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Chapter

Satellite Control System:
Part II - Control Modes, Power, Interface, and Testing

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Abstract

This part II of the chapter Satellite Control System (SCS) was originally planned for publishing in the book Satellite Systems (Acad. Ed. Dr. T. Nguyen), dedicated to the Systems Design, Modeling, Simulation, and Analysis, together with the Part I (SCS Architecture and Main Components). However, restricted volume of this book did not let the publisher to put then this part in the book. The book Recent Applications in Remote Sensing (Acad. Ed. Prof. M. Marhgany) considers the various aspects of the optical and radiolocation sensing and imaging of the Earth surface from Space. Consequently, as it was presented in the Part I, the author adheres to the point of view here that satellite is not just a platform to carry in Space a payload, but is equipment integration system and its designer is in charge for fully integrated and Space-qualified Space segment, which with the corresponding operation and ground equipment would be capable to successfully execute dedicated mission (Remote Sensing). The material, presented in this part, briefly highlights the basic aspects of SCS control modes, electric and informational interface, and ground testing, which would promote successful interaction with satellite payload, such as Remote Sensing subsystem and mission success.

Keywords: satellite control, attitude and orbit, determination, estimation, sensors, actuators, coordinate systems, reference frame, state estimation and Kalman filtering, earth gravity and magnetic fields, interface, assembling integration and testing (AIT), space qualification

1. Introduction

For a remote sensing satellite, equipped with a remote sensing payload, the Satellite Control system (SCS) is very important, providing for the payload required attitude a position in space.

Hence, the payload functionality and its performance essentially depend on SCS characteristics.

Often, specifically, the payload provider is responsible for satellite system integration and mission success. That is why payload company engineers should be familiar with SCS and its role in the satellite mission performance. Below in the
introduction some physical principles, showing dependence of the remote sensing payload characteristics on SCS, are briefly discussed. Historically radiolocation sensing of underlying Earth surface (footprints) started from the high-altitude air patrol aircrafts, performing the military reconnaissance purposes. They were equipped with special on-board Radio Location Stations (RLS), having Side-Looking Antenna (SAR). The resolution (image quality) of such an RLS is essentially dependent on the SAR available length, which for an aircraft cannot exceed a few meters. With the development of space exploration, special remote sensing satellites became available for the Earth observation, and the military reconnaissance purposes were essentially extended for the civil applications such as exploration of Earth-borne disasters, rescuing, agriculture, forestry, and others. Using for Earth observation space platforms brought to this process new important benefits. The main observations from them are as follows: broader instantaneously observed from the space areas (spot 20–30 km), high resolution (3–5 m), and the capability to observe in relatively short periods of time big areas of Earth with periodically repeatable underlying tracks. Technologically, by developing RLS equipment, new benefits were achieved with using much longer antennas (about 10–15 m) and synthesized analytically aperture (SSAR—side looking synthetic aperture radar). More information about RLS SSAR technology can be found in Refs. [1–6]. Considering Earth remote sensing system, we have to emphasize that successful operation of such a system is not available without physical (stabilization) or analytical availability of satellite angular attitude and its position between SSAR coherent radio pulses, used in system memory for building SSAR analytical aperture.

Any Space country using remote sensing satellites has to take care first about satellite attitude and orbital determination and control or, in other words, about SCS and its precision [7–9].

In Figure 1 below three generations of Canadian Earth observation satellites are presented, Radarsat-1, Radarsat-2, and Radarsat Constellation (RCM). Canada is a pioneer of using SAR technology for the civil tasks for Earth Remote sensing and since launch of Radarsat-1 in November of 1995 has accumulated a big experience and gained tremendous achievements in this area.

Figure 2 presents the basic SAR principles to get radiolocation image reflected from Earth radio signal.

The satellite with on-board RLS-SAR orbits the Earth with the orbital velocity \( \nabla \) in the flight direction. The microwave beam is transmitted obliquely at right angles to the direction of flight illuminating a swath. Slant range \( \rho \) (reflected signal corresponding time delay) is measured, assuming that the flight altitude \( h \) and the look angle and satellite position in orbit are known and nominal. Let us consider SAR with a real aperture antenna, Figure 3.

Figure 1. Canadian family of earth observation SAR satellites. Copyright: CSA//www.asc-csa.gc.ca.
Then, SAR beam footprint dimensions in lateral (Y) and longitudinal (X) directions ($R_y$ and $R_x$ resolutions) can be found with the following formulas [1, 4, 6]:

Figure 2. SAR principles. $\rho$ is slant range, $\theta$ is look angle, $h$ is flight altitude, SC is satellite, SAR is RLS SAR antenna, XYZ is satellite body frame, $\vec{V}$ is flight velocity, $l$ is the antenna length.

Figure 3. Left (red): SAR antenna, right (green): SAR beam footprint. $L$ is antenna length, $W$ is antenna width, $R_x$ and $R_y$ are SAR beam footprint dimensions, $\vec{V}$ satellite ground track vector.

Then, SAR beam footprint dimensions in lateral (Y) and longitudinal (X) directions ($R_y$ and $R_x$ resolutions) can be found with the following formulas [1, 4, 6]:
\[ R_y = \frac{c \tau}{2 \sin \theta_y} = \frac{\Delta \lambda}{2 \sin \theta_y} \quad (1) \]
\[ R_x = \frac{h \lambda}{L \cos \theta_x} \quad (2) \]

where \( c \) is light speed in vacuum (~300,000 km/s), \( \tau \) is RLS transmitted pulse duration, \( \lambda \) is RLS transmitted carrier frequency wavelength, \( \Delta \lambda \) is RLS transmitted pulse bandwidth, \( h \) is flight altitude, \( \theta_x \) and \( \theta_y \) are SAR look angles in X and Y directions correspondingly.

For example, for following numerical data (Radarsat-1):
\[ L = 15 \text{ m}, \ \ W = 1.5 \text{ m}, \ \ h = 800 \text{ km}, \ \ \theta_x = 85^\circ, \ \ \theta_y = 23^\circ, \]
\[ \lambda = 0.06 \text{ m (C-band)}, \ \ \Delta \lambda = 10 \text{ m} \]

\[ R_y = \frac{\Delta \lambda}{2 \sin \theta_y} = \frac{10}{2 \sin 23^\circ} = 12.8 \text{ m} \]
\[ R_x = \frac{h \lambda}{L \cos \theta_x} = \frac{800000 \cdot 0.06}{15 \cos 8^\circ} = 3212 \text{ m} \]

As one can see from the numerical example above, the physical aperture SAR antenna (15 m) can provide quite a good resolution in the lateral direction, but not good enough in the longitudinal. However, this resolution can be drastically improved with the synthetic aperture SAR antenna (SSAR) [1, 4, 6] when all reflected pulses, collected during certain period of time of Earth radiation, are summarized in SSAR RLS on-board computer analytically with the purpose to get RLS image, like it could be created by a physical antenna with a big length.

In this case, much higher longitudinal resolution can be achieved that theoretically can be expressed by the formula:
\[ R_x = \frac{L}{2} \quad (3) \]

This formula for the numerical example below provides the longitudinal resolution \( R_x = 7.5 \) m that drastically improves the resolution of the same RLS, but with physical SAR aperture.

It should be mentioned that formulas ((1)–(3)) assume a certain steady nominal satellite angular orientation (e.g., zero), set by the three Euler angles: Pitch (\( \alpha_y = 0 \)), Roll (\( \alpha_x = 0 \)), and Yaw (\( \alpha_z = 0 \)) that in turn assume absolutely accurate satellite attitude determination and control. Also, the process of synthesis of the artificial SAR aperture–SSAR assumes absolute accurate knowledge of satellite position. In practice, the attitude and the position are measured and controlled with certain errors that lead to change of SAR RLS resolution and as a result, to the distortion of RLS picture (deterioration of image quality). Not only big steady errors impact on the image quality, but a small jitter also. Therefore, for the SCS of Remote Sensing satellites where the payload is SSAR RLS essential are accuracy requirements that can be transformed in resulted SSAR resolution distortion errors. Analysis of this effect can be found in special literature [3, 10]. Here, we just introduce the reader to some specific SCS tasks that can help for understanding of the integration process of SSAR RLS and SCS on satellite. The reader himself can carry out the impression about the importance of proper installation (mechanical interface) and mutual alignment of the mechanical axes of SAR, satellite bus, GPS antenna, and attitude determination devices (e.g., the Star Tracker). For modern
satellites, we can provide some approximate numbers, related to a precise SCS performance (attitude knowledge: $15' - 2''$, attitude control: $0.5'' - 1''$, position knowledge: $10 - 30$ m, position control $100 - 150$ m).

Required SCS functionality and the performance must be ensured by a certain design order, prescribed by the System Engineering discipline and validated during Assembling Integration and Test complain in the Space Qualification Functional test. Some of these aspects, namely: Control Modes, Power, Interface, and Testing are covered by this Part II material presented in the book.

2. Typical SCS control modes

SCS dynamics can usually be presented by closed negative feedback control loop, which consists of three typical components: plant (satellite), observer-estimator (sensors and navigation-attitude/orbit determination algorithms), and controller (actuators and control algorithms$^1$). Modern approach to its design is analytical synthesis, based on optimal/suboptimal algorithms, provided by System Estimation and Control Theory [11, 12] and Mathematical Model-Based Design tools from the MathWorks Inc. [13].

However, in practice (after evaluation of potentially available optimal solution), conventional engineering design, based on former experience, still has been widely used. This approach brings some generic system (SCS) architecture, components, and operational modes.

Typical SCS (mainly, attitude control ACS) modes are as follows: Idle, Acquisition, Pointing, Maneuvering, and Safe Hold Mode. Mainly, all of them can be activated/deactivated by the ground commands from the Mission Control Center (MCC) and/or automatically (by on-board software). Orbital maneuvering and sometimes attitude (slew) are executed exceptionally by the ground commands.

Anyway, a special command flag is generated upon reception of the TLM mode transition command from MCC or after analyzing some internal SCS flags, generated following the operational logic, time, and system components' state and status.

2.1 Idle

After satellite separation from the launch vehicle before starting AODCS operational modes, it could be in the so-called IDLE mode. It checks the system and its components’ state, satellite attitude and angular velocity, and orbit. The system is powered (ON) and activated (operational), except of the actuators. It provides from the sensors’ TLM data to MCC for operational analysis. Satellite actuators are not controlled, and it has random attitude and free rotation initially initiated by the separation pulse from the launch rocket separation mechanism. If ground analysis confirms the system state is nominal and angular motion is safe to have sufficient electric power and thermal conditions, then SCS actuators can be activated to start satellite control.

2.2 Acquisition

This mode can consist of two phases: de-tumbling and initial acquisition mode. If premature de-spinning is required and applied, then it is usually performed with MAG and MTR, using B-dot algorithm. Three-axis magnetometer (MAG)

$^1$ Single satellite orbital control is usually executed by the telemetry (TLM) control command from ground.
output provides measured Earth magnetic field induction vector $\mathbf{B}$. Its components are differentiated and become proportional to $\mathbf{B}_x, \mathbf{B}_y, \mathbf{B}_z$. With appropriate control gains, these signals are applied to the magnetic torque rods (MTRs). Initial spinning is decelerated and the dumping process is finalized to the slow satellite rotation (a few degrees per sec) about the local vector of Earth magnetic induction $\mathbf{B}$. Next starts the acquisition phase with coarse three-axis attitude determination and control. It starts with three-axis attitude determination and PID control (e.g., MAG, Sun Sensor-SS, MTR, and the Reaction Wheels unit—RW [9]). In this mode, satellite body axes $\text{XYZ}$ are prematurely aligned in parallel with desired reference axes $\text{X}_r, \text{Y}_r, \text{Z}_r$. This mode is usually fast. Control loop bandwidth in this mode is wide, transfer process termination time is as short as possible, final attitude accuracy is coarse (about a few degrees). In some applications, this mode can start directly without previous application of the de-spinning sub-mode.

2.3 Pointing

This usually is the operational working mode, required for the successful payload operation. In this mode, the sensitivity axis of satellite payload instrument is accurately pointed in the required direction. Accurate alignment (about a few angular minutes) with the reference frame axes (where the required direction for the payload instrument is set) should be achieved in this mode. The most accurate attitude sensors (e.g., Star Tracer-ST) are applied. The control loop bandwidth can be narrowed to filter external disturbances and measured noise more effectively. In this mode, the control is slow but precise.

2.4 Maneuvering

2.4.1 Orbital

Orbital control thrusters are activated by the computed autonomously on-board or sent from ground TLM command to perform scheduled orbit correction/ maneuver. Pre-calculated thruster activation time $\Delta t$ is used to execute satellite orbit correction pulse $\Delta V = T \Delta t$ (where $T$ is thrusters’ force). For example, in the orbital flight direction to increase degraded with time satellite orbit altitude.

2.4.2 Attitude (slew)

In this mode, satellite is controlled in the closed control loop to turn it by the desired angle to point it in new desired direction. Control law in this mode can be as follows:

$$T_c = -k_p(\alpha - \alpha_c) - k_d\dot{\alpha}$$  \hspace{1cm} (4)

where $T_c$ is control torque, $k_p$ and $k_d$ are proportional and damping control gains, $\alpha$ and $\dot{\alpha}$ are angular deviation, and the velocity $\alpha_c$ is desired attitude angle.

2.5 Safe hold mode (SHM)

This mode is commanded in some dangerous situations, when satellite life critical failure or flight anomaly is automatically detected on-board by on-board computer software (OBC SW) or identified on ground by satellite operators after TLM data analysis. In this mode, SCS system task is to keep an appropriate satellite angular orientation with respect to the Sun and the Earth providing
sufficient thermal, electrical (solar panel energy generation), and radio communication (antennas orientation) conditions for as long as possible time (ideally the indefinite SHM) and consuming as less power as possible (battery electric energy and attitude control thrusters cold gas). Idealistically, a satellite should have a passive SHM, when SCS system can be in the state OFF. During the SHM, the Operation Team should resolve the problem that caused this mode transition and start the recovery procedure (transition in the Acquisition mode). Commands to transition in and recovery from SHM can be considered as the final results (command flags) of the special satellite Failure Detection Isolation and Recovery (FDIR) [14] algorithm that can be realized outside of SCS.

3. Electric power and informational interface

3.1 Electric power

Satellite bus using solar panels (SP), on-board rechargeable batteries, and centralized Power Management Unit (PMU) [15] supplies SCS with available DC voltage. For example, 28 V/50 V and power 500 W from one SP at Sun incidence angle <5 deg. If satellite has 2 SP nominally permanently facing sun, then available-electric power is about 1 kW. During Sun eclipse periods and when sunlight is not sufficient for the SP to generate enough electric power, on-board batteries are used. For example, let us assume that two lithium-ion batteries (voltage 28 V, capacity 12 Ah (350 Wh), depth of discharge—DOD = 50% each) are used to provide storage power during Sun eclipse periods, contingency shadowing, or SP failure.

AEU power convertors convert PMU voltage in lower voltages required for SCS OBC, sensors, and actuators (e.g., 3.3 V, 5 V, 12 V). SCS power consumption depends on its current operational mode and may vary from the nominal Pn to the minimum (Pm) value. For example, SCS power budget is as follows:

- OBCS (related to SCS part)—10 W/3 W, GPS-7 W, 3-axis MAG-0.5 W, SS-0 W (photo sensor) ST-8 W, 3xRS-1.2 W, 3xRW(s)-240 W, 3zMTR(s)-60 W.
- Then for the nominal operation Pn = 397.5 W and for the active SHM (only OBCS—(3 W), SS, MAG, MTRs are “ON”) Pm = 63.5 W.
- This example shows that satellite can nominally operate with fully lightened SP consuming for SCS approximately 40% of available generated on-board by SP electric power.
- SHM: if only satellite life essential equipment is powered on in this mode and if by some reason Sun direction is totally lost or has not been captured, then AODCS can work for about 6 hours without recharging the batteries.

3.2 Electric interface

Mainly two types of interface are used for informational connection SCS equipment and date exchange, as follows:

1. Analogue interface with separate pair of wires in satellite harness. This type is usually applied within AODCS for simple analog devices thermistors, Sun sensors, horizon sensors, etc.
2. Digital bus lines [15] that are applicable for all satellite digital equipment. This type is used with digital devices that have embedded digital computer such as GPS, ST, etc. Electrical interface for all AODCS equipment is usually is defined in Interface Control Document(s) (ICD), that is, essentially, data exchange protocol defining also signal electrical characteristics, connectors, and pins. It is worth to mention here, at least, two following busses:

a. MIL-STD-1553B [16] is US military standard that defines a TDM multiple-source-multiple-sink data bus. By definition, MIL-STD-1553B is a bidirectional, half-duplex (when transmit cannot receive) deterministic communications protocol with central control (i.e., on-board computer or OBC), where each member (i.e., remote terminal) can receive or transmit data. A 1553B network consists of four major components: transmission media, remote terminals, a bus controller, and a bus monitor. The transmission media is a twisted, shielded wire pair with direct or transformer coupling. The data rate is 1 Mbps of Manchester-encoded, bi-phase data stream. Up to 32 words can comprise a single message in which each word is 20 bits long. One system can accommodate up to 31 remote terminals, a bus controller, and a bus monitor (Figure 4).

b. RS-422 (TIA/EIA-422) [17], as it was named by the American Standard National Institute ANSI, is a technical standard that specifies electrical characteristics of a digital signal circuit used by International Electronic Industry. It is digital, serial, asynchronous, one direction, differential, point-to-point line (1 transmitter and 10 receivers, 10 Mbps) interface.

Figure 5 shows that a differential signaling interface circuit consists of a driver with differential outputs and a receiver with differential inputs.

With using voltage between the wires A and B and the ground, the transmitter transmits and receives serial flow of digital data in the binary form 0/1.

Figure 4.
MIL STD 1553 bus.
4. SCS space environment protection and testing

The problems with Ground Tests of Space Systems (SS) at first appeared together with the launch of the first human-made Earth orbiting satellites: Sputnik 1 (1957, USSR, launcher R-7), Explorer-1 (1958, USA, launcher RS-29/Juno), Alouette-1 (1962, Canada, launcher DM-21/Thor-Agena B, USA). Unlike air flight vehicles, flying mainly below an altitude of 25 km in the Earth’s atmosphere, space vehicles-SS (under the acronym SS in further consideration we will understand Space vehicle (spacecraft-S/C) and their equipment and components—S/C subsystems) should fly in Space at altitude above 225 km, practically without atmospheric pressure, in other words in a vacuum, and in addition be affected by the cosmic radiation.

For the air vehicles (airplanes), environmental conditions at that time were already studied and well known, and ground test procedures existed and were almost conventional. But for the SS they were totally new, as well as the launch mechanical impact. Therefore, they were to be studied and ground test types, methodology, and the procedures developed.

By now, it has already been done and presented in many International and National standards and regulations.

After studying space environment and accumulation of some experience with launch and operation of SS, a new group of special ground tests was developed and introduced in the form of related standards and following procedures and documents presented in [8, 18, 19]. This group of tests generally includes the following types: Thermo and Vacuum (TVAC), Vibration and Strength, Radio Communication and Electro Magnetic Compatibility (EMC), final refinement and verification of system assembling and integration (AIT). These tests are finalized by the customer or authorized independent expert conclusion about launch readiness and named Space Qualification (SQ).

4.1 Environmental conditions

SCS system should be designed to work in Space environment conditions that briefly can be characterized by the data below. It has to have special protection measures to satisfy space requirements [8]. It also should be taken into account that different system components may be located inside or outside (SS, ST, HS, Thrusters, GPS antenna) of the satellite and be installed close to the nominally hottest or the coldest surface of its body. However, typically all system components are subjects of the environmental tests [8, 14] to verify different requirements for the internal and external system devices.

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Under the acronym SS in further consideration we will understand Space vehicle (spacecraft-S/C) and their equipment and components—S/C subsystems.
4.1.1 External pressure is close to vacuum \[ h = 500 \text{ km}, \ P = 10^{-7} \text{ Pa} \ (1 \text{ Pa} \approx 10^{-5} \text{ atm}) \]

Impact: outgassing, change of material strength. Protection: hermetic sealing, application of special materials, filling by the inert gas.

4.1.2 Temperature (no direct contact with Earth atmosphere, hence no heat convection as in Aviation)

The heat balance between S/C and space takes place exclusively because of the particles radiation. The main sources of external radiation are Sun radiation and Earth-infrared reflection (Earth albedo).

Impact: S/C temperature significantly depends on its orientation relatively to Sun and Earth. For example, for a small satellite \( m = 200 \text{ kg}, \) cube: \( 1 \times 1 \times 1 \text{ m} \), Temperature: Sun side +90°C, Space side -20°C, Earth side -10°C. There are extreme temperature gradients between S/C sides.

Protection: Special thermal design (thermos radiators and plates, materials, and painting) and Thermal Control System (TCS) (thermistors and heaters, convective ventilation) are applied.

4.1.3 Electromagnetic Space radiation (Special effects in South Atlantic region)


4.1.4 Disturbing influence of Earth magnetic field, residual atmosphere, and solar pressure

Impact: disturbing forces and torques, affecting satellite orbit and the attitude.

Protection: periodic orbital correction, demagnetization on ground, and minimization of the ballistic coefficient, effective attitude control.

4.1.5 Hard electromagnetic compatibility (EMC) conditions because of small volume for the accommodation

Impact: mutual electromagnetic interference.

Protection: appropriate allocation, screening.

4.1.6 Sun eclipse

Impact: A-Solar panels cannot be used, and satellite power is provided only by the on-board batteries that cannot be discharged during the eclipse period less the critical voltage. AODCS should minimize power consumption.

B-SS cannot be used for the attitude determination.

Protection: A-Application of the sufficient type of batteries, starting eclipse with previously charged prime and redundant batteries.

D-Switching during the eclipse period attitude determination method to another sensor that does not need Sun visibility (e.g., HS) and/or using the Momentum wheel for gyro stabilization of the satellite.
4.1.7 Gravity acceleration

The gravity field acceleration is decreasing with increasing of satellite attitude. It influences on the required velocity for the space flight in circular orbit at this altitude. This velocity can be calculated with the following formula [7]:

\[ V_0 = \sqrt{gR} \]  

where \( g \) is the gravity acceleration, \( R = R_e + h \) is the distance between satellite and the Earth center, \( R_e = 6378.137 \) km is Earth spherical model radius (equatorial), \( h \) is satellite altitude.

Gravity gradient torque \( T_g \) impacting on a cylindrical shape satellite attitude is as follows [7]:

\[ T_g = -\frac{3}{2} \omega_0^2 \left( J_e - J_p \right) \sin \alpha \]  

where \( \omega_0 = \frac{V_0}{R} \) is satellite orbital rate, \( J_e \) is satellite equatorial moment of inertia, \( J_p \) is satellite polar moment of inertia, \( \alpha \) is angle of deviation of satellite from local horizontal plane.

Earth gravity acceleration can be calculated with the following formula [7].

\[ g = \frac{\mu}{(R_e + h)^2} \]  

where \( \mu = \gamma M_e = 398600 \text{ km}^3/\text{s}^2 \) is Earth gravity constant, \( \gamma \) is Universal Gravity Constant, and \( M_e \) is the Earth mass. Calculated with this formula, gravity acceleration at the Earth surface is \( g(R_e) = 9.798 \text{ m/s}^2 \). The graph of the gravity acceleration calculated with (7) is presented in Figure 6.

4.2 Environmental tests

To verify that SCS does meet the environmental requirements, AODCS usually examines with special environmental tests.

Figure 6.
Gravity acceleration versa altitude \( g(h) \) (m/s²), \( h \) (km).
The following tests related to environmental conditions could be performed:

1. Thermo, Vacuum (TVAC-cyclic), and Humidity Category.
   a. For internal components: Temperature 20°C ±5°C for the unit with thermostat, but -60°C — +40°C — worst case of the thermostat failure.
   b. For external component, for example, -100°C — +150°C.

2. Pressure: h = 0 km - P = 1 atm (101 kPa); h = 20 km - P = 0.05 atm (5.06 kPa); h = 609.6 km- P = 4.74 · 10⁻¹² atm (4.8 · 10⁻¹⁰ kPa);

3. Electromagnetic compatibility and interface (EMC/EMI and magnetic cleanness) (depends on system accommodation and RF antenna patterns).

4. Radiation Hardness, Radiation hardness designators M, D, FG, P, L, H indicate unit capability to withstand to a certain radiation dose, for example, M = 3 · 10³ rad (Si);

5. Mechanical launch impacts: sinusoidal/random/acoustic vibration in a certain range of frequencies, static, and shock (depends on planned launcher).

4.3 Space qualification (SQ) functional test (FT)

Usually, SQ FT is carried out in specially equipped for these purpose facilities by trained personal and highly qualified experts as the final part of system Assembling, Integration, and Testing Activities (AIT).

It must be mentioned that AIT activities should include this final functional test for SS Flight Model that should demonstrate its capabilities to perform in Space required functions after all other type of SQ (environmental) tests under the system have been performed.

In this SQ FT SS is completely assembled and integrated, as well as refined (calibrated). Specifically, in this test SS hardware (H/W) and software (S/W) working jointly should be verified. This test should finalize SQ procedures, preceding release of the Space Qualification Report (SQR), and declaring readiness SS for launch and operation in Space.

Unfortunately, in common practice due to many various reasons, SQ FT does not occupy the right place in a number of SQ tests. For example, for such important for any spacecraft system as Attitude Control System (ACS), this test often comes down to checking electric interface and right direction of rotation of the Reaction wheels (“polarity test”). Sometimes, such a superficial attitude to SQ FT leads to very stressful and even dramatic situations after the launch during SS operation in space. That is why many authors [20–22] draw attention SS developers to this problem and present some simulation tools and procedures to resolve it.

With regard to the satellite control system (SCS) and its components [9, 23], the main difficulty for FSQ is to model on ground orbital flight with relevant gravitation and magnetic field, and orbital motion. For these purposes, for modern
small satellite, very sophisticated test beds, based on three degrees of freedom air bearing tables, have being used [22].

Here, the author presents a different approach from System Dynamics Identification point of View [24]. This general approach allows to identify SS (in particular, SCS) dynamics in open control loop using currently commonly available for engineers Matlab/Simulink Identification Toolbox. It does not require complex test (control and verification) equipment. Mainly special laboratory emulators, activating SS sensors must be used additionally to conventional AIT SQ equipment (assembling stand, laboratory registration console for simulation radio link to satellite Tracing, Telemetry, and Control System (TTCS), power supply, installation devices, and mass property determination machine).

Looking at the problem of SS SQ FT from the point of view of System Dynamics theory, we can allege that if system has proper dynamics, previously verified with mathematical simulation (MSim), which meets design requirements, and its (structure and parameters) is validated with semi-natural simulation (SNSim); then, this system will be capable to perform expected functions in space, at least in some mission essential operation modes. The process of evaluation of system dynamics by the experimental way is named System Identification process [25]. Presently, identification methods have been developed to be practically used in many engineer applications. The most known and commonly used engineer tool for the identification is Matlab/Simulink Identification Toolbox (ITB) [26].

It is applicable for both cases; when system structure (mathematical model) is partly known and only unknowns are system parameters (mathematical model coefficients)—“gray box” case and when considered system is totally unknown —“black box” case. For both these cases, ITB allows to identify (estimate) system mathematical model. Only experimentally measured system input and output signals are used. The ITB adjusts the most suitable model estimate to minimize the difference between the output measured experimentally and its estimation, provided by the estimated model. Briefly, the essential elements of this identification are presented below. Let us consider an SS as a unit consisting of the hardware (HW) and the software (SW) components as presented in Figure 7.

From the System Dynamics point of view, this system can be characterized by its input $x(t)$, output $y(t)$ and some mathematical operation, determining system conversion from the output to the input

$$y(t) = [y(t)]$$

(8)

At the first approximation, many Aerospace devices and systems dynamic can be considered in the scope of Linear Time Invariant (LTI) dynamic system theory. In this case, (8) can be represented as follows:

![Figure 7. Space system unit.](http:/images/.../105084)
\[ y(t) = \int_0^t g(t - \tau)x(\tau) d\tau \quad (9) \]

where \( g(t) \) is system’s impulse characteristic response to the Dirac’s input impulse \( x(t) = \delta(t) \). Using Laplace transformation to (9), it can be represented as

\[ y(s) = G(s)x(s) \quad (10) \]

where \( y(s) = L[y(t)] \), \( x(s) = L[x(t)] = L[\delta(t)] \) are Laplace transformation of output and input signals and \( G(s) = L[g(t)] \) is Laplace transformation of system impulse function. In other words,

\[ G(s) = \frac{y(s)}{x(s)} \quad (11) \]

is the ratio of Laplace transformations of output to input signals.

In general case, LTI system transfer function can be expressed as the two polynomial ratios:

\[ G(s) = \frac{b_m s^m + b_{m-1}s^{m-1} + \cdots + b_1 s + b_0}{a_n s^n + a_{n-1}s^{n-1} + \cdots + a_1 s + a_0} \quad (12) \]

where \( b_i, a_j \) are constant polynomial coefficients, \( m \leq n \). Usually, (12) represents a stable system with the characteristic equation

\[ a_n s^n + a_{n-1}s^{n-1} + \cdots + a_1 s + a_0 = 0 \quad (13) \]

which roots \( s_{k1,2} = \text{Re}_k \pm j\text{Im}_k \) satisfy the following condition

\[ \text{Re}_k \leq 0 \quad (14) \]

Usually, for any designed SS assumable (before identification) transfer function \( G(s) \) for system unit, presented in Figure 4, is known from its design documentation. Identification experiment provides measured input \( X_m(t) \) and output \( Y_m(t) \) data (Figure 8) and the identification procedures used in ITB allows to estimate this function coefficients \( \hat{a}_i \) and \( \hat{b}_i \).

Theoretical ratio between the input \( x \) and the output \( y \) of a LTI system \( G(s) \) is (11). However practically, it takes place experimentally measuring input \( x_m \) and output data, distorted by some input \( V_i \) and output \( V_o \) errors

\[ x_m = x + V_i \quad (15) \]

and

\[ y_m = y + V_o \quad (16) \]

The difference between expected and experimental output signals is as follows:

\[ e = y_m - y = \hat{G}(s)x_m + V_o - G(s)x = \hat{G}(s)(x + V_i) + V_o - G(s)x = \\
= \left[ \hat{G}(s) - G(s) \right] x + \hat{G}(s) V_i + V_o \quad (17) \]

where \( \hat{G}(s) \) is estimate of system transfer function.

This difference (14) is used in ITB to tune (adjust) model coefficients \( \hat{a}_i \) and \( \hat{b}_i \) to minimize it so that the outputs \( y_m \) and \( y_e \) would coincide as much as possible.
It can be mentioned that such identification does not require simulating of system dynamic in closed feedback control loop configuration. To identify open-loop transfer function is enough, then closed-loop transfer function can be recalculated with the following formula [27]:

\[ W(s) = \frac{G(s)}{1 + G(s)} \]  

where \( W(s) \) is negative feedback control closed-loop transfer function, \( G(s) \) is transfer function of this loop in open state (assuming that feedback has unit transfer function \( F(s) = 1 \)).

This is important for SS and specifically for SCS because it does not require unique complex equipment to simulate space flight and closed feedback control loop formed by the SCS in it.

Basic ideas of such a simulation for the identification of transfer function of open loop of SCS are presented in Figure 6.

The flight model of SS is installed on laboratory AIT table and electrically connected to the Laboratory Control-verification console.

SS expected transfer function \( G(s) \) is known and should be verified with ITB, or in other words, its experimental estimate \( \hat{G}(s) \) should be identified.

SS is switched on in special Ground Test Mode (GTM) (Figure 9). Its power, reference, and control data \( D \) are supplied via special data link from the laboratory Control and Verification Console (CVC).

It is important to note that in GTM SS should use special reference data about its state in SQ facility: \( \Phi_0 \)—latitude, \( \Lambda_0 \)—longitude, \( h_0 \)—altitude, \( V_0 = 0 \)—velocity, \( B_0 \)—magnetic induction vector. Its input is physically activated with a kind of laboratory imitator (red arrow in Figure 6). SS input and output data \( X_m \) and \( Y_m \) are recorded in real time in the CVC. After the end of the experiment, these data are
reformatted in the form of mat. File and downloaded into the flash memory chip (FM in Figure 6) that using regular USB interface is connected to laboratory PC for the data post-processing in ITB. This ITB carries out the estimate of SS transfer function \( \hat{G}(s) \). If it is close to expected due the SS design function \( G(s) \), then we can allege that \( G(s) \) is verified by SQ FT.

**Examples of identification of basic dynamic units**

With purpose to validate identification method for SS SQ FT before performing seminatural simulations, some typical liner time-invariant (LTI) dynamic unit transfer functions were identified with mathematical (quasi-seminatural simulation). Some examples can be also found in [28].

The same methodology for this “quasi seminatural simulation” was used. At first, system was simulated without measured errors, idealistic (“clear” measurements) input \( X \) and output \( Y \) and its step response \( Y \) was received. After input \( X_m = u \) and output \( Y_m \) were distorted with superimposed Gaussian white noises, imitated measured errors and these signals were used for identification system dynamics (transfer function, step response, amplitude/phase frequency diagrams, characteristic polynomial roots).

**Example 1:** Simplest aperiodic system, first-order unit.

Given system is first-order dynamic unit that has transfer function.

\[
G(s) = \frac{1}{Ts + 1}
\]  

(19)

where \( T = 10 \text{ s} \) is system time constant.

Characteristic equation \( Ts + 1 = 0 \) root is \( s_* = -\frac{1}{T} = -0.1 \text{ s}^{-1} \).

Simulink block diagram of this system is presented in **Figure 10**.

This scheme allows analyzing the step response of the system without and with measured noise.

1a- Mathematical simulation

Step response of the system (16) without noise is shown in **Figure 11**.

1b-“Quasi semi-natural” simulation

Step response of the system (19) with noise is shown in **Figure 12**.

1c- Identification results

System (19) identification results are presented in **Figures 13–15**.
Estimated characteristic equation of the system (12) is \( \hat{T}s + 1 = 0 \) with the root \( s_* = -0.09 \), estimated Time constant is \( \hat{T} = \frac{1}{s_*} = 11.1 \text{ s} \).

Estimated transfer function of the system (19) is

\[
G(s) = \frac{1}{\hat{T}s + 1} \tag{20}
\]
Figure 12.
Step response of the system (19) with noise in measurements. \( X_m \) is input-blue, \( Y_m \) is output-red.

Figure 13.
Step response \( h(t) \) of the identified system (19).

Figure 14.
Amplitude \( A(\omega) \) and phase \( \varphi(\omega) \) diagrams of the identified system (19).
Example 2: Damped oscillator, second-order unit

Given system is second-order dynamic unit that has transfer function

$$G(s) = \frac{k}{T^2 s^2 + 2dT s + 1}$$ (21)

where $T = 10$ s is system time constant, $d = 0.707$ is specific damping coefficient, $k = 5$ is static control gain.

System characteristic equation is $T^2 s^2 + 2dT s + 1 = 0$. Its roots are $s_{1,2} = -0.0707 \pm 0.0714 j$.

Simulink block diagram of this system is presented in Figure 16. This scheme allows analyzing the step response of the system without and with measured noise.

2a—Mathematical simulation
Step response of the system (21) without noise is shown in Figure 17.

2b—“Quasi semi-natural” simulation
Step response of the system (21) with noise is shown in Figure 18.

2c—Identification results
System (21) identification results are presented in Figures 19–21.

Estimated characteristic equation of the system (14) is $\hat{T}^2 s^2 + 2\hat{d}T s + 1 = 0$. It has two complex roots $s_{1,2} = -0.0622 \pm 0.0688 i$.

Estimated transfer function of the system (21) is

$$G(s) = \frac{\hat{k}}{\hat{T}^2 s^2 + 2\hat{d}T s + 1}$$ (22)

where estimated Time constant is $\hat{T} = 10.78$ s, specific damping coefficient is $\hat{d} = 0.6707$, static control gain is $\hat{k} = 4.99$. 

Figure 15.
Root of the characteristic equation of the identified system (19).
Example 3: PID controller

Given system is Proportional, Integral, and Damping controller that has transfer function

\[ G(s) = \frac{k_p + \frac{k_i}{s} + k_ds}{T^2s^2 + 2ds + 1} \]  

(23)

where the control gains are as follows: \( k_p \) is positional gain, \( k_i \) is integral gain, \( k_d \) is damping gain.

Figure 16.
Simulink block diagram of second-order unit.

Figure 17.
Step response of the system (21) without noise. X is input-blue, Y is output-red.
Practically, ideal differentiation assumed in (23) cannot be realized. Realistically, (23) should be represented as

$$G_c(s) = k_p + k_i \frac{s}{\tau s + 1}$$

where $\tau$ is a small time constant. In other words, the differentiation with filtering takes place and $\omega_c = \frac{1}{\tau}$ is the cut frequency (bandwidth) of this differentiating filter.

Let us given, that $k_p = 0.1 Nm/rad$, $k_i = 0.03 Nm/rad/s$, $k_d = 0.05 Nm/rad \cdot s$, $\tau = 10s$ (assuming that the input of this controller is an angle in radians—rad and output is the control torque in Newton meters—Nm).

Figure 18.
Step response of the system (18) with noise in measurements. $X_m$ is input-blue, $Y_m$ is output-red.

Figure 19.
Step response $h(t)$ of the identified system (21).
After algebraic transformation (24) can be represented as follows

$$G(s) = \frac{(k_p \tau + k_d)s^2 + (k_p + k_i \tau)s + k_i}{s(\tau s + 1)}$$

(25)

Figure 20. Amplitude $A(\omega)$ and phase $\varphi(\omega)$ diagrams of the identified system (21).

Figure 21. Roots of the characteristic equation of the identified system (21).
or in the numerical form

\[ G(s) = \frac{1.03s^2 + 0.6s + 0.05}{s(10s + 1)} \]  

(26)

Denominator of (23) \( s(10s + 1) = 0 \) has following roots (poles): \( s_1 = 0, s_2 = -0.1 \) and the nominator \( 1.03s^2 + 0.6s + 0.05 = 0 \) following (nulls) \( s^*_1 = -0.482, s^*_2 = -0.101 \).

Simulink block diagram of this PID controller is presented in Figure 22.

3a—Mathematical simulation
Step response of the system (24) without noise is shown in Figure 23.
3b—“Quasi semi-natural” simulation
Step response of the system (24) with noise is shown in Figure 24.
3c—Identification results
Identification results of the system (24) are presented in Figures 25–27.

Estimated transfer function of the system (23)/(24) is

\[ G(s) = \frac{1.03s^2 + 0.0618s + 0.004929}{s^2 + 0.0986s + 1.289 \times 10^{-16}} \]  

(27)

Formula (27) can be approximately represented as follows:

\[ \hat{G}(s) \approx \frac{1.0487s^2 + 0.6103s + 0.05}{s(10.142s + 1)} \]  

(28)

Figure 22.
Simulink block diagram of PID controller.
Denominator of (28) \( s(10.142s + 1) = 0 \) has following roots (poles): \( s_{+1} = 0, s_{+2} = -0.0986 \) and the nominator \( 1.0487s^2 + 0.6103s + 0.05 = 0 \) following (nulls) \( s_{-1}^* = -0.4818, s_{-2}^* = -0.1008 \).

Comparing coefficients (28) with (24), we can determine PID control gains and the time constant

\[
k_p = 0.1033 \text{Nm/rad}, k_d = 0.025 \text{Nm/rad}/s, k_i = 0.05 \text{Nm/rad} \cdot s, \quad \tau = 10.142s
\]

Comparing identification results obtained with “quasi seminatural” simulation with real mathematical model, we can see that for all three considered above examples, identified model takes place in close coincidence between real and identified models that show effectiveness of application of ITB for identification purposes.
Presented above examples show that Matlab Identification Toolbox, at least for simple LTI units, can be successfully used for identification their dynamic characteristics. Further studies should verify mathematical simulation with real physical experiments (semi-natural simulation), involving system hardware. More complex, nonlinear, and nonstationary systems also should be studied.

Related to these tests methodology, requirements and standards can be found in [8, 14, 18, 19, 29].

Practically, implementation of the presented above functional Space Qualification Test can essentially decrease of many unexpected flight anomalies that occurred and were learned during operation of first Canadian SSAR satellite Radarsat-1 (Figure 28).
5. Conclusion

This chapter (Part II) continues (see Part I in [9]) to present a basic ground for Satellite Control System to integrate it with such a payload as satellite on-board SSAR RLS. Namely, it presents SCS: Control Modes, Power, Interface, and Testing. This material presented from the point of view of integration both systems into the satellite bus, considering satellite as the integration platform and seeing the satellite designer as the Prime Contractor, responsible for Earth observation mission.
successful execution in Space. However, in some cases the payload (for example, SAR) provider can perform the integration function as well. Of special interest can be, presented above, methodology of SCS Space Qualification (SQ) Functional Test (FT) that can be similarly applied to the remote sensing payload also and, finally, to the integrated system identifying its dynamic at the final stage of satellite Space Qualification program.

The chapter can serve to a wide pool of Space system specialists as an introduction to Satellite Control System development.

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