and inertia of the spinning body, to be large enough to keep the cyclic motion within our pointing specification without active control during an orbit. Periodic torquing will still be required to counteract the secular disturbances. The more angular momentum in the body, the more resistant it is to external torques. An upper limit on the stored momentum, if one exists, may be defined by the fuel cost to precess this angular momentum.

For high-torque applications, *control-moment gyros* may be used instead of reaction wheels. These are single- or double-gimbaled wheels spinning at constant speed. By turning the gimbal axis, we can obtain a high-output torque whose size depends on the speed of the rotor and the gimbal rate of rotation. Control systems with control moment gyros can produce large torques about all three of the spacecraft's orthogonal axes, so we most often use them for agile (high-rate) maneuvers. They require a complex control law and momentum exchange for desaturation. Other disadvantages are high cost and weight.

Spacecraft also use *magnetic torquers* as actuation devices. These torquers use magnetic coils or electromagnets to generate magnetic dipole moments. Magnetic torquers can compensate for the spacecraft's residual magnetic fields or attitude drift from minor disturbance torques. They also can desaturate momentum-exchange systems but usually require much more time than thrusters. A magnetic torquer produces torque proportional (and perpendicular) to the Earth's varying magnetic field. Electromagnets have the advantage of no moving parts, requiring only a magnetometer for field sensing and a wire-wound, electromagnetic rod in each axis. Because they use the Earth's natural magnetic fields, they are less effective at higher orbits. We can easily specify the rod's field strength in amp·turn·m² and tailor it to any application. Table 11-12 describes sizing rules of thumb for wheels and magnetic torquers.

TABLE 11-12. Simplified Equations for Sizing Reaction Wheels, Momentum Wheels, and Magnetic Torquers. FireSat momentum wheels are sized for the baseline requirements. Reaction wheels are sized for the optional design with 30-deg slew requirement.

Parameter	Simplified Equations	Application to FireSat Example
Torque from Reaction Wheel for Disturbance Rejection	Reaction-wheel torque must equal worst-case anticipated disturbance torque plus some margin: $T_{RW} = (T_D)$ (Margin Factor)	For the example spacecraft, $T_D = 4.5 \times 10^{-5}$ N·m (Table 11-9). This is below almost all candidate reaction wheels. We will select a wheel based on storage requirements or slew torque, not disturbance rejection. See below.
Slew Torque for Reaction Wheels	For max-acceleration slews (1/2 distance in 1/2 time): $$\frac{\theta}{2} = \frac{1}{2}\frac{T}{I}\left(\frac{t}{2}\right)^2$$	For the 30-deg slews of the 90 kg·m² spacecraft (Fig. 11-1) in 10 min, this becomes: $$T = 4\theta\frac{I}{t^2} = \frac{4 \times 30 \text{ deg} \times (\pi/180 \text{ deg}) \times 90 \text{ kg·m}^2}{(600 \text{ sec})^2}$$ $$= 5.2 \times 10^{-4} \text{ N·m}$$ This is also a small value.
Momentum Storage in Reaction Wheel	One approach to estimating wheel momentum, h, is to integrate the worst-case disturbance torque, T_D, over a full orbit. If the disturbance is gravity gradient, the maximum disturbance accumulates in 1/4 of an orbit. A simplified expression for such a sinusoidal disturbance is: $$h = (T_D)\frac{\text{Orbital Period}}{4}(0.707)$$ where 0.707 is the rms average of a sinusoidal function.	For $T_D = 4.5 \times 10^{-5}$ N·m (Table 11-9B) and a 700-km orbital period of 98.8 min $$h = (4.5 \times 10^{-5} \text{ N·m})\left(\frac{98.8 \text{ min}}{4}\right)\left(\frac{60 \text{ sec}}{\text{min}}\right)(0.707)$$ $$= 4.7 \times 10^{-2} \text{ N·m·s}$$ A small reaction wheel which gives us storage of 0.4 N·m·s would be sufficient. It provides a margin of > 9 in storage for the worst-case torques.
Momentum Storage in Momentum Wheel	Roll and yaw accuracy depend on the wheel's momentum and the external disturbance torque. A simplified expression for the required momentum storage is: $$T \times \frac{P}{4} = h\theta_a$$ T = torque, P = orbit period, h = angular momentum, θ_a = allowable motion	The value of h for a 0.1 deg yaw accuracy would be $$h = \frac{(4.5 \times 10^{-5} \text{ N·m}) \times 1482 \text{ sec}}{0.1 \times \frac{\pi}{180 \text{ deg}}}$$ $$= 38.2 \text{ N·m·s}$$ T_D is from Table 11-9A. For a 1 deg accuracy, we would need only 3.8 N·m·s
Momentum Storage in Spinner	Same as for a momentum wheel, but with the spin rate: $$\omega_s = \frac{h}{I}$$	For the 0.1 deg accuracy, the spin rate is: $$\omega_s = \frac{(37.3)\text{N·m·s}}{90 \text{ kg·m}^2} = 0.42 \text{ rad/sec} = 4.1 \text{ rpm}$$
Torque from Magnetic Torquers	Magnetic torquers use the Earth's magnetic field, B, and electrical current through the torquer to create a magnetic dipole (D) that results in torque (T) on the vehicle: $$D = \frac{T}{B}$$ Magnets used for momentum dumping must equal the peak disturbance + margin to compensate for the lack of complete directional control.	Table 11-9B estimates the worst-case Earth field, B, to be 4.5×10^{-5} tesla. We calculate the torque rod's magnetic torquing ability (dipole) to counteract the worst-case gravity gradient disturbance, T_D, of 4.5×10^{-5} N·m as $$D = \frac{T}{B} = \frac{4.5 \times 10^{-5} \text{ N·m}}{4.5 \times 10^{-5} \text{ tesla}} = 1 \text{ A·m}^2$$ which is a small actuator. The Earth's field is cyclic at twice orbital frequency; thus, maximum torque is available only twice per orbit. A torquer of 3 to 10 A·m² capacity should provide sufficient margin.

Note: For actuator sizing, the magnitude and direction of the disturbance torques must be considered. In particular, momentum accumulation in inertial coordinates must be mapped to body-fixed wheel axes, where necessary.

Gas jets or *thrusters* produce torque by expelling mass, and are not governed by the same concerns as momentum storage devices. We consider them to be a *hot-gas system*, either bipropellant or monopropellant, when a chemical reaction produces the energy. They are a *cold-gas system* when energy comes from the latent heat of a phase change or from the work of compression without a phase change. Cold-gas systems usually apply to small spacecraft and low-impulse requirements.

Thrusters produce torques and forces that:

- Control attitude
- Control nutation
- Maneuver spacecraft over large angles
- Adjust orbits
- Control the spin rate
- Dump extra momentum from a momentum wheel, reaction wheel, or control moment gyro

Unfortunately, their plumes may impinge on the spacecraft, contaminating surfaces, and they require expendable propellant, dictating spacecraft life. An advantage is that they can provide large, instantaneous torques at any point in the orbit.

We must decide whether we need thrusters, how many we need, and where to locate them. For applications that demand fine control from the thrusters, we may have to specify the minimum impulse from a single thruster pulse—usually 20 ms or greater. Single thrust levels are usually used, unless the complication of dual or variable thrust is required.

Although the baseline FireSat spacecraft will use magnetic torquers, we illustrate the thruster sizing calculations for momentum dumping and the optional slew requirement. We will assume the thruster's moment arm is 0.5 m. Table 11-13 gives procedures and simplified equations, where applicable, for sizing thrusters and estimating propellant. Refer to Chap. 17 for a thorough discussion of propulsion subsystems.

The size of the thrusters and required propellent are small for this example. For the optional system with reaction wheels, slewing can be accomplished with the wheels, avoiding use of propellent. For the baseline momentum bias system, we would use thrusters for the optional slews, though large electromagnets could be used if thrusters were not available and maneuver time were not important.

Sensors. We complete this hardware unit by selecting the sensors needed for control. Consult Table 11-14 for a summary of typical devices, as well as their performance and physical characteristics. Note, however, that sensor technology is evolving rapidly, promising more accurate, lighter-weight sensors for future mission.

Sun sensors are visible-light detectors which measure one or two angles between their mounting base and incident sunlight. They are popular, accurate and reliable, but require clear fields of view. They can be used as part of the normal attitude determination system, part of the initial acquisition or failure recovery system, or part of an independent solar array orientation system. Since most low-Earth orbits include eclipse periods, Sun-sensor-based attitude determination systems must provide some way of tolerating the regular loss of this data without violating pointing constraints.

Sun sensors can be quite accurate (< 0.01 deg) but it is not always possible to take advantage of that feature. We usually mount Sun sensors near the ends of the vehicle to obtain an unobstructed field of view. Sun sensor accuracy can be limited by structural bending on large spacecraft. Spinning satellites use specially designed Sun sensors that measure the angle of the Sun with respect to the spin axis of the vehicle. The data may be sent to the ground for processing or used in a closed-loop control system on board the vehicle.

TABLE 11-13. Simplified Equations for Preliminary Sizing of Thruster Systems. FireSat thruster requirements are small for this low-disturbance, minimal slew application.

Simplified Equations	Application to FireSat Example
Thruster force level sizing for external disturbances: $F = T_D / L$ F is thruster force, T_D is worst-case disturbance torques, and L is the thruster's moment arm	For the worst case T_D of 4.5×10^{-5} N·m (Table 11-7) and a thruster moment arm of 0.5 m $F = \frac{4.5 \times 10^{-5}\ \text{N·m}}{0.5\ \text{m}} = 9.0 \times 10^{-5}\ \text{N}$ This small value indicates slewing rate, not disturbances, will more likely determine size. Also, using thrusters to fight cyclic disturbances uses much fuel.
Sizing force level to meet slew rates (optional zero momentum system): Determine highest slew rate required in the mission profile. Develop a profile that accelerates the vehicle to that rate, coasts, then decelerates. We calculate the thruster force from the acceleration value using the following relationships: $T = F\,L = I\ddot{\theta}$ Solve for F	Assume a 30-deg slew in less than 1 min (60 sec), accelerating for 5% of that time, coasting for 90%, and decelerating for 5%. $\text{Rate}\ (\dot{\theta}) = 30\ \text{deg} / 60\ \text{sec} = 0.5\ \text{deg/sec}$ To reach 0.5 deg/s in 5% of 1 min, which is 3 sec, requires an acceleration $(\ddot{\theta}) = \frac{\dot{\theta}}{t} = \frac{0.5\ \text{deg/sec}}{3\ \text{sec}} = 0.167\ \text{deg/sec}^2 = 2.91 \times 10^{-3}\ \text{rad/sec}^2$ $F = \frac{I\ddot{\theta}}{L} \frac{(90\ \text{kg·m}^2)(2.91 \times 10^{-3}\ \text{rad/sec}^2)}{0.5\ \text{m}} = 0.52\ \text{N}$ This is small but feasible.
Sizing force level for slewing a momentum-bias vehicle: The applied torque T is $T = FLd = h\omega$ where F = average thruster force L = moment arm d = thruster duty cycle (fraction of spin period) h = angular momentum ω = slew rate	For FireSat, allowing 10 min for a 30-deg slew, with 10% duty cycle $= \frac{h\omega}{Ld} \frac{(38.2\ \text{N·m·s})\left(\frac{30\ \text{deg}}{600\ \text{sec}}\right) \times \frac{\pi}{180\ \text{deg}}}{(0.5\ \text{m})(0.1)}$ $= 0.67\ \text{N}$
Thruster pulse life: Develop detailed maneuver profile from mission sequence of events and determine pulse number and length for each segment	Assume example spacecraft uses thrusters only for large mission maneuvers and momentum dumping for 1 sec each wheel once a day. The large maneuver, 30 deg, in 2 axes each week includes a 3-sec acceleration pulse and a 3-sec deceleration pulse. Total Pulses = 2 pulses (start & stop) × 2 axes × 12/yr × 5 yr (maneuver) + 1 pulse × 3 wheels × 365 days/yr × 5 yr (momentum dump) = 240 + 5,475 = 5,715 pulses This is below the typical 20,000 to 50,000 pulse ratings for small thrusters.

TABLE 11-13. Simplified Equations for Preliminary Sizing of Thruster Systems. (Continued) FireSat thruster requirements are small for this low-disturbance, minimal slew application.

Simplified Equations	Application to FireSat Example
Sizing force level for momentum dumping: $F = \frac{h}{Lt}$ where h = stored momentum (from wheel capacity or disturbance torque × time) L = moment arm t = burn time	For FireSat with 0.4 N·m·s wheels and 1-sec burns, $F = \frac{0.4\ \text{N·m·s}}{(0.5\ \text{m}) \times (1\ \text{sec})} = 0.8\ \text{N}$
Propellant: Estimate propellant mass (M_p) by determining the total pulse length, t, for the pulses counted above, multiplying by thruster force (F), and dividing by specific impulse (I_{sp}), and g as follows: $M_p = \frac{Ft}{I_{sp}g}$	To derive the propellant weight from the pulses above, use 3 sec for on-time for each large maneuver pulse, and 1 sec for each momentum-dump pulse at the computed force levels (actual times will change when a thruster is chosen, but the total impulse will be the same). Total Impulse = I = 240 pulses 3 sec/pulse 0.52 N + 5,475 pulses × 1 sec/pulse × 0.8 N = 4,754 N·s then $M_p = \frac{I}{I_{sp}g} = \frac{4{,}754\ \text{N·s}}{200\ \text{sec} \times 9.8\ \text{m/sec}^2} = 2.43\ \text{kg}$ where an I_{sp} of 200 sec for hydrazine is a conservative estimate.

TABLE 11-14. Typical ADCS Sensors. Sensors have continued to improve in performance while getting smaller and less expensive.

Sensor	Typical Performance Range	Wt Range (kg)	Power (W)
Inertial Measurement Unit (Gyros & Accelerometers)	Gyro drift rate = 0.003 deg/hr to 1 deg/hr, accel. Linearity = 1 to 5 × 10^{-6} g/g^2 over range of 20 to 60 g	1 to 15	10 to 200
Sun Sensors	Accuracy = 0.005 deg to 3 deg	0.1 to 2	0 to 3
Star Sensors (Scanners & Mappers)	Attitude accuracy = 1 arc sec to 1 arc min 0.0003 deg to 0.01 deg	2 to 5	5 to 20
Horizon Sensors • *Scanner/Pipper* • *Fixed Head (Static)*	Attitude accuracy: 0.1 deg to 1 deg (LEO) < 0.1 deg to 0.25 deg	 1 to 4 0.5 to 3.5	 5 to 10 0.3 to 5
Magnetometer	Attitude accuracy = 0.5 deg to 3 deg	0.3 to 1.2	< 1

Star sensors have evolved rapidly in the past few years, and represent the most common sensor for high-accuracy missions. Star sensors can be scanners or trackers. *Scanners* are used on spinning spacecraft. Stars pass through multiple slits in a scanner's field of view. After several star crossings, we can derive the vehicle's attitude. We use *trackers* on 3-axis attitude stabilized spacecraft to track one or more stars to derive 2- or 3-axis attitude information. The most sophisticated units not only track the stars as bright spots, but identify which star pattern they are viewing, and output the sensor's orientation compared to an inertial reference. Putting this software inside the sensor simplifies processing requirements of the remaining attitude control software.

While star sensors excel in accuracy, care is required in their specfication and use. For example, the vehicle must be stabilized to some extent before the trackers can determine where they point. This stabilization may require alternate sensors, which can increase total system cost. Also, star sensors are susceptible to being blinded by the Sun, Moon, or even planets, which must be accommodated in their application. Where the mission requires the highest accuracy and justifies a high cost, we use a combination of star trackers and gyros. These two sensors complement each other nicely: the gyros can be used for initial stabilization, and during periods of sun or moon interference in the trackers, while the trackers can be used to provide a high-accuracy, low frequency, external reference unavailable to the gyros. Work continues to improve the sample rate of star trackers and to reduce their radiation sensitivity.

Horizon sensors are infrared devices that detect the contrast between the cold of deep space and the heat of the Earth's atmosphere (about 40 km above the surface in the sensed band). Simple narrow field-of-view fixed-head types (called *pippers* or *horizon crossing indicators*) are used on spinning spacecraft to measure Earth phase and chord angles which, together with orbit and mounting geometry, define two angles to the Earth (nadir) vector. *Scanning horizon sensors* use a rotating mirror or lens to replace (or augment) the spinning spacecraft body. They are often used in pairs for improved performance and redundancy. Some nadir-pointing spacecraft use *staring sensors* which view the entire Earth disk (from GEO) or a portion of the limb (from LEO). The sensor fields of view stay fixed with respect to the spacecraft. This type works best for circular orbits.

Horizon sensors provide Earth-relative information directly for Earth-pointing spacecraft, which may simplify onboard processing. The scanning types require clear fields of view for their scan cones (typically 45, 60, or 90 deg, half-angle). Typical accuracies for systems using horizon sensors are 0.1 to 0.25 deg, with some applications approaching 0.03 deg. For the highest accuracy in low-Earth orbit, it is necessary to correct the data for Earth oblateness and seasonal horizon variations.

Magnetometers are simple, reliable, lightweight sensors that measure both the direction and size of the Earth's magnetic field. When compared to the Earth's known field, their output helps us establish the spacecraft's attitude. But their accuracy is not as good as that of star or horizon references. The Earth's field can shift with time and is not known precisely in the first place. To improve accuracy, we often combine their data with data from Sun or horizon sensors. When a vehicle using magnetic torquers passes through magnetic-field reversals during each orbit, we use a magnetometer to control the polarity of the torquer output. The torquers usually must be turned off while the magnetometer is sampled to avoid corrupting the measurement.

GPS receivers are commonly known as high-accuracy navigation devices. Recently, GPS receivers have been used for attitude determination by employing the differential signals from separate antennas on a spacecraft. Such sensors offer the promise of low cost and weight for LEO missions, and are being used in low accuracy applications or as back-up sensors. Development continues to improve their accuracy, which is limited by the separation of the antennas, the ability to resolve small phase differences, the relatively long wavelength, and multipath effects due to reflections off spacecraft components.

Gyroscopes are inertial sensors which measure the speed or angle of rotation from an initial reference, but without any knowledge of an external, absolute reference. We use them in spacecraft for precision attitude sensing when combined with external references such as star or sun sensors, or, for brief periods, for nutation damping or attitude control during thruster firing. Manufacturers use a variety of physical phenomena, from simple spinning wheels (*iron gyros* using ball or gas bearings) to *ring lasers, hemispherical resonating surfaces*, and *laser fiber optic bundles*. The gyro manufacturers, driven by aircraft markets, steadily improve accuracy while reducing size and mass.

Error models for gyroscopes vary with the technology, but characterize the deterioration of attitude knowledge with time (degrees per hour or per square-root of time). When used with an accurate external reference, such as star trackers, gyros can provide smoothing (filling in the measurement gaps between star tracker samples) and higher frequency information (tens to hundreds of hertz), while the star trackers provide the low frequency, absolute orientation information that the gyros lack. Individual gyros provide one or two axes of information, and are often grouped together as an *Inertial Reference Unit*, IRU, for three full axes. IRUs with accelerometers added for position/velocity sensing are called *Inertial Measurement Units*, IMUs.

Sensor selection. Sensor selection is most directly influenced by the required orientation of the spacecraft (e.g., Earth- or inertial-pointing) and its accuracy. Other influences include redundancy, fault tolerance, field of view requirements, and available data rates. Typically, we identify candidate sensor suites and conduct a trade study to determine the best, most cost-effective approach. In such studies, the existence of off-the-shelf components and software can strongly influence the outcome. In this section we will only briefly describe some selection guidelines.

Full 3-axis knowledge requires at least two external vector measurements, although we use inertial platforms or spacecraft angular momentum (from spinning or momentum wheels) to hold the attitude between external measurements. In some cases, if attitude knowledge can be held for a fraction of an orbit, the external vectors (e.g., Earth or magnetic) will have moved enough to provide the necessary information.

For Earth-pointed spacecraft, horizon sensors provide a direct measurement of pitch and roll axes, but require augmentation for yaw measurements. Depending on the accuracy required, we use Sun sensors, magnetometers, or momentum-bias control relying on roll-yaw coupling for the third degree of freedom. For inertially-pointing spacecraft, star and Sun sensors provide the most direct measurements, and inertial platforms are ideally suited. Frequently, only one measurement is made in the ideal coordinate frame (Earth or inertial), and the spacecraft orbit parameters are required in order to convert a second measurement or as an input to a magnetic field model. The parameters are usually uplinked to the spacecraft from ground tracking, but autonomous navigation systems using GPS are also in use (see Sec. 11.7).

FireSat sensors. The external sensors for FireSat could consist of any of the types identified. For the 0.1 deg Earth-relative pointing requirement, however, horizon sensors are the most obvious choice since they directly measure two axes we need to control. The accuracy requirement makes a star sensor a strong candidate as well, although its information needs to be transformed to Earth-relative pointing for our use. The 0.1 deg accuracy is at the low end of horizon sensors' typical performance, and we need to be careful to get the most out of their data.

TABLE 11-15. FireSat Spacecraft Control Components Selection. A simple, low-cost suite of components fits FireSat's needs.

Type	Components	Rationale
Actuation Devices	(1) Momentum Wheel	• Pitch axis torquing • Roll and yaw axis passive stability
	(3) Electromagnets	• Roll and yaw control • Pitch wheel desaturation
Sensors	Horizon Sensor	• Provide basic pitch and roll reference • Can meet 0.1 deg accuracy • Lower weight and cost than star sensors
	Sun Sensors	• Initially acquire vehicle attitude from unknown orientation • Coarse attitude data • Fine data for yaw
	Magnetometer	• Coarse yaw data

TABLE 11-16. FireSat Spacecraft Control Subsystem Characterized. The baseline ADCS components satisfy all mission requirements, with thrusters available if required.

Components	Type	Weight (kg)	Power (W)	Mounting Considerations
Momentum Wheel	Mid-size, 40 N·m·s momentum	< 5 total, with drive electronics	10 to 20	Momentum vector on pitch axis
Electromagnets	3, 10 A·m^2 capacity each	2, including current drive electronics	5 to 10	Orthogonal configuration best to reduce cross-coupling
Sun Sensors	4 wide-angle coarse sensors providing 4 π steradian coverage; ≈ 0.1 deg accuracy	< 1 total	0.25	Free of viewing obstructions and reflections
Horizon Sensors	Scanning type (2) plus electronics; 0.1 deg accuracy	5 total	10	Unobstructed view of Earth's horizon
Optional Thrusters	Hydrazine; 0.5 N force	Propellant weight depends on mission	N/A	Alignments and moment arm to center of gravity are critical
Magnetometer	3-axis	< 1	5	Need to isolate magnetometer from electromagnets, either physically or by duty-cycling the magnets.

We assume we also need a yaw sensor capable of 0.1 deg, and this choice is less obvious. (Often, it is useful to question a tight yaw requirement. Many payloads, e.g., antennas, some cameras, radars, are not sensitive to rotations around their pointing axis. For this discussion, we will assume the requirement is firm.) We could use sun sensors, but their data needs to be replaced during eclipses. Magnetometers don't have the necessary accuracy on their own, but with our momentum-bias system, roll-yaw coupling, and some yaw filtering, a magnetometer-Sun sensor system should work.

At this point, we consider the value of an inertial reference package. Such packages, although heavy and expensive, provide a short-term attitude reference which would permit the Earth vector data to be used for full 3-axis knowledge over an orbit. A gyro package would also reduce the single-measurement accuracy required of the horizon sensors, simplifying their selection and processing. Such packages are also useful in the control system if fast slews are required. Although nice to have, an inertial package does not seem warranted for FireSat. Table 11-15 summarizes our hardware selections.

Once the hardware selection is complete, it must be documented for use by other system and subsystem designers as follows:

- Specify the power levels and weights for each assembly
- Establish the electrical interface to the outside world
- Describe requirements for mounting, alignment, heating, or coding
- Determine what telemetry data we must process
- Document how much software we need to support equations of motion

Specific numbers depend on the vendors selected. A typical list for FireSat might look like Table 11-16, but the numbers could vary considerably with only slight changes in subsystem accuracies or slewing requirements.

ADCS Vendors. Typical suppliers for ADCS components are listed in Table 11-17.

TABLE 11-17. ADCS Component Suppliers. Aerospace mergers can result in sudden name changes.

	Sensors					Actuators			
Company	Sun	Earth	Magnetometers	Star	Inertial (Gyros)	Momentum/ Reaction Wheels	CMGs	Electro-magnets	Thrusters
Adcole Corporation	✔								
AlliedSignal					✔	✔			
Ball Aerospace and Technologies Corp.	✔			✔					
Billingsley Magnetics			✔						
CAL Corporation				✔					
EDO (Barnes) Corp.	✔	✔							
Honeywell Space Systems					✔	✔	✔		
Ithaco Space Systems Inc.	✔	✔	✔			✔		✔	
Kearfott Guidance & Navigation Corporation					✔	✔			
Litton Industries					✔				
Lockheed Martin	✔				✔				
Kaiser Marquardt Corp.									✔
Matra Marconi Space						✔			
Meda, Inc.			✔						
Microcosm, Inc.								✔	
Primex Technologies									✔
Raytheon Systems, Inc.				✔					
Servo Corp. of America		✔							
Smiths Industries Defense Systems					✔				
Teldix GmbH						✔			
TRW									✔
Watson Industries, Inc.			✔		✔				

11.1.5 Define the Control Algorithms

Finally, we must tie all of the control components together into a cohesive system. We begin with a block diagram for a single-axis control system (See Fig. 11-3). As we refine the design, we add or modify feedback loops for rate and altitude data, define gains and constants, and fine tune the equations of motion. To do so, we need good mathematical simulations of the entire system, including internal and external disturbances. Usually, linear differential equations with constant coefficients describe the dynamics of a control system, thus allowing us to analyze its performance with the highly developed tools of linear servomechanism theory. With these same tools, we can easily do linear compensation to satisfy specifications for performance.

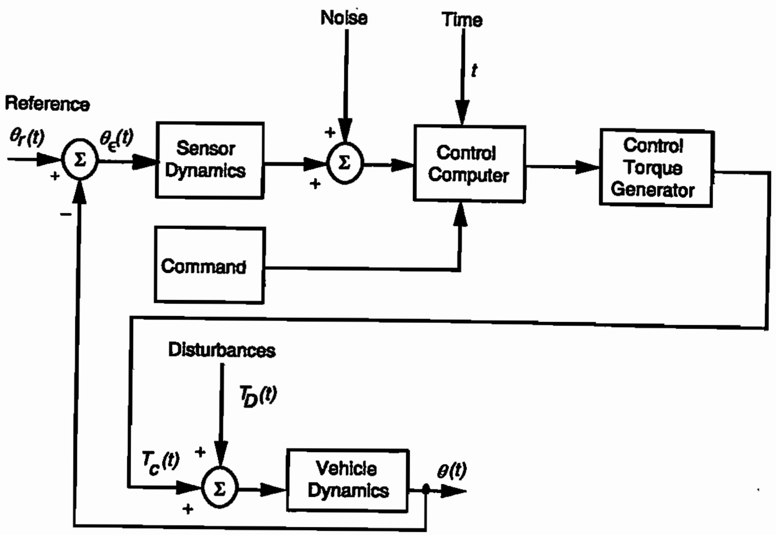

Fig. 11-3. Diagram of a Typical Attitude Control System with Control Along a Single Axis. Control algorithms are usually implemented in an onboard processor and analyzed with detailed simulations.

We typically apply linear theory only to preliminary analysis and design. As the design matures, nonlinear effects come strongly into play. These effects may be inherent or intentionally introduced to improve the system's performance. Feedback control systems are of two kinds, based on the flow of their control signals. They are *continuous-data* systems when their control signals flow continuously without interruption; they are *sampled-data* systems when sampling occurs at equal intervals. Most modern spacecraft process data through digital computers and therefore use control systems that sample data.

Although it is beyond the scope of this handbook to provide detailed design guidance on feedback control systems, the system designer should recognize the interacting effects of attitude control system loop gain, capability of the attitude control system to compensate for disturbances, accuracy of attitude control, and control system bandwidth.

Three-axis stabilization. Different types of active control systems have different key parameters and algorithms. Frequently, 3-axis control can be decoupled into three independent axes. The most basic design parameter in each axis is its *position gain*, K_p.* This is the amount of control torque which results from a unit attitude error and can be expressed in N·m/deg or N·m/rad. The position gain is selected by the designer and must be high enough to provide the required attitude control accuracy in the presence of disturbances, or $K_p \geq T_D / \theta_\varepsilon$, where K_p is position gain, T_D is peak disturbance torque, and θ_ε is allowable attitude error.

The value of the position gain also determines the attitude control system bandwidth and speed-of-response. The *bandwidth* is given by $\omega_n = (K_p / I)^{1/2}$, where I is the spacecraft moment of inertia. The bandwidth defines the frequency at which control authority begins to diminish. Attitude control and disturbance rejection are effective from 0 frequency (d.c.) up to the bandwidth. *Speed of response* is approximately the reciprocal of bandwidth. Note that position gain is inversely proportional to allowable error and bandwidth is proportional to the square root of position gain. Therefore, high accuracy implies high position gain and high bandwidth. However, high bandwidth may cause bending resonances to affect control system performance.

With the relations given, the system designer can estimate required position gain from his estimates of disturbance torque and accuracy requirements. He can use this estimate to compute control system bandwidth. This allows him to specify minimum bending frequencies as discussed below.

In defining algorithms for the control system, we must also consider whether the vehicle will have flexible-body effects that can make the vehicle unstable. Spacecraft with flexible appendages such as antennas, booms, and solar panels may produce slight warping at their natural frequencies. Control torques and external-disturbance torques will cause structural vibrations, in some cases close to or within the control system's bandwidth. The lowest natural frequencies of flexible components should be at least an order of magnitude greater than the rigid-body frequencies before we can neglect flexibility. For further discussion of how structural flexibility affects the control subsystem, see Sec. 3.12 of Agrawal [1986].

Spin stabilization and momentum bias. The fundamental concept in spin stabilization is the nutation frequency of the vehicle. For a spinning body, the inertial nutation frequency (ω_{ni}) is equal to

$$\omega_{ni} = \frac{I_s}{I_T} \omega_s \tag{11-1}$$

where I_s is the spin axis inertia, I_T is the transverse axis inertia, and ω_s is the spin frequency.

For a momentum-bias vehicle with a stable body and a momentum wheel (or a dual-spin vehicle with a stable platform and a spinning rotor), the nutation frequency is

$$\omega_{ni} = \frac{h}{I_T} \tag{11-2}$$

* In its simplest form, a spacecraft attitude control system can be represented in s-domain as a $1/I_s^2$ plant and may be controlled by a *proportional plus derivative* (*PD*) *controller* where $T_c \equiv K_p\, \theta_\varepsilon + K_r\, \dot{\theta}_\varepsilon$. The position gain, K_p, controls system bandwidth and the rate gain, K_r, controls damping.

where h is the angular momentum of the spinning body. Thus, spacecraft with large inertias and small wheels have small nutation frequencies (i.e., long periods).

Attempting to move the vehicle with a bandwidth faster than the nutation frequency causes it to act more like a 3-axis vehicle. In general, we attempt to control near the nutation frequency or slower, with correspondingly small torques. In this area, the vehicle acts like a gyroscope, with the achieved angular rate, ω, proportional to the applied torque, T:

$$\omega = T/h \tag{11-3}$$

where h is the system angular momentum.

A lower limit on control bandwidth is usually provided by the orbit rate ω_o, which for a circular orbit is

$$\omega_0 = \sqrt{\mu / r^3} \tag{11-4}$$

where $\mu = 3.986 \times 10^{14}$ m^3/s^2 and r is the orbit radius.

Attitude determination. A full discussion of determination algorithms requires a dedicated reference such as Wertz [1978]. We will highlight only some of the basic concepts.

The basic algorithms for determination depend on the coordinate frames of interest (e.g., the sensor frames, local vertical frame, or Earth-centered inertial frame), and the geometry of the measurements, parameterized by *Euler angles* (such as roll, pitch, and yaw) or *quaternions* (which are scaled vectors for Eigen-axis rotations of coordinate frames). Inertial platforms and star sensor data usually are suited to inertial quaternions, while Earth-pointing spacecraft often use a local-vertical, aircraft-like set of Euler angles.

Simple spacecraft may use the sensor readings directly for control, while more complex vehicles or those with higher accuracy requirements employ some form of averaging, smoothing, or Kalman filtering of the data. The exact algorithms depend on the vehicle properties, orbit, and sensor types used.

FireSat algorithms. For our momentum-bias FireSat example, control separates into pitch-axis control using torque commands to the momentum wheel, and roll-yaw control using current commands to the electromagnets. The pitch-wheel desaturation commands must also be fed (at a slow rate) to the magnets. The pitch-wheel control is straightforward, using proportional-plus derivative and, optionally, integral control. The roll-yaw control design starts by using the linearized nutation dynamics of the system, and is complicated by the directional limitations of electromagnetic torque (the achievable torque is perpendicular to the instantaneous Earth magnetic field).

The nadir-oriented control system may use an Earth-referenced, aircraft-like Euler angle (roll, pitch, yaw) set, although quaternions should also be considered for their lack of singularities during off-nominal pointing. The horizon sensors directly read two of the angles of interest, pitch and roll. Yaw needs to be measured directly from Sun position (during orbit day) or from the magnetometer readings (using a stored model of the Earth's field), or inferred from the roll-yaw coupling described earlier. The magnetic field and Sun information require an uplinked set of orbit parameters, and increase the computational requirements of the subsystem. Overall, meeting the 0.1 deg yaw requirement when the Sun is not visible will be the toughest challenge facing the ADCS designer, and a form of coasting through the blackouts, without direct roll-yaw control, may be most appropriate.

11.2 Telemetry, Tracking, and Command

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Adapted from SMAD II, Sec. 11.2 "Communications," by John Ford

The *telemetry, tracking, and command* (*TT&C*) or *communications subsystem* provides the interface between the spacecraft and ground systems. Payload mission data and spacecraft housekeeping data pass from the spacecraft through this subsystem to operators and users at the operations center. Operator commands also pass to the spacecraft through this subsystem to control the spacecraft and to operate the payload. We must design the hardware and their functions to pass the data reliably for all the spacecraft's operating modes. For a discussion of how we collect and manipulate housekeeping and payload data, see Sec. 11.3, Chap. 9, and Chap. 16. Chapter 13 discusses the communication link design, and Morgan and Gordon [1989] provide a wealth of information on spacecraft communications.

The subsystem functions include the following:

- Carrier tracking (lock onto the ground station signal)
- Command reception and detection (receive the uplink signal and process it)
- Telemetry modulation and transmission (accept data from spacecraft systems, process them, and transmit them)
- Ranging (receive, process, and transmit ranging signals to determine the satellite's position)
- Subsystem operations (process subsystem data, maintain its own health and status, point the antennas, detect and recover faults.)

Table 11-18 presents specific subfunctions to accomplish these main functions. Subsystem designers must ensure that all of these functions operate reliability to accomplish the spacecraft mission.

As part of carrier tracking, most satellite TT&C subsystems generate a downlink RF signal that is phase coherent to the uplink signal. *Phase coherence* means that we transmit the downlink carrier so its phase synchronizes with the received phase of the uplink carrier. This process is the *coherent turnaround* or *two-way-coherent mode.* The coherent turnaround process creates a downlink carrier frequency precisely offset from the uplink carrier by a predefined numerical *turnaround ratio*. This is the ratio of the downlink carrier frequency to the uplink carrier frequency.This operational mode can only exist when the transmitter phase-locks to the received uplink carrier. For a given uplink signal, the downlink signal has a constant phase difference. For NASA's GSTDN-compatible transponders, the receiver downcoverts the uplink carrier, and creates a voltage such that the receiver's voltage-controlled oscillator runs at precisely 2/221 times the uplink carrier frequency. The oscillator frequency goes to the transmitter which multiplies it by a factor of 120. Therefore, the composite transmitter downlink is 120 × 2/221 = 240/221 times the uplink frequency, which is the turnaround ratio for NASA-compatible transponders. The turnaround ratio for transponders compatible with SGLS is 256/205. The two-way-coherent mode allows the ground station to know more exactly the downlink signal's frequency and to measure the Doppler shift, from which it computes the *range rate* or line-of-sight velocity between the spacecraft and the tracking antenna. This knowledge allows operators to